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PROCEEDINGS

WADC SYMPOSIUM

sponsored by
ARDC

FATIGUE OF AIRCRAFT STRUCTURES

WRIGHT AIR DEVELOPMENT CENTER
11 - 12 - 13 AUGUST 1959

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WADC TECHNICAL REPORT 59-507

PROCEEDINGS OF THE SYMPOSIUM
on
FATIGUE OF AIRCRAFT STRUCTURES

AUGUST 1959

WRIGHT AIR DEVELOPMENT CENTER
AIR RESEARCH AND DEVELOPMENT COMMAND
UNITED STATES AIR FORCE
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

1,300 -- October 1959 -- 12-345

PREFACE

The first USAF Symposium on Fatigue of Aircraft Structures, sponsored by the Air Research and Development Command, was held under the auspices of the Wright Air Development Center in Dayton, Ohio, 11 - 13 August 1959. The purpose of the symposium was to emphasize the importance of structural fatigue in current aircraft and future aerospace vehicles to management and technical personnel of the airlines, airframe and metal industries, universities, non-profit institutions, and Government agencies.

The proceedings are set forth in this report in the same order of presentation followed during the course of the symposium. They consist of twelve addresses by outstanding men in the field of technical management and thirty-five technical papers by well-known experts in the engineering and research areas. The latter are grouped by subject into five basic categories. The affiliation of the speaker with his organization is given in the head of each paper. Postal addresses of speakers, authors, and session chairmen may be found in the back of this report.

Open forum discussions were held following each session. Session chairmen and speakers answered both written and oral questions originating from the floor. The forum discussions and impromptu questions and answers were recorded by a court stenographer and are included at the end of each session. They were edited by the Technical Editing Committee only to the extent necessary to present the material in a clear and concise form.

During the course of the symposium it was announced that the proceedings would be published within thirty to sixty days. This short period allowed no time for individuals concerned to proofread the final copy of speeches, impromptu questions and answers and forum discussions. Every effort has been made to keep errors and omissions to a minimum and still allow publishing within the specified time period. If serious errors or omissions are noted, they should be brought to the attention of Mr. A. E. Muhlhauser, Wright Air Development Center, WCLSym, Wright-Patterson Air Force Base, Ohio, by 1 November 1959. Necessary errata sheets will be prepared and distributed.

The success of the symposium is primarily due to the fine contributions of the speakers, session chairmen, and authors of technical papers. The Symposium Committees gratefully acknowledge these contributions and express their appreciation. The Symposium Technical Committee likewise expresses its appreciation to the fatigue specialists of the Aeronautical Research, Materials, and Aircraft Laboratories for their invaluable assistance in selecting speakers and technical papers, and for the work leading to the successful Symposium-Tour of typical Wright Air Development Center facilities.

John P. Taylor

JOHN P. TAYLOR
Colonel, USAF
Symposium Deputy Chairman

Abstract

The Symposium Proceedings on Fatigue of Aircraft Structures are presented in this report. The General Session speeches point out the impact of fatigue on Air Force operations and present in detail the aircraft and personnel losses and related costs due to fatigue failures; the resulting supply, maintenance and logistic difficulties associated with fatigue problems; the effects of fatigue on the retaliatory potential of the military commands; and the philosophies and approaches of the military, government agencies and trade associations toward solving the fatigue problem.

The papers of the technical sessions present the detailed aspects of the fatigue picture. The seven papers of the first session deal with the analytical approach to the fatigue problem discussing all phases of analysis from statistical aspects of a load spectrum to stress concentration factors. The nine papers of Session II-A center about the problems of recognizing, measuring, and analyzing the aircraft loadings which produce critical fatigue damage. Papers of Session II-B delve into the material aspects of fatigue, covering such topics as the proper selection of materials for fatigue resistance, the importance of process control, high temperature considerations, and progressive damage theories. Session III papers expand on criteria to be used in the design of air and space vehicle and research objectives. The problems associated with simulation of fatigue load conditions in the laboratory are discussed in the papers of Session IV. Some of the problems included are: complete vehicle testing, sonic fatigue testing, fail safe test methods, elevated temperature testing and equivalent test spectrums.

The six forum discussions, impromptu questions and answers and a general summary on the results of a questionnaire are also presented in the Proceedings. The summary explains the use of the questionnaire, its background, purpose and results.

Publication Review

The publication of this report does not constitute approval by the Air Force of the findings or conclusions contained herein. It is published only for the exchange and stimulation of ideas.

FOR THE COMMANDER:

Harvey P. Huglin

HARVEY P. HUGLIN
Colonel, USAF
Deputy Commander
for Development

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General Curtis E. LeMay, Air Force Vice Chief of Staff stated, on November 19, 1958, "The successful accomplishment of this program (the Aircraft Structural Integrity Program) is vital to the assigned mission of the Air Force and requires the complete and active support and cooperation of all Staff and Command levels. The total aircraft structural integrity program encompasses all first line aircraft and therefore, warrants support at a priority level higher than that established for any other aircraft involved." Publication of this statement was the result of a concerted six months assault on the fatigue problem and cleared the field for an all-out Air Force wide attack on the problem of metal fatigue.

The opening assault was generated earlier by General Thomas S. Power, Commander-in-Chief, Strategic Air Command, in April 1958 when he stated, "I believe it absolutely essential that we learn what we bought, in terms of service life, in the B-47 and B-52 and what we will buy in the B-58 and B-70." An immediate check by the Air Research and Development Command with the manufacturers of these aircraft types indicated that fleet service life estimates made within a period of six months varied by as much as 100 percent. In other words, the state of the art did not permit accurate aircraft service life estimates at that time.

Serious structural fatigue incidents in high performance aircraft had been occurring for more than a decade. The Martin 202 in 1948, the F-89 and B-36 in 1952, the F-84 in 1953, the Comet in 1954, the F-86 in 1955, and the F-101 in 1958 are typical examples. Quick action in the form of expedited investigation and retrofit programs usually sufficed to correct the situation in each type of aircraft. Then, just as it began to appear that the situation was under control and that a degree of compatibility had been reached between aircraft structural dependability and metal fatigue, the problem was again brought into sharp focus. This time the critical factor was a series of catastrophic fatigue failures in the B-47 fleet early in 1958.

Because of the urgent necessity for an operational B-47 fleet in national defense planning, an accelerated B-47 rehabilitation program was instituted immediately by the Air Materiel Command. As a result of a series of full scale cyclic-fatigue tests, additional retrofixes were designed and installed and an estimated service life for the B-47 fleet as a whole was established. Through the extensive cooperation of all interested organizations, the B-47 fatigue situation appears to be under control at this time.

General LeMay's remarks, quoted earlier, have resulted in increased support of the Air Force wide multi-million dollar effort already under way. The Air Research and Development Command and the Air Materiel Command in particular have drawn all elements of this effort into one major program and published a suitable document entitled "ARDC/AMC Program Requirements for the Structural Integrity Program for High Performance Aircraft." This document has received not only the approval of the Commanders of the two Commands but also of the Vice Chief of Staff, USAF.

In general the joint program comprised a series of inter-related steps which broadly accomplishes the following objectives:

1. Improved understanding of typical operational environments on high performance aircraft structures,

2. Improved analytical and test methods for predicting fatigue life of materials and service life of airframe structures,

3. Improved structural fatigue criteria reflecting extensive theoretical, laboratory, and field experimentation to include temperature and sonic effects.

In support of this laboratory, flight, and analytical effort, an initial basic research program was deemed necessary to get at the fundamental nature of the fatigue mechanism in vehicle structure. In order that such a program have maximum flexibility to provide greater insight into the fundamental causes of the fatigue phenomena, it was suggested that the next logical step would be the mobilization of national effort in this area.

To accomplish this goal and to expedite efforts leading to structural fatigue prevention, a Symposium on Fatigue of Aircraft Structures was planned for 11-13 August 1959, in Dayton, Ohio, of interested industrial, educational, research, and Government representatives, for the purpose of accomplishing the following specific objectives:

1. To acquaint interested agencies with the current aircraft structural fatigue problem,

2. To promote increased interest in the technological field of structural fatigue prevention,

3. To stimulate thinking on potential fatigue problems of future aero-space vehicles,

4. To foster exchange of technical information between experts in specific critical areas of fatigue,

5. To encourage coordination of related technical efforts by interested groups in the national and international scientific engineering community.

Just how long it will take to solve the fatigue problem depends to a large extent on the degree to which the symposium attendees are able to accomplish the indicated objectives. Judging from the enthusiasm indicated by the number and the reaction of the attendees, this symposium represents a major stride in the right direction.

INTRODUCTORY REMARKS

BY

COLONEL HARVEY P. HUGLIN

Deputy Commander for Development
Wright Air Development Center

As of now, the first symposium on "Fatigue of Aircraft Structures" is officially open. I am your chairman for this meeting. Before proceeding further, I would like to introduce some WADC people who do not appear in the program. These persons have been very instrumental in forwarding the WADC part of the Air Force program to determine a solution to the problem of preventing fatigue in aircraft structures.

First, I would like to introduce Colonel Fred Ascani, Director of Laboratories at WADC. Through his excellent efforts, this program was planned and coordinated with our various laboratories. Next, Colonel Randall Keator, Chief of the Aircraft Laboratory, without whose support, Colonel John Taylor and the rest of us could never have set up this symposium in the short period of three months. Next, Mr. Hugh Lippman, Technical Director of the Aircraft Laboratory, an old timer at WADC, and Mr. Bill Miller of the Aircraft Laboratory Structures Branch; both of these gentlemen are well known to most of you for their association with Air Force fatigue problems. These men acted in an advisory capacity on formulating this symposium.

Finally, I would like to introduce Colonel Wes Anderson, our new Chief of the Materials Laboratory, and Colonel E. C. Mallary, Chief of the Aeronautical Research Laboratory; these men play an important part in the fatigue picture by heading the laboratories also deeply involved in the structural fatigue problem.

Now, gentlemen, without further ado, I give you Major General Stanley T. Wray, Commander of the Wright Air Development Center.

WELCOME ADDRESS

By

Major General Stanley T. Wray, Commander

Wright Air Development Center

Most of you will remember from your early boyhood a poem by Oliver Wendell Holmes about a one horse shay:

"The wonderful one-hoss shay
Built in such a logical way
It ran 100 years to the day."

The wheels were just as strong as the spokes and the spokes were just as strong as the yoke and the sills were just as strong as the yoke. In fact, this wonderful contrivance <u>was</u> built to last. Its design was such that one hundred years to the day the old one horse shay ... (At this point General Wray collapsed a 2/3 scale model of the one-hoss shay located on the speaker's platform).

Now no military weapon system can be expected to last "a hundred years to a day," - but the basic principle of having sufficient structural integrity built into all of its parts to permit it to just live out its useful operational life is a goal that I think all of us must strive to achieve.

It is on this note that I welcome you to our first Fatigue Symposium.

Out latest effort in connection with improving structural integrity of aircraft, which has been intense, results from the series of fatal crashes which occurred early in 1958. However, the roots of this problem go so deeply into the basics of airpower and spacepower that we felt it mutually valuable to survey the total problem with you. As you know, within the last three years there has been a series of world-wide technical symposia in which various technical aspects of high temperature, fatigue, and materials have been extensively discussed. However, to my knowledge, this is the first time that a group such as this, composed of management and technical experts, has assembled from the academic world and from interested airlines, airframe manufacturers, metal companies, and various government agencies to discuss how we are to meet the total challenge of these fields in relation to practical flight vehicles of the future.

The prevention of catastrophic failure due to fatigue, during the useful life span of flight vehicles, is a goal that must be recognized as one of prime importance to the Air Force mission. As an arch enemy of flight safety and national security, the fatigue problem must be attacked on all fronts, from the basic and applied research areas to those of design, manufacturing, maintenance, inspection, and ultimately, to the operational utilization of the flight vehicle.

Our national effort in fatigue has lacked unity of purpose and direction. I would like to emphasize that our past concepts in this field are not sacrosanct. We must extend our

thinking to find new methods of approach to the problem. In short, gentlemen, we must figuratively redesign our one-hoss shay, take up its new shafts and pull together in the direction of our goal, seeking out and crossing new frontiers, as necessary, to accomplish our objective.

At the end of today's session, Colonel John P. Taylor will present you with a questionnaire which will give each of you an opportunity to express your thoughts on what needs to be done in your specialized field of activity. This simple questionnaire, which will not take much of your time to complete, will be collected tomorrow morning at the start of the session. These questionnaires will be studied by a team of experts from this assembly in order to present to you, prior to your adjournment, the results of what you think should be done, on a combined basis, for resolving the structural integrity research problem.

We have already informally discussed, with some of the representatives of government agencies present here and with the representatives of Aerospace Industries Association and Air Transport Association, how your suggestions can best be used so that each sponsoring group from government, industry, and the academic world can contribute its effort to the over-all program. In giving us your best estimate of the situation in your own technical speciality you will provide us with the guidance necessary to proceed so that structural integrity will not become a seriously limiting factor in the future design of flight vehicle structure. Should you wish to extend your comments after this symposium we would be glad to give them careful consideration and share them with the other agencies that are represented here.

Now, we must recognize that the solutions to the structural integrity problems we face will cost us much time and energy, particularly in the fields of basic and applied research. Further, the research accomplished will be vital not only to aircraft but also to guided missiles and aerospace vehicles. It is essential therefore that the fatigue problem be studied and resolved on a wide front so that we can prevent unnecessary duplication in effort.

Today's program has been designed to review the philosophy, policies and future planning of those groups most directly involved in the fatigue area. I am particularly pleased to have here such outstanding speakers from the management and technical sides of the house. I would like to exhort these people to speak frankly. Let's have discussion that is frank and easy, but let's not have arguments - in other words, let's not harass the speaker.

Everybody does not agree that there is a fatigue problem, but I am sure in my own mind that there is; and I think we're going to have some very strong statements made on the platform. I hope that you will take them as being objective remarks. Take them home with you and think about them.

Our first speaker is familiar with the problems of fatigue and structural integrity in their many aspects. He learned a lot of this when he was here at Wright Field a long time ago, and even more in his recent activities as Air Force Ballistic Missile expert. His current position as Commander, Air Research and Development Command, I am sure, will make him acquainted with many more of them. I have the honor and pleasure of presenting to you, Lieutenant General Bernard A. Schriever, Commander of the Air Research and Development Command.

KEYNOTE ADDRESS

By

Lieutenant General Bernard A. Schriever, Commander

Air Research and Development Command

General Wray, Colonel Taylor, all the people who are responsible for setting up this symposium, and the guests here today, I am glad to be here. I would like to say just a few words about the fatigue problem because it certainly is, in my opinion, a very vital one to the Air Force - not only the Air Force, but I think commercial aviation may be benefited as well by this meeting.

Structural fatigue in aircraft and missiles is a dull subject to laymen, but the engineers, scientists and technicians who deal with it daily realize the importance of the subject. We also realize that we must solve many of the problems associated with aircraft and missile fatigue, if we are to maintain a posture of positive military security.

We first encountered the problem in a big way in the B-47. We were faced with a new era in aircraft design. Here was the backbone of the Air Force retaliatory power growing old long before its time. All the usual precautions had been taken to insure a dependable weapons system. Yet, the airplane fell far short of its expectations.

Data gathered during the early phase of the airplane's life supported theoretical analyses that the aircraft's structure was adequate for its intended use. However, when the plane's mission changed for the purpose of maintaining maximum operational flexibility, structural failures followed. In all fairness to the design engineers, I must mention that it is not always feasible to restrict a military aircraft to design use only. We know the importance of new low altitude missions which come along, and the problem of loading the aircraft for longer ranges. It was following the establishment that the B-47's failures were due to fatigue that a complete fatigue analysis was conducted. Results revealed that the low altitude maneuvers - low altitude missions and long range missions had aged the structure much faster than had been anticipated. Two obvious steps then had to be taken - first, rehabilitate the B-47, and second, insure that nothing like this could happen again.

In keeping with the first objective, the Air Force initiated a full-scale laboratory test program on the B-47. Three specimens were tested at three separate facilities. As a result, retrofixes were designed and installed and an estimated service life was established.

We anticipate that the installation of this modification, plus a specially designed inspection procedure, will eliminate the serious B-47 fatigue problem. When the final cost is tallied, it will vividly illustrate the advantage of foresight over hindsight as an approach to the solution of fatigue problems.

In this regard, we are pursuing our second objective in which we inaugurated a flight load survey project to provide comprehensive information about gust environment below 1,000 feet. The information thus gathered will aid materially in preventing any recurrence of the B-47 type of problem on other aircraft.

In addition to this laboratory and flight effort, a preliminary fatigue research program was started. This particular program may not result in complete solution of the fatigue problem, but its design as a basic program with maximum flexibility should provide greater insight for further research work.

We all know that fatigue in aircraft was a problem as early as the days at Kitty Hawk. Failure of a strut fitting and warping-wire pulley bracket on a Wright brothers' airplane at Fort Meyer, Virginia, in 1908, was attributed to what was then referred to as "crystallization." I remember that we used to refer to it as that here at Wright Field in the late '30's. Now this is fatigue as we know it today. Nuisance failures of lift and landing wires and wire fittings on early aircraft such as the Curtiss NBS-4 and the JN-4 were chargeable to repeated stress and vibration. Aside from engine components and propellers, however, fatigue did not become a major problem until after World War II. And no doubt this was because aircraft structural design up to that time was generally conservative.

Performance of today's aircraft, of course, has made fatigue a very major problem. Unlike aircraft, missiles are usually thought to be designed and fabricated to fulfill their own basic design criteria with no subsequent changes in structural load factors or design application. Thus we might be inclined to minimize structural fatigue factors in missiles but here, too, they are ever present.

Structural components of missiles are subjected to repeated stresses during transportation and erection. I might point out here that in some of our failures at Cape Canaveral we've worried much about the transportation problem, and we have made very detailed analyses of all the parts that were subjected to vibration during transportation. I can't say to date that we have attributed any failures to this, but it is something that we certainly worry about. We are also concerned about the captive and flight testing phases as a result of cryogenic and pressurization effects from loading and unloading propellants. From the information available, there has been no direct case of structural failure in flight of Atlas, Titan, and Thor missiles due to fatigue. But were there such failures, it would not be surprising if they could all be traced to damage incurred on the ground.

Now, I would like to look back just a little bit and also to look ahead. Much work has been done in recent years. As early as 1953 we established requirements for spectrum type fatigue testing of aircraft under carriage subsystems, and since then, we have been slowly working toward fatigue testing of complete aircraft. Considerable progress also was recorded in solving temperature and sonic problems and in the investigation of material properties. Yet, in spite of a gradually intensified effort during the last 30 years, it is obvious that a completely satisfactory solution has not yet been found.

Gentlemen, as you have shown by your enthusiastic attendance, fatigue is of increasing general interest. The B-47 incident illustrated that fatigue is of grave importance. Success or failure of our nation's defense effort may very well depend on the progress we make in solving this problem completely. Structural fatigue, admittedly bad, can be expected to worsen unless steps are taken immediately to eliminate the problem.

In the past, fatigue problems centered in aerodynamic vehicles. Today, as we contemplate aerospace operations, we are confronted with structural considerations of a completely new variety, such as cosmic ray and meteorite bombardment and oxygen-free environment. Very high strength-to-weight ratios will be required in conjunction with almost complete elimination or containment of adverse temperature and creep effects. In other words, future air and spacecraft will encounter increasingly severe environmental conditions and will necessarily be more stringently designed.

It would be unfair to those who have labored long and hard in this area to say that our present problems result from lack of foresight by those involved. If it weren't for the many contributions made by dedicated persons and organizations, our present problems would be immeasurably greater. The big question now is: What must be done to quickly and finally solve this fatigue riddle? That, in a sense, is why we are here today.

We can be sure of one thing. Our past efforts, though good, are not good enough. As the first step in remedying this situation, the Air Force has given fatigue investigation a very high priority. In fact, the structural fatigue problem was considered sufficiently serious by General LeMay that he stated on 19 November 1958 (and I quote), "The successful accomplishment of this program is vital to the Air Force's capability to perform its assigned mission, and requires complete and active support and cooperation of all staff and command levels of the Air Force organization. The total aircraft structural integrity program encompasses all first-line aircraft, and, therefore, warrants support at a priority level higher than that established for any individual aircraft involved."

I think it is obvious that this program will receive extensive high level management attention.

Some of our research has already been redirected. Our working force has been reoriented and consolidated. Sufficient money has been provided to start the research considered to be of basic necessity.

The need for new structural fatigue design criteria is critical. The importance of this effort and the teamwork necessary to pursue it to a successful conclusion cannot be overemphasized.

In closing, I should like to impress upon you gentlemen that the Air Force alone cannot hope to solve the problem of fatigue nor sponsor all the research required to attain that goal. Private industry's assistance and participation are urgently needed. What we seek are major breakthroughs all along the line. Dramatic and revolutionary progress will insure us positive military superiority and national security. I am quite confident that we in the Air Force can count on your cooperation. Thank you.

THE IMPACT OF METAL FATIGUE MISHAPS
IN THE U. S. AIR FORCE

By

Major General Joseph D. Caldara

Deputy Inspector General for Safety

I had a lot of clever stories I will not be able to tell you because they are dirty and there is a lady in the house. That is not part of the written speech; I really didn't have any, but I always take advantage of a gal when I can.

But let us proceed to a discussion of aircraft accidents - like everything else, the cost of these accidents has been steadily increasing. A year ago the average aircraft accident cost the Air Force and you people as taxpayers (I am a taxpayer too, of course) somewhere in the neighborhood of $400,000. For the first six months of 1959, the average aircraft accident cost you and me 537,000 bucks. It would be all right if these were only beans, but they are dollars and most of the cost is unnecessary. Let us take a look at some pictures for a couple of minutes.

In Figure 1, I'm taking the period of 1947 to 1959. The reason for this is that it is the period when the Air Force changed over to jets. First, we changed to jets in tactical units. Then we went into all supersonic jets in the fighter units. The accident rate dropped in this time period from 44 to 9.3 - I'm talking about major accidents per hundred thousand flying hours.

There is a grim paradox to the total number of accidents we've had this year. We see from Figure 2 that in 1947, some 35% or about a third of the aircraft involved in major accidents were destroyed. Eleven years later or now, 69% of the aircraft involved in major accidents are being destroyed - almost double the figure for 1947.

If that wasn't shocking enough, let us get to the humanities of this thing. As Figure 3 shows, in 1947, 17% or one out of every six pilots involved in an accident was killed. Twenty-five years ago, one out of every thirteen involved in an accident was killed. Last year 28% or just about one out of every four pilots involved in an accident was killed. These figures mean that, while we have been successful in reducing the total number of aircraft accidents each year, we have not been as successful in saving the aircraft or saving the pilots involved. In other words, as the equipment performance increased, the requirements on the pilot and crews went up with the result that accidents are more costly in blood and treasure.

Now let us translate this to combat potential because that is what we deal with in the Air Force (Figure 4). In the first six months of 1959 we graduated 1,444 pilots. While we did this we were busily engaged in killing 110 and another 55 suffered major injuries. Most of the latter will never go back to flying, so that one out of every nine of the new graduates replaced someone who had been killed or injured. Thus, 11% of these new graduates can be considered lost. Now the significance of this, and I'm not mad at the new graduates, goodness knows, is that they don't have the combat potential of the bomb wing commander who

was killed last year or of the fighter wing commander who was killed in the first half of this year. Both of these commanders had World War II experience. A new pilot cannot possibly get to this combat potential prior to the next conflict. This should impress upon you gentlemen the importance of accident prevention.

Next, we shall look at the number of aircraft affected. In the first six months of 1959, the Air Force accepted 705 aircraft. We destroyed 269, of which 153 were jet fighters, the equivalent of 6 squadrons; 26 were bombers, the equivalent of 1.7 squadrons; and 90 were miscellaneous type aircraft. We destroyed 38% as many aircraft as were produced. This means that the aircraft industry was only 62% effective because of our accidents. Four out of every ten aircraft we received in the Air Force last year went to replace those that we rolled up in a ball. Since 1950 we have killed 3,474 pilots and we have destroyed 7,062 aircraft. That is a bigger Air Force than any country in the world has except the United States or the Soviet Union and we have thrown it away. Two-thirds of it should never have happened. In fact, none of it ever should have happened. Getting back to the economics of accidents for a moment, three billion, one hundred and forty-eight million, four hundred and twenty-eight thousand dollars worth of equipment was lost, and no personnel training costs are included. I am talking about hardware. Three billion dollars worth! What would three billion dollars do for ARDC in its research program on metal fatigue? They would have contracts coming out of their ears.

Now we should have some causes before we get to the effects (Figure 5). Here I'm talking about "all causes" of a major accident. The reason I am using "all causes" is that an aircraft accident is never the result of a single dramatic or catastrophic event. It results from a series of unhappy coincidences that, collectively, result in a major accident. Pilot factors over the period of time from 1954 to 1957 decreased from 57% to 53%, and in the first six months of 1959, this factor is up to 56%. The materiel factor increased from 28% in 1954 to 40% in 1957, and to 47% in 1959. This is what I want to talk about!

When I speak of materiel failure, I am not implying that our materiel has suddenly gone to pot (Figure 6). I am saying that the same reliability today that we had ten years ago brings about more accidents because of the higher performance of the aircraft. Power plants accounted for 75% of materiel failures in 1958. This is up to 76% in 1959. Landing gears - 21% in 1958, 22% in 1959. Airframes - 4% in 1958 to 5% in 1959.

Next, I will give you a picture of the broadness of metal fatigue type accidents in the Air Force. We will start with early jet fighter mishaps (Figure 7). I shall skip most of the data and point out that we had 67 major accidents, 13 minor accidents, and 39 incidents. Of the 67 major accidents, the power plant accounted for 56.

Airframes did not account for many of the accidents mentioned above, but we have a shining example of the airframe problem in the early jets in Figure 8. This is the attachment bar for the outer wing of the F-89. It was originally designed in such a manner that we had very small radii in the attachment area adjacent to the bolt hole. The grain of the metal ran in such a direction that, when a fatigue failure started, it ran with the grain. The correction for this was very simple - enlarge the radii around the bolt hole to reduce the stress and change grain direction. This is no secret to design engineers, but they still do it. The F-89 was designed, goodness knows how many years ago, and we still have aircraft coming out in the Century series with the same type design. What did we do to correct

the condition? We enlarged the radii, changed the grain direction in the material and solved the problem. This is one job where I have developed perfect eyesight - and even better - after four and a half years in accident investigation, I have 20-20 hindsight! This, I highly recommend!

Now let us proceed to the Century series aircraft. So far as shown in Figure 9, we have 6 major accidents, 1 minor accident and 16 incidents that are attributable to metal fatigue accidents. Remember our Century series inventory is not as large as some of the earlier inventories.

Another category of aircraft I want to discuss is the non-jet bomber (Figure 10). The reason is that people in our Air Force have a tendency to ignore them. In non-jet bomber mishaps, we have as a result of metal fatigue, 12 major accidents, 2 minor accidents, and 12 incidents. This is something we found out in the B-36 - may it rest in peace - for that airplane was a flying machine of the first order. In the landing gear, there was a main tension section which was threaded - a very sharp angle thread - and failure due to metal fatigue occurred after only 19 landings on one B-36, and after 29 landings on another (Figure 11). The fatigue occurred in the last thread. The only fix on this whole assembly was to redesign the screw threads in this area to relieve the high concentration of stress. Yet we have a Century series fighter with a landing gear that is failing today due to this same design mishap, the same lack of consideration of stress concentration in an area of basic design.

Now you've heard General Schriever and General Wray refer earlier today to the B-47. So far, in our jet bombers, we have had 10 major accidents, 3 minor ones, and 41 incidents attributable to metal fatigue (Figure 12). These were disastrous type accidents, not only from a cost standpoint, but from the crew loss standpoint. Now let us get back to the B-47. There was fatigue failure of the lower skin aluminum alloy plate, through the bolt holes in the vicinity of the wing body attachments, as shown in Figure 13. We had the Tulsa, Homestead and MacDill accidents. The conclusion that metal fatigue caused these accidents was not easily arrived at. There are still certain dissenting voices debating whether this was a bona fide case of metal fatigue. Nevertheless, the condition seriously compromised the combat potential of your Air Force, and thus it had a definite impact on the free western world's capabilities in defense.

Lest you think that only jets have problems from metal fatigue, I have summarized cargo mishaps in Figure 14 - 18 major accidents, 2 minor ones, and 11 incidents have occurred in our cargo aircraft. Few people recognize this as a metal fatigue problem. The power plant was responsible for 12 out of the 18 accidents. These plants suffer fatigue in jet turbines as well as the jet engines. These accidents are serious.

The picture on trainer mishaps is shown in Figure 15. Even the trainers have accidents - I say "even"- good heavens! Look at the T-birds - 24 out of the 38 accidents were in a T-bird; 20 out of the 24 were traceable to power plant failure in the T-bird - what happened? We had blade and bucket failure problems in the early engines in the T-bird. We resolved it. We resolved it so well that the engineers said, "Well, this turned out so good, why not fix it a little better by adding one little thing." With this little fix, they managed to restore

to the engine the same lousy condition that it had in the first place. We had more engine failures in T-birds last year than we had in the first year we had the bird in the air. Now, what was involved? The design was good. The prototype of the particular piece of equipment was excellent. But in the process of manufacturing this thing, and we're talking about blades and buckets, something as well as somebody let go. There is more than one phase of this metal fatigue problem that affects you people as engineers. It is not all design.

Statistics on the Whirlybird and our light "off the shelf" aircraft are summarized in Figure 16. No one thinks of these as involving metal fatigue. Yet, we have had 9 major accidents and 5 incidents in our Whirlybird. Let that whirly thing get loose and there is, generally, a 100% catastrophe. This is very discouraging to whirlybird crews and I can understand why.

A summary of all the accident data is given Figure 17. The data shows that 160 major accidents, 27 minor accidents and 140 incidents were attributable to metal fatigue only. Now obviously, there were other factors involved in these accidents. Perhaps the pilot properly executing a flame-out landing could have avoided an accident from a power plant failure. Perhaps an auto-rotation in a Whirlybird would have avoided anything but an incident. Still, there have been 160 mishaps.

In the outline I had for this talk, I was requested to present some "impact." This is the impact. Remember, in the first six months of 1959 we have killed or have had major injuries to 165 pilots. This is impact. It is a particularly forceful impact on the pilots' immediate families. The impact, money-wise, has become more important since we must operate within a tight budget.

What is the philosophy on improvement of aircraft potential that has been developed from all the accidents including metal fatigue accidents? We have decided over the last four and a half years that the only thing we can do is educate the manufacturer including the design engineer; educate the commander, and educate the operator on maintenance. Now when you talk about a philosophy of education, a lot of people, particularly in the flying end of the business, and I happen to be in this business, sort of bow their necks and say, "Well, we just do not have the time for propaganda." Goodness knows this is true. No one has less time for propaganda than I have in this job. But, we must educate people on the situations and the requirements that exist. We must learn that one can't take a piece of equipment which is stressed to the absolute maximum of the materiel, change the environment, and then escape being forced to pay a price.

I've talked a lot about the design factor, the basic design problems, this business of small radii and stress concentration, design consideration of the grain material, and importance of design and engineering supervision. The philosophy of developing flight safety objectives requires everyone to take the step for which they are responsible. This must be done. The first great step of good detail design, as well as design engineering and design and engineering supervision is vested in you people. WADC is the responsible agency for the monitoring of the development of this equipment; and in turn, you people from industry are the responsible agency for the design of the equipment. I am not implying that people miss the mark deliberately, but, whether the mark is missed deliberately or accidentally, the point is that it is missed with the result that we have aircraft failure. So much then for

education on the engineering side of the problem. This first step of good design engineering and of proper engineering supervision must be taken home by you people here and discussed with your co-workers, whether you be military or civilian, whether you are part of the Air Force, the transportation industry, or part of the aircraft industry. This first step is your step.

There is another step which we must take within the Air Force, and when I say "we", I mean the entire safety organization. This step is the education of the commander. We can and must face up, in the Air Force, to several things. If we demand that an aircraft be designed light enough to give us maximum speed and a high degree of performance at a maximum altitude, we must realize that the weight reductions and the power increases with the resultant vibration necessary to obtain this performance at altitude, are not necessarily compatible with maximum performance on the deck. I do not say that we won't bring that piece of equipment down and fly it on the deck. But I do say that if we do, and if we design a piece of equipment for performance in certain areas and then change the ground rules, it doesn't do us a darn bit of good to drop on the deck when you pull off the tail. This is called calculated risk in the military life, and somehow, we must face it.

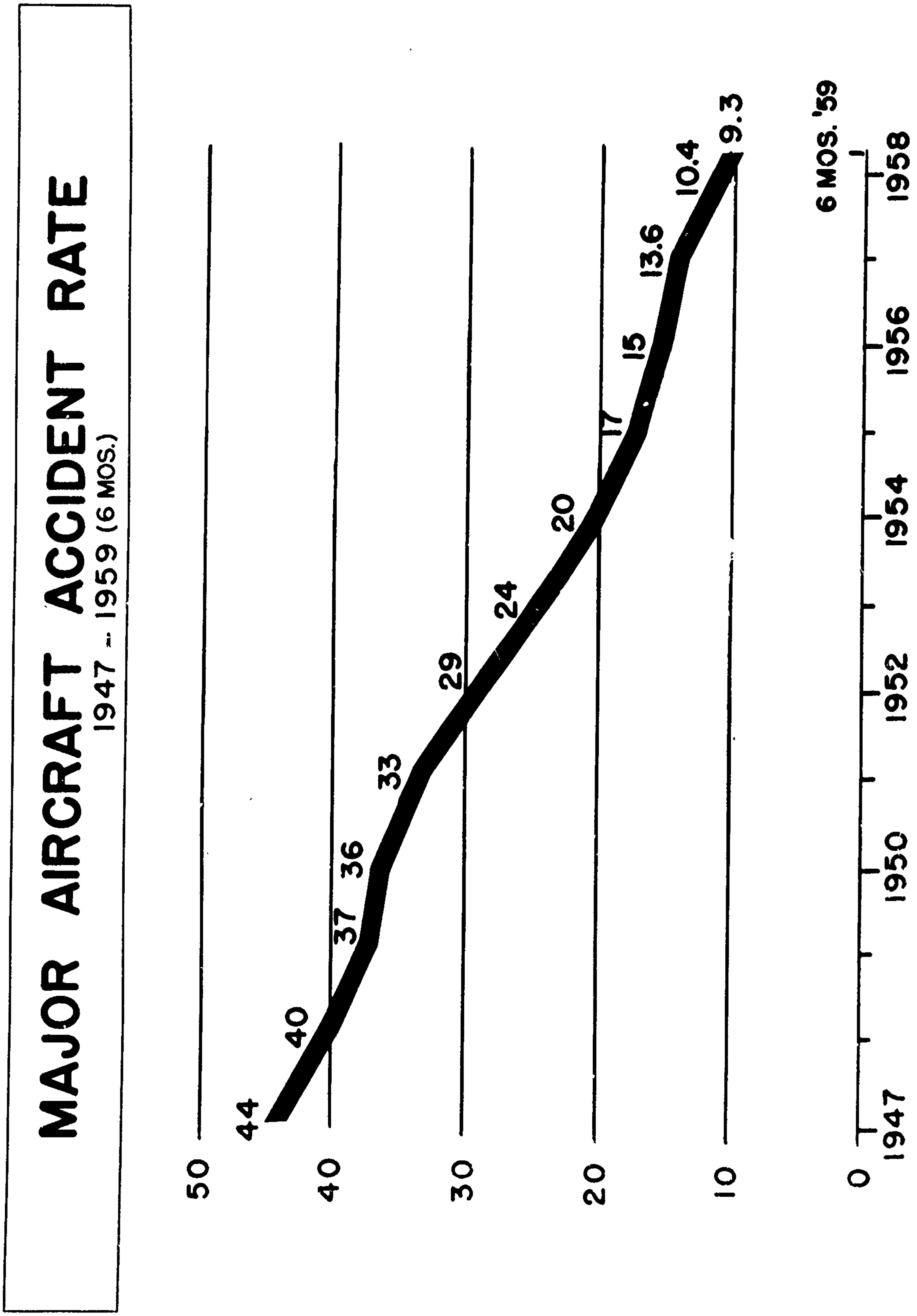
We are interested in the Air Force only in the combat potential of the equipment whether it be aircraft or missile. I now find that I am an expert in the missile business according to a General Order published on the first day of July, which told me so! For four and a half years, I've been asking questions of aeronautical engineers strictly from the aircraft maintenance viewpoint. I am now going to do it in the missile business. It is surprising the number of answers I have been given which don't seem to gee with each other; but, it is interesting.

This job of mine is the most exciting one I have ever had. But no, it is not. The most exciting job I ever had was playing the piano, and you will just have to excuse this story ma'am, for it is true. The most exciting job I ever had was playing a piano in a house of ill-repute. I had to work my way through school at the University of Maryland, and this was the job that I had in the evenings. But I was very young then; I never knew what went on upstairs. Now I'm older, I know what goes on upstairs, and it wouldn't do me a bit of good to go back to the house!

This story wasn't told just to lighten the mood. The point that I am trying to make here is that you may find out what goes on upstairs, but, as in my case, it may not do you any good. If we are talking about a missile or some high altitude aircraft that has come unglued, it will not do us any good unless we can do something about it. It doesn't do us any good to know what goes on upstairs, if we can't do something about it, gentlemen, and this is your job today. It is your number one problem.

The third educational area is the education of the pilot and the crew. We continue to have these bright young fellows who every now and then go out and pull something apart just to prove that no matter how tough you make it, somebody can be tougher and fly it to pieces. These accidents are rapidly dwindling, and are a minority. Commanders are at times forced by national or international complications to commit the equipment, which you have designed and built, into an area for which it was not intended. We cannot help this.

Basically, the problem of licking accidents due to metal fatigue must be solved by you people and your co-workers, whatever office you are in. Let me assure you now that you will have the maximum cooperation from the Directorate For Safety's Office, whether it be at Norton or in Washington, and certainly from me.



MAJOR AIRCRAFT ACCIDENT RATE
1947 - 1959 (6 MOS.)
50
40
30
20
10
0
44
40
37
36
33
29
24
20
17
15
13.6
10.4
9.3
1947
1950
1952
1954
1956
1958
6 MOS. '59

Figure 1

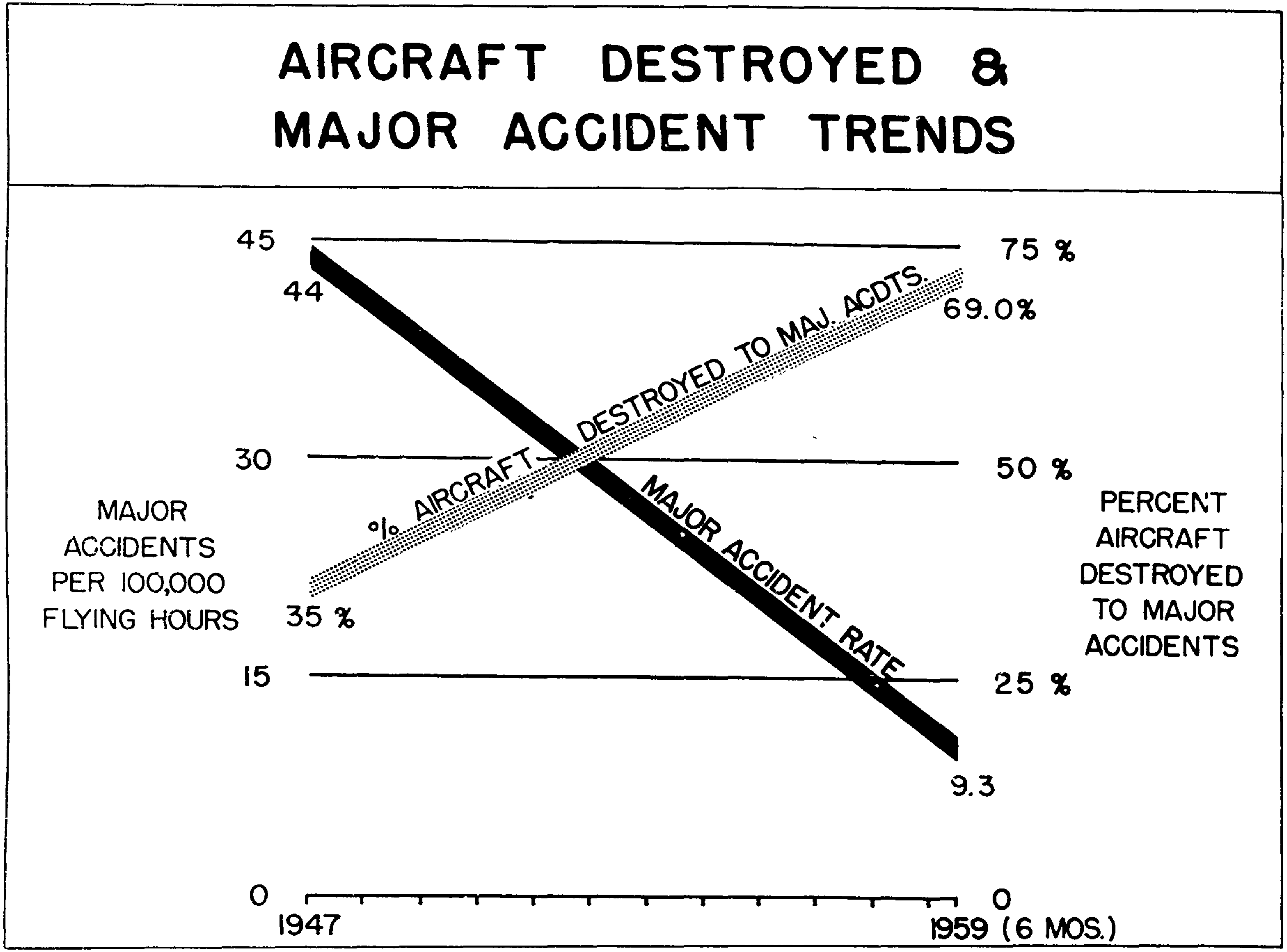

AIRCRAFT DESTROYED &
MAJOR ACCIDENT TRENDS
45
30
15
0
MAJOR
ACCIDENTS
PER 100,000
FLYING HOURS
44
35 %
% AIRCRAFT DESTROYED TO MAJ. ACDTS.
69.0%
MAJOR ACCIDENT RATE
9.3
75 %
50 %
25 %
0
PERCENT
AIRCRAFT
DESTROYED
TO MAJOR
ACCIDENTS
1947
1959 (6 MOS.)

Figure 2

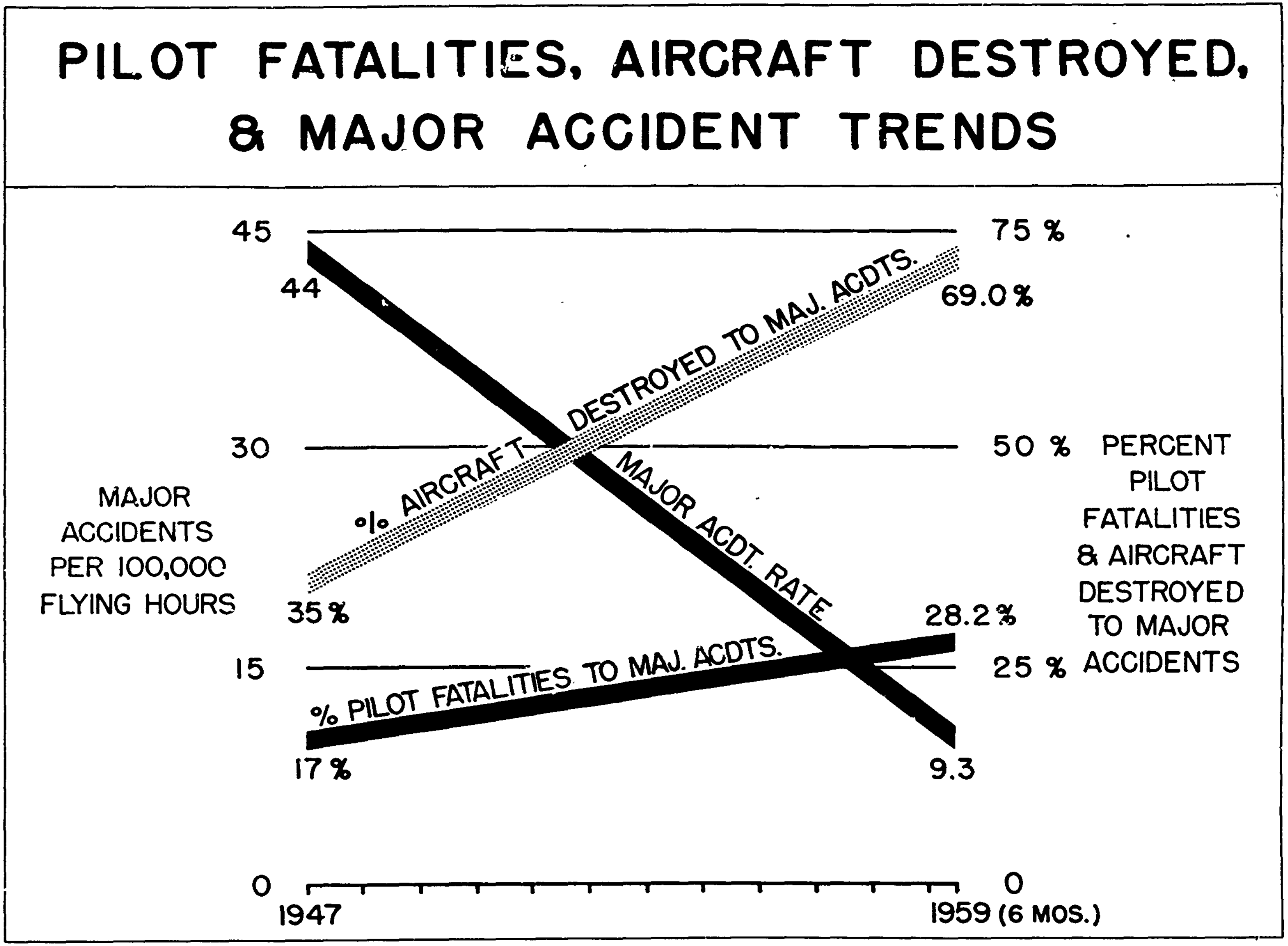

PILOT FATALITIES, AIRCRAFT DESTROYED, & MAJOR ACCIDENT TRENDS
45
30
15
0
MAJOR ACCIDENTS PER 100,000 FLYING HOURS
75 %
50 %
25 %
0
PERCENT PILOT FATALITIES & AIRCRAFT DESTROYED TO MAJOR ACCIDENTS
44
69.0 %
% AIRCRAFT DESTROYED TO MAJ. ACDTS.
MAJOR ACDT. RATE
35 %
28.2 %
% PILOT FATALITIES TO MAJ. ACDTS.
17 %
9.3
1947
1959 (6 MOS.)

Figure 3

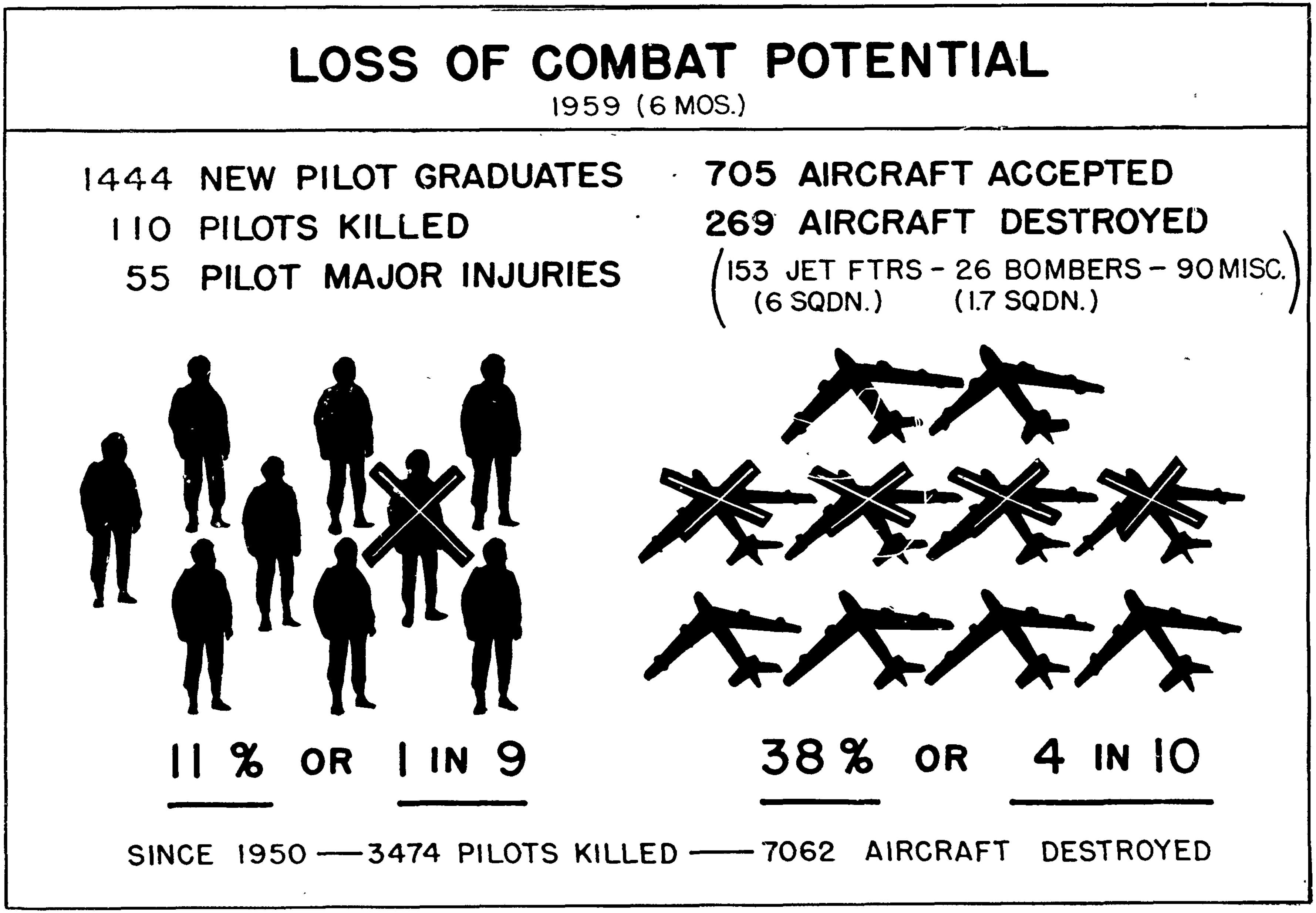

LOSS OF COMBAT POTENTIAL
1959 (6 MOS.)
1444 NEW PILOT GRADUATES
110 PILOTS KILLED
55 PILOT MAJOR INJURIES
705 AIRCRAFT ACCEPTED
269 AIRCRAFT DESTROYED
(153 JET FTRS - 26 BOMBERS - 90 MISC.
(6 SQDN.) (1.7 SQDN.))
11 % OR 1 IN 9
38 % OR 4 IN 10
SINCE 1950 — 3474 PILOTS KILLED — 7062 AIRCRAFT DESTROYED

Figure 4

CAUSES OF MAJOR ACCIDENTS

ALL CAUSES

	1954	1955	1956	1957	1958	(6 MOS.) 1959
PILOT	57 %	57 %	54 %	53 %	53 %	56 %
MATERIEL	28 %	32 %	34 %	40 %	42 %	47 %
MAINTENANCE	8 %	7 %	8 %	9 %	9 %	9 %
SUPPORT	22 %	24 %	26 %	24 %	23 %	20 %
SUPERVISION	17 %	15 %	22 %	23 %	27 %	24 %

Figure 5

CAUSES OF MATERIEL FAILURE ACCIDENTS

ALL CAUSES

	6 MOS. 1958	6 MOS. 1959
POWER PLANT	75 %	76 %
LANDING GEAR	21 %	22 %
FLIGHT CONTROLS	18 %	11 %
HYDRAULIC SYSTEM	6 %	6 %
INSTRUMENTS	8 %	5 %
ELECTRIC SYSTEM	7 %	5 %
AIRFRAME	4 %	5 %
BRAKES	4 %	4 %
FUEL SYSTEM	5 %	3 %
ORDNANCE	4 %	2 %

Figure 6

EARLY JET FIGHTER MISHAPS RESULTING FROM METAL FATIGUE

1 Jan 53 - 31 Mar 59 (Major Acdt. - Minor Acdt. - Incident)

	WING	FUSELAGE	TAIL	LAND. GEAR	Power Plant	SYSTEMS	TOTAL
F-80				0 - 0 - 1	2 - 0 - 0		2 - 0 - 1
F-84	1 - 0 - 0	1 - 0 - 0		1 - 1 - 1	25 - 1 - 4	0 - 0 - 1	28 - 2 - 6
F-86	2 - 1 - 9	1 - 0 - 1	0 - 0 - 1	1 - 2 - 9	24 - 7 - 4	1 - 0 - 0	29 - 10 - 24
F-89				1 - 0 - 1	3 - 0 - 3	0 - 0 - 2	4 - 0 - 6
F-94			0 - 0 - 2	2 - 1 - 0	2 - 0 - 0		4 - 1 - 2
TOTALS	3 - 1 - 9	2 - 0 - 1	0 - 0 - 3	5 - 4 - 12	56 - 8 - 11	1 - 0 - 3	67 - 13 - 39

Figure 7

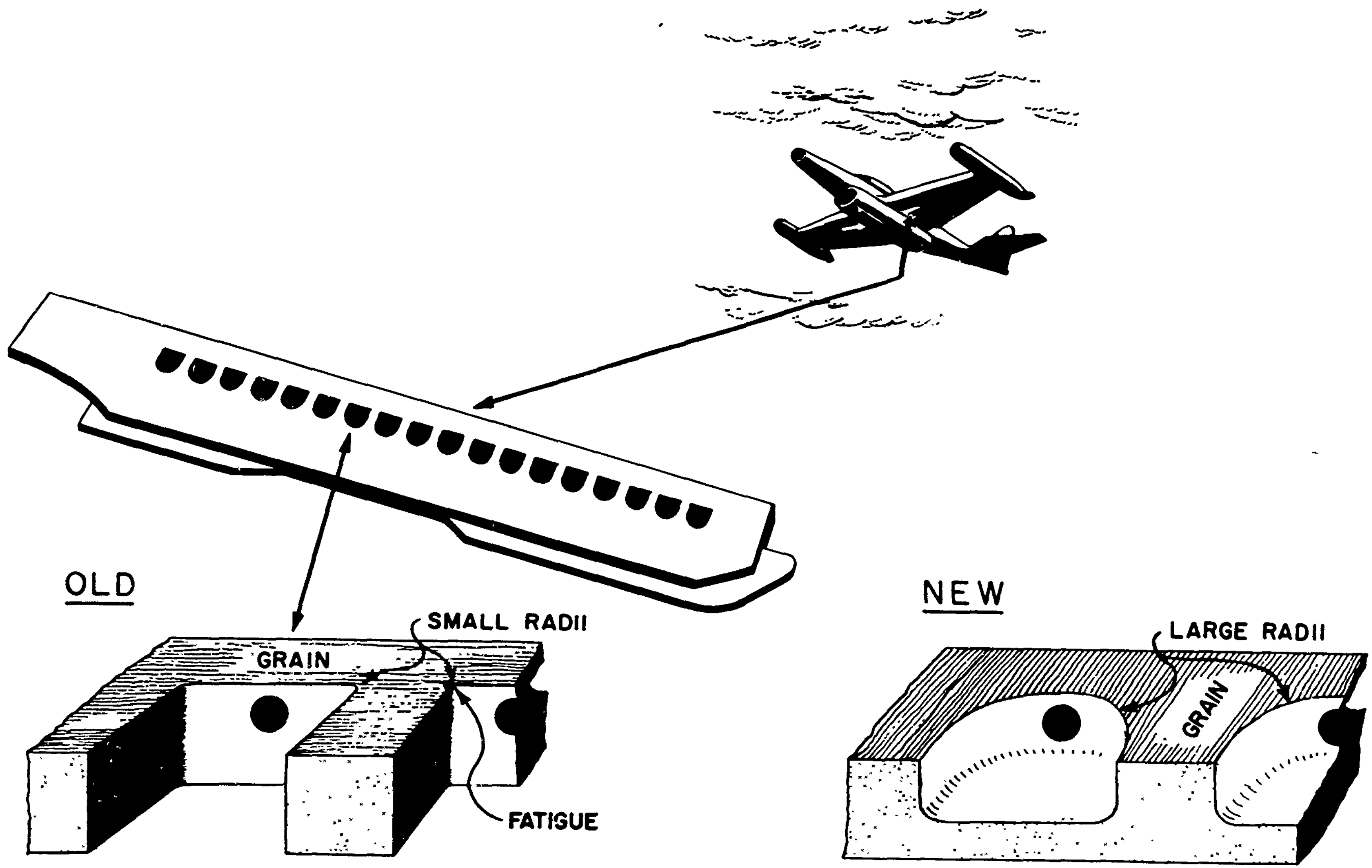


Figure 8

CENTURY SERIES FIGHTER MISHAPS RESULTING FROM METAL FATIGUE

1 Jan 53 - 31 Mar 59 (Major Acdt. - Minor Acdt. - Incident)

	WING	FUSELAGE	TAIL	LAND. GEAR	Power Plant	SYSTEMS	TOTAL
F-100	0 - 0 - 1	0 - 0 - 1		0 - 0 - 1	1 - 0 - 4	3 - 0 - 1	4 - 0 - 8
F-101				0 - 0 - 1	0 - 1 - 0	0 - 0 - 1	0 - 1 - 2
F-102				1 - 0 - 1			1 - 0 - 1
F-104					1 - 0 - 3		1 - 0 - 3
F-105				0 - 0 - 1			0 - 0 - 1
F-106				0 - 0 - 1			0 - 0 - 1
TOTALS	0 - 0 - 1	0 - 0 - 1		1 - 0 - 5	2 - 1 - 7	3 - 0 - 2	6 - 1 - 16

Figure 9

NON-JET BOMBER MISHAPS RESULTING FROM METAL FATIGUE

1 Jan 53 - 31 Mar 59 (Major Acdt. - Minor Acdt. - Incident)

	WING	FUSELAGE	TAIL	LAND. GEAR	Power Plant	SYSTEMS	TOTAL
B-17				1 - 1 - 0	1 - 0 - 1		2 - 1 - 1
B-25				2 - 0 - 1			2 - 0 - 1
B-26				3 - 0 - 0	2 - 0 - 1		5 - 0 - 1
B-29					0 - 0 - 1		0 - 0 - 1
B-36	1 - 0 - 2		1 - 0 - 1	1 - 0 - 1	0 - 0 - 3		3 - 0 - 7
B-50		0 - 1 - 0		0 - 0 - 1			0 - 1 - 1
TOTALS	1 - 0 - 2	0 - 1 - 0	1 - 0 - 1	7 - 1 - 3	3 - 0 - 6		12 - 2 - 12

Figure 10

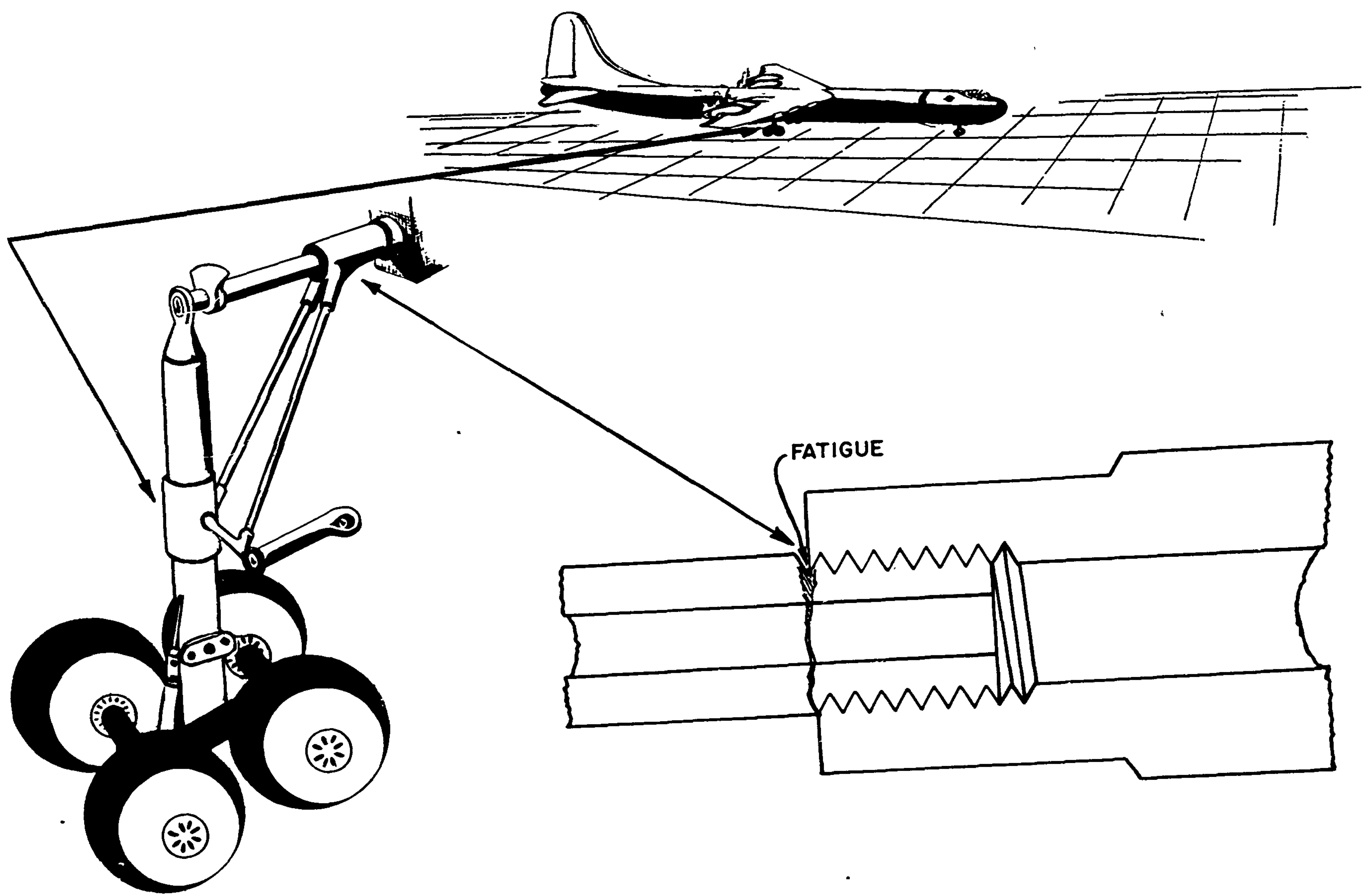


Figure 11

JET BOMBER MISHAPS RESULTING FROM METAL FATIGUE

1 Jan 53 - 31 Mar 59 (Major Acdt. - Minor Acdt. - Incident)

	WING	FUSELAGE	TAIL	LAND. GEAR	Power Plant	SYSTEMS	TOTAL
B-45					1 - 0 - 0		1 - 0 - 0
B-47	1 - 0 - 1	1 - 0 - 0		0 - 1 - 16	1 - 0 - 8	0 - 0 - 13	3 - 1 - 38
B-52	2 - 0 - 1						2 - 0 - 1
B-57		1 - 0 - 0			2 - 2 - 1		3 - 2 - 1
B-66				1 - 0 - 1			1 - 0 - 1
TOTALS	3 - 0 - 2	2 - 0 - 0		1 - 1 - 17	4 - 2 - 9	0 - 0 - 13	10 - 3 - 41

Figure 12

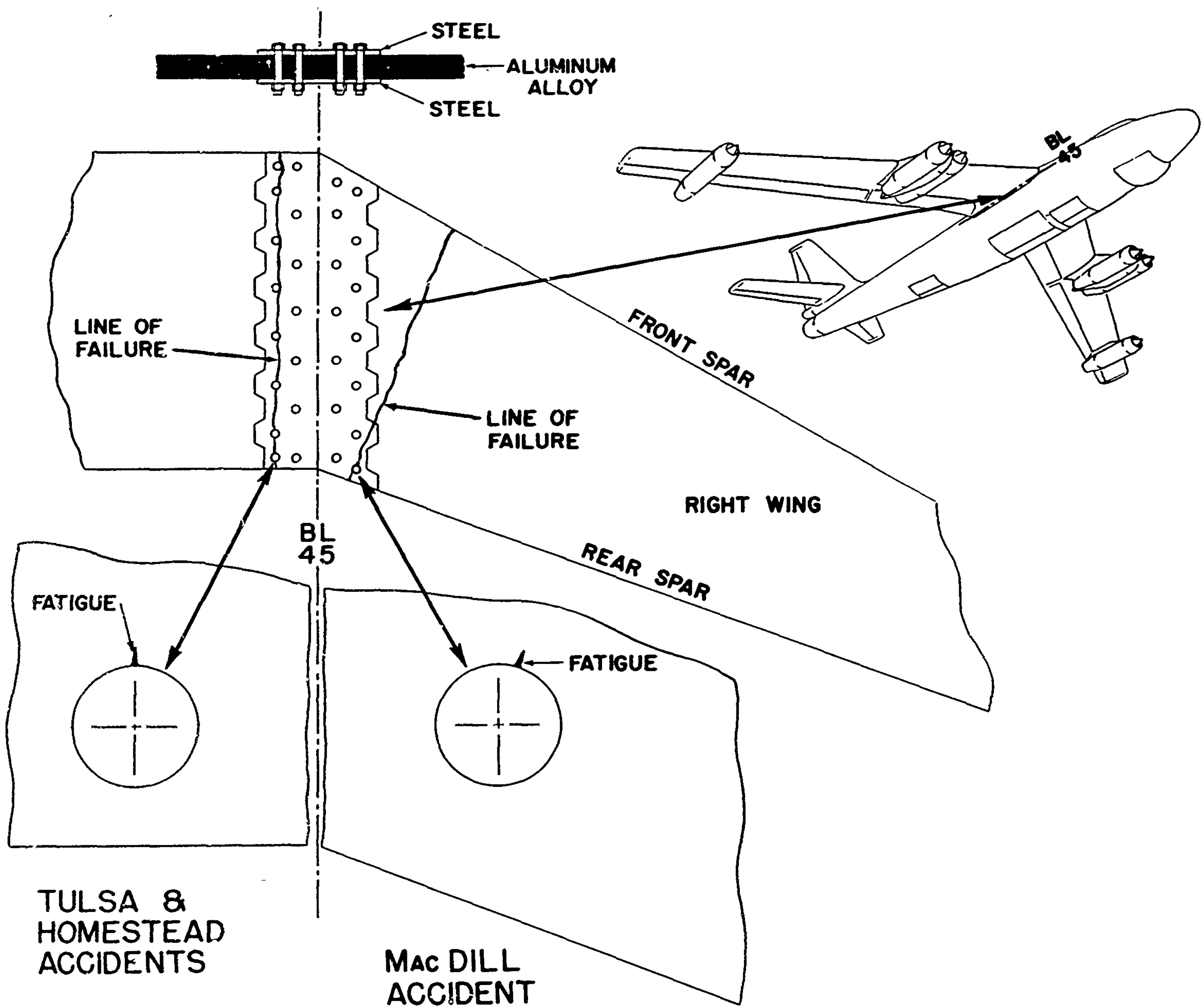

STEEL
ALUMINUM ALLOY
STEEL
BL 45
LINE OF FAILURE
LINE OF FAILURE
FRONT SPAR
RIGHT WING
REAR SPAR
BL 45
FATIGUE
FATIGUE
TULSA & HOMESTEAD ACCIDENTS
MAC DILL ACCIDENT

Figure 13

CARGO MISHAPS RESULTING FROM METAL FATIGUE

1 Jan 53 - 31 Mar 59 (Major Acdt. - Minor Acdt. - Incident)

	WING	FUSELAGE	TAIL	LAND. GEAR	Power Plant	SYSTEMS	TOTAL
C-45				2 - 0 - 0	5 - 1 - 2		7 - 1 - 2
C-47				1 - 0 - 0	1 - 0 - 2		2 - 0 - 2
C-97					4 - 0 - 1		4 - 0 - 1
C-119				2 - 0 - 0	0 - 0 - 2	0 - 0 - 1	2 - 0 - 3
C-121					0 - 0 - 1		0 - 0 - 1
C-123				0 - 0 - 1	1 - 0 - 0		1 - 0 - 1
C-124				1 - 0 - 0	1 - 1 - 1		2 - 1 - 1
TOTALS				6 - 0 - 1	12 - 2 - 9	0 - 0 - 1	18 - 2 - 11

Figure 14

TRAINER MISHAPS RESULTING FROM METAL FATIGUE

1 Jan 53 - 31 Mar 59 (Major Acdt. - Minor Acdt. - Incident)

	WING	FUSELAGE	TAIL	LAND. GEAR	Power Plant	SYSTEMS	TOTAL
T-6				1 - 0 - 0	0 - 1 - 1		1 - 1 - 1
T-7				1 - 0 - 0	1 - 0 - 0		2 - 0 - 0
T-28				2 - 0 - 1	9 - 2 - 3		11 - 2 - 4
T-29					0 - 1 - 0		0 - 1 - 0
T-33		1 - 0 - 0		2 - 0 - 3	20 - 2 - 3	1 - 0 - 5	24 - 2 - 11
TOTALS		1 - 0 - 0		6 - 0 - 4	30 - 6 - 7	1 - 0 - 5	38 - 6 - 16

Figure 15

MISCELLANEOUS MISHAPS RESULTING FROM METAL FATIGUE

1 Jan 53 - 31 Mar 59 (Major Acdt. - Minor Acdt. - Incident)

	WING	FUSELAGE	TAIL	LAND. GEAR	Power Plant	SYSTEMS	TOTAL
F-51	1 - 0 - 0		1 - 0 - 0		1 - 0 - 2		3 - 0 - 2
H-5					2 - 0 - 0		2 - 0 - 0
H-19					2 - 0 - 0		2 - 0 - 0
H-21			0 - 0 - 1		1 - 0 - 0		1 - 0 - 1
L-20				1 - 0 - 0			1 - 0 - 0
L-27					0 - 0 - 1		0 - 0 - 1
PA-18		0 - 0 - 1					0 - 0 - 1
TOTALS	1 - 0 - 0	0 - 0 - 1	1 - 0 - 1	1 - 0 - 0	6 - 0 - 3		9 - 0 - 5

Figure 16

SUMMARY - MISHAPS RESULTING FROM METAL FATIGUE

1 Jan 53 - 31 Mar 59 (Major Acdt. - Minor Acdt. - Incident)

	WING	FUSELAGE	TAIL	LAND. GEAR	Power Plant	SYSTEMS	TOTAL
Early-Jet Fighters	3 - 1 - 9	2 - 0 - 1	0 - 0 - 3	5 - 4 - 12	56 - 8 - 11	1 - 0 - 3	67 - 13 - 39
Century Fighters	0 - 0 - 1	0 - 0 - 1		1 - 0 - 5	2 - 1 - 7	3 - 0-2	6 - 1 - 16
Non-Jet Bombers	1 - 0 - 2	0 - 1 - 0	1 - 0 - 1	7 - 1 - 3	3 - 0 - 6		12 - 2 - 12
Jet Bombers	3 - 0 - 2	2 - 0 - 0		1 - 1 - 17	4 - 2 - 9	0 - 0 - 13	10 - 3 - 41
Cargo				6 - 0 - 1	12 - 2 - 9	0 - 0 - 1	18 - 2 - 11
Trainer		1 - 0 - 0		6 - 0 - 4	30 - 6 - 7	1 - 0 - 5	38 - 6 - 16
Misc.	1 - 0 - 0	0 - 0 - 1	1 - 0 - 1	1 - 0 - 0	6 - 0 - 3		9 - 0 - 5
TOTALS	8 - 1 - 14	5 - 1 - 3	2 - 0 - 5	27 - 6 - 42	113 - 19 - 52	5 - 0 - 24	160 - 27 - 140

Figure 17

AIRCRAFT STRUCTURAL FATIGUE PROBLEMS IN SAC

By

H. E. WATKINS
Colonel, USAF

Headquarters Strategic Air Command
Offutt Air Force Base, Nebraska

Just to set the stage for this discussion, let me review the SAC mission -- Be prepared to conduct Strategic Air Operations on a global basis so that in the event of sudden aggression, SAC could immediately mount simultaneous attacks.

This, in itself, identifies the command as an operational combat command. It is mobilized and in its battle position with its weapons and people. If you were to grant us 72 hours' warning, we still would have no more aircraft with which to fight than we now have. This is obvious -- so Strategic Air Command will use only that equipment it has at hand when the whistle blows. If we contract the warning time to 48 hours or to 15 minutes, those weapons systems in heavy repair drop out like gold -- the heaviest maintenance first; the lightest maintenance last -- but the droppings are still as in gold. This is why we are pushing the day-to-day availability higher and higher as our reaction time grows shorter. Structural modifications to fix skin, longerons and other cracks are generally heavy maintenance and eliminate the aircraft from the SAC inventory.

Basically, a flying machine is designed and built to satisfy a specific requirement. The requirement, of course, is a composite of existing tactics and a projection of future needs tempered by "state of the art" (if you will) design capabilities. Our projection of future needs cannot be 100% accurate without complete stagnation of planning and tactical studies. Obviously, a weapon system must incorporate a high degree of versatility to provide for growth and economical use over a period of years. In this regard, the state of the art begins to throw up a barrier to the degree of versatility that can be achieved. As I see it, this barrier can be separated into two categories, only one of which is real. That one is simply an absolute lack of know-how. The other seems to be real but is not. It consists of structural weaknesses that could have been designed

out of the weapon if full advantage had been taken of the many scattered clues awaiting correlation and application.

This latter comment is not a finger-pointing exercise because we are all tarred with the same brush. In-service aircraft have failed and been modified without the experience so gained getting beyond a tight little group of directly responsible individuals. Even our security system works to our disadvantage when new materials, designs and techniques are not made widely available at an early date because they have been developed within a classified project. It would seem that we could use some sort of a machine that would take UR's, maintenance staff studies, engineering development reports, accident reports and flight test data, integrate the lot and print out revisions to the HIAD. It is reasonable to assume that comprehensive gathering and integration into design of all available experience would produce the same result as a major breakthrough in technology. Suffice to say that everything that we in the using command attempt to get out of our aircraft in the way of additional tactics imposes an additional fatigue factor that is inherent with use. This is also true of some of the things that we put into the aircraft such as uprated engines with their high sonic noise levels.

In closing, I will briefly review SAC's participation in the Air Force Structural Integrity program as applied to the VGH program.

After installation in the aircraft is complete, SAC will be responsible for making periodic operational checks of the system with a go/no-go type of tester. In the event that a component malfunctions, local maintenance will consist only of "remove and replace" with a like item. The reparable item will then be returned to a central point for repair.

It is proposed that the maintenance man will be held responsible to check the recorder magazine time. As magazines are used and replaced they will be routed to a central point on the base where correlative flight crew data will be packed with them for shipment to the data reduction agency.

It is at this point in the cycle that the over-all program becomes obscure to the SAC observer and we are quite concerned. It is a safe assumption that the data will be compiled and reduced to some usable form that will be quite adequate for broad scale structural load analysis -- and I hasten to add that this is a valid endeavor. Defining the exact environment encountered by aircraft in flight has been accomplished on only a very limited basis to date and the VGH program should improve this situation. Using information thus gathered, aircraft designers will be able to provide us with much safer and more reliable machines at some future date; however, we feel that an additional capability should be realized. Namely, specific load analyses of the individual aircraft should be fed back to the SAC maintenance people for direct comparison to the actual condition of critical structures during maintenance and inspection. It is obvious that furnishing them with this additional parameter of measurement

would greatly enhance the validity of their endeavors. I am not prepared to specify exactly what type of feedback information should be provided to SAC because we are not cognizant at this time of how comprehensive a presentation can be made from the data to be recorded with the data reduction facilities available. However, to be meaningful, this information should point to areas of the aircraft likely to be affected and the magnitude and direction of stresses applied. The presentation might be provided as a life factor index that would contain an integration of stress magnitude and frequency with the design limits.

STRUCTURAL INTEGRITY PROGRAM FOR HIGH PERFORMANCE AIRCRAFT

By

Colonel John P. Taylor

Air Research and Development Command

Aircraft Laboratory, Wright Air Development Center

INTRODUCTION

The serious problems of structural integrity that have recently become apparent in our first-line military aircraft emphasize the fact that frontier and state of the art type knowledge in this area must advance more rapidly than at present. This is particularly true if this nation is to proceed with its future program for higher performance aero-space vehicles, without incurring grave delays and excessive costs, due to the lack of enough basic structural integrity information.

Some eighteen months ago, a considerable but diverse effort was underway at the working level of the military services to resolve the varied problems of structural fatigue. The program was sound as far as it went, but it lacked the essential and basic elements of: a point of focus, a coordinated effort, adequate recognition by higher authorities and a suitable priority. These basic deficiencies had to be met - and were - principally by the creation and approval of a joint Air Materiel Command - Air Research and Development Command plan and program. It is the ARDC part of the program which I wish now to highlight briefly for you.

First, we needed a point of focus. The B-47 was it.

At about the same time and due to the series of B-47 structural fatigue accidents early in 1958, General Power, Commander-in-Chief, Strategic Air Command, sent a letter in late April to the U. S. Air Force Chief of Staff, General White, stating: "I believe it absolutely essential that we learn what we bought in terms of service life in the B-47 and B-52, and what we will buy in the B-58 and B-70". This question and its resolution then became one of the early bases for our program.

However, even before this letter had been signed, the Air Research and Development Command had approached the contractors manufacturing first-line aircraft for the Strategic Air Command and had asked them for their best estimate of service life, based on the parameters shown in Figure 1. Several types of missions are plotted therein against the original design, current use and desired ultimate gross weight of an aircraft. The missions selected were those which were believed to result in an increasing incidence of fatigue and consequent potentially shortened airframe service life. It is interesting to note that contractor estimates of service life, submitted in reply to our question, in some cases, varied as much as 100% with those which had been submitted voluntarily by them within the previous six months' period. This study in itself made it clear that the state of the art did not then permit accurate estimates of expected service life of our first-line Strategic Air Command type aircraft. In fact the shortened life estimates submitted for desired ultimate gross weight and low level-alert type operational work were generally shocking.

With this data firmly in mind, we decided to see how to improve aircraft service life by setting up a series of "Steps to Improve Structural Integrity of Hi-Performance Aircraft".

STATIC TEST

The first step was a re-evaluation of the traditional "Static Test," initiated in 1918. As you know, one of the first aircraft to be assembled in each new weapon system series is subjected to a complete laboratory static test. Until a decade ago, sand and lead shot bags were in common use to simulate flight loads, as indicated in Figure 2, a photograph of an F-84 undergoing static test. How complicated the art has become is shown by Figure 3, a photograph of a B-58 recently taken while the aircraft was undergoing the vertical stabilizer portion of the static test. You will note the myriad of high intensity heat lamps which are used to simulate the effects of aerodynamic heating at the high speeds to which the aircraft can go. Almost without exception, since static test was first initiated, every new type of vehicle submitted to this test has failed, in some way, before ultimate load has been reached in all parts of the structure. Static test work is performed, in effect, to see whether or not the complicated structure of the aircraft will actually sustain loads up to and including the ultimate loads it is expected to encounter in the normal operational maneuvers for which it has been designed and constructed by the airframe manufacturer (Figure 4).

A complete static test results in the destruction and scrapping of the vehicle undergoing test. The tests have more than paid their way down through the years. They have, for example, pointed out critical areas of structure which needed to be re-designed and replaced prior to the actual operational utilization of the aircraft up to its load limits. In addition, they have resulted, as shown in Figure 5, in a material improvement of design criteria, which, as you know, is one of the fundamental tools with which the flight vehicle designer creates future weapons system. Our re-evaluation of the static test process not only convinced us that it was and would continue to be a necessary part of the structural integrity program but also that considerable research and development work was necessary to insure that the static test process would meet the more demanding requirements of future weapons systems. Such work is under way.

FLIGHT LOADS SURVEY

The next step in our joint AMC-ARDC program involves that of a "Flight Loads Survey". Since there has been some question as to what the term "Flight Loads Survey" involves, the following is a comprehensive definition of the term: A "Flight Load Survey" is a flight test maneuver program performed by the airplane contractor on a well instrumented and representative airplane model to substantiate the structural integrity of the airplane for limit load service operation. The dynamic response portion of the flight loads survey is a recent extension which includes gust loads and landing-taxi tests to determine elastic response characteristics of the structure to these dynamic load inputs.

The curve in Figure 6 represents a generalized flight envelope of a typical high performance aircraft of today. Since 1952 one of the early aircraft in each new weapons system series has been thoroughly instrumented and flown at critical flight load points, typically shown by the small circles, in selected manuevers in order to check, experimentally, the theoretical calculations of loads which the aircraft designer has expected his aircraft would encounter in flight maneuvers for

which the aircraft was designed. The process of instrumentation of such a test may cost several million dollars per vehicle, and flight test programs, on occasion, may last up to two years or more. A flight load survey will continue to be made on each new type of future weapon system, concurrently with the static test.

The flight test aircraft can be placed in the active operational inventory as soon as the flight load survey has been completed and the special instrumentation has been removed. As indicated in Figure 7, flight load test data feeds back into and checks out the theoretical static test loads data, submitted by the designer, thereby not only improving the static test, but also resulting in the improvement of design criteria.

FATIGUE TEST

The next step in our joint AMC-ARDC program involves that of "Fatigue Test".

Fatigue is not a recent problem with the Air Force. These few incidents will emphasize its recurrence. The disastrous F-89 accident at the 1952 Detroit Air Show resulted from the fatigue failure of wing-attach fittings. Those of you who worked with the F-84 will remember the serious problems the Tactical Air Command had with this aircraft type during the 1953, -54, and -55 time period. In 1958, the F-101-A, undergoing laboratory cyclic fatigue tests at McDonnell Aircraft Corporation in St. Louis, suffered a catastrophic wing failure, which, if it had occurred after the aircraft had been placed in operational use, would have resulted not only in the loss of aircraft and crews, but also in an expensive retrofit program.

The Air Force is not alone in its encounters with the problem of fatigue. Many of you will remember the Martin 202 problem in 1948 and the Comet incident in 1954.

Since the B-47 fleet required extensive special attention in 1958, let us digress from our program discussion, for a moment, to point out several interesting events that occurred in the accelerated cyclic fatigue tests of this one particular type of aircraft. Figure 8 shows the B-47 wing structure and points out several of the critical areas where fatigue resulted in fatal accidents in operational aircraft. These included: Butt-Line 45, Body Station 515, and Wing Station 354. Through the cooperation of the Strategic Air Command, the Air Training Command and the Air Materiel Command, three low-time B-47 aircraft were made available for comprehensive fatigue-cyclic test purposes. Working jointly with the Strategic Air Command and the contractor, ARDC and AMC developed a typical composite SAC mission and a suitable flight spectrum to simulate the bending moments at two of these most crtical points--WS 354 and BL 45.

Half-way through one of the tests at the Boeing-Wichita plant, one of three B-47's undergoing test broke in two. This event was totally unexpected at the initiation of the test and would have resulted in an extremely serious problem if it had occurred in the operational fleet. In order to determine if this were a laboratory phenomenon only, an immediate examination was made of high time fleet aircraft. A similar condition was found to be incipient in these high time aircraft.

This one event alone more than paid for the cost of the three test aircraft and the entire series of laboratory tests, since it permitted a relatively simple retro-fit to be made long before a majority of the operational B-47 fleet had acquired an equivalent amount of actual flight time.

Another thing that the tests showed us was that the three aircraft undergoing exactly comparable tests at the Boeing-Wichita plant, the Douglas-Tulsa plant, and the NASA-Langley Laboratory, did not necessarily develop cracks at similar locations nor at similar test times. Cracks shown in Figure 9 are typical of those which could have ended catastrophically in operational service if they had gone undetected for a long enough period of time. On the other hand, an amazing degree of similarity occurred in the time of final failure of these three vehicles, at least, in terms of what the structures expert might have expected (Figure 9).

One of the things which was accomplished, as a result of the 1958 inspection of every B-47 in the entire fleet, was a detailed analysis of the incipient fatigue cracks to be found in the joints at WS 354 and BL 45. The results are revealing and clearly show that fatigue cracks occurred in a majority of bolt holes at these two locations, as shown in Figure 10. This statistical analysis also clearly pointed out certain critical areas in the structure which henceforth will be inspected with great care. To my knowledge, this is the first time an intensive survey of this type has been made of such a large statistical sampling of operational aircraft. It should prove of value to the structures and criteria expert.

In an effort to comply with General Power's desire to forecast B-47 fleet service life as soon as possible, a series of steps were set in motion early in 1958, of which the cyclic load test just described was only one. In May of this year, personnel from all interested Air Force Commands and the contractor sat together to come up with an improved service life estimate. This current estimate is felt to have a 70% - 80% degree of reliability. By January, 1960, it is hoped that this estimate can be improved to approximately an 80% - 90% basis on careful analysis of all the work accomplished to that time. It should be noted that a completely accurate 100% service life estimate cannot be made until the last B-47 has been phased out of active inventory.

Let us now return to the basic problem of fatigue, as a part of our total AMC-ARDC program. Fatigue or cyclic testing results in the complete destruction of the vehicle involved. This can be a costly item. The state of the art at present, however, does not permit the use of one airframe for both static and fatigue test purposes. The Air Force will complete a fatigue test on each new aircraft weapon system henceforth. The cases of the F-101 and the B-47 convinced us of the real need for such a test and proved, beyond a shadow of a doubt, that such tests pay their way in ultimate savings of life and property (Figure 11).

LOW ALTITUDE GUST ENVIRONMENT

The next step in our program was that of determining what a low altitude gust profile actually looks like. Even though aircraft have been flying for over fifty years, there is only the most limited literature available which clearly outlines for the designer, the exact characteristics of typical gusts through which his vehicle must fly at low altitude.

Low altitude gust environment, as you all know, is caused by differences in local air flow over different types of terrain. The object of the program which was already in effect was to fly at altitudes below one thousand feet, and preferably at around 500 ft.

The B-66 used for this work had been thoroughly instrumented for one of the Pacific tests in order to record air loads resulting from blast waves. For this new work, incorporated in it was a "boom", sticking out of the nose, which could sense and measure a gust prior to the time its effect could be felt by the rest of the aircraft. This "boom" is instrumented to determine gust load input, strength, differential rate of change of gust loading, and frequency and direction of occurrence, and, to correlate this information with bending and shear moments, and accelerations; pitch, yaw, and roll rates are also measured in other parts of the aircraft, as shown in Figure 12. For the first time, also, a definite effort was made to correlate this data with terrain photographs and pilot reaction.

At the present time, we are getting some two and one-half million data points for each five minute record run.

The B-66 is flown out of Edwards Air Force Base, Kirtland Air Force Base, Shaw Air Force Base, Wright-Patterson Air Force Base, plus a yet-to-be-selected northern base, in order to fly over areas which duplicate terrains typical in all parts of the world. Since the program will extend over a year's period, the aircraft will also fly in all types of climatological and meteorological conditions.

Since the degree of instrumentation required is extremely expensive, only one aircraft is being used for this project. However, two B-47's, one flying out of the Wright Air Development Center and the other flying out of Boeing-Wichita, are also developing much useful information on the effect of low altitude environment on B-47's which will feed back into improved design and fatigue spectra for use in other weapon systems still to be cyclic load tested.

The low altitude gust program (Figure 13), whose results will be made widely available, will materially improve spectra used in future fatigue tests and will certainly be of great value particularly to the designer of low altitude weapon systems.

MISSION PROFILE DATA

Through the years, the design criteria and fatigue expert has felt an increasing need for up-to-date "Mission Profile Data" from the Operational Commands.

Due to the recent intensive cooperation of the senior commanders and staffs of major Air Force field commands, the designer now has available to him an important tool. "Decks" of cards in a typical format, as shown in Figure 14, have been forwarded from each Field Command in the Zone of the Interior, to the Air Research and Development Command, so that our engineers can determine to what loads the weapons systems in-being will probably be exposed during their future operational life.

Mission profile data will most assuredly help in the development of improved fatigue spectra and will further result in the improvement of design criteria (Figure 15).

CONTINUING FLIGHT LOADS PROGRAM

Early in the development of a total AMC-ARDC joint program, General Anderson, then Commander of the Air Research and Development Command, sent a message to General Rawlings, then Commander of the Air Materiel Command, and to General Wray, the Commander of the Wright Air Development Center, stating: "I request a joint AMC-ARDC proposal for an adequate flight loads survey program for all type of aircraft on a continuing basis to pinpoint critical failure areas, develop adequate engineering fixes, determine effects of operational missions and techniques on service life, improve fatigue data, and develop better design criteria". The Wright Air Development Center had been accomplishing such work for many years, but on a low priority and a limited basis. This instrumentation program is divided into three major sub-programs. The first one to consider is the immediate "Interim Service Load Program".

INTERIM SERVICE LOADS

This program, initiated in selected SAC aircraft to measure airspeed, acceleration and altitude against time, as shown in Figure 16, has resulted in the acquisition of over 2000 hours of actual flying time, on typical service missions performed at selected SAC bases. This data is now being reduced and analyzed.

In effect, if we look at a plot of acceleration, both positive and negative, versus aircraft speed in knots, many of you will recognize the typical VG type diagram, shown in Figure 17, which results from the measurement of flight loads encountered during a normal mission. The curve in Figure 20 is, in effect, the straight line portion of the curve in Figure 4. The points of particular interest to the criteria expert and the maintenance engineer are those which fall well outside the limit load envelope.

This Interim Service Load Program will be continued for several months. Its results have already helped to improve criteria for fatigue test work and will, in due course, be reflected in overall design criteria as shown in Figure 18.

LIFE HISTORY PROGRAM

As a result of the directive from General Anderson, the Air Materiel Command and the Air Research and Development Command, working jointly with the Operational Commands, developed a "Life History Program", as the second phase of the special instrumentation program. The heart of this phase is a light weight flight data recording instrument, officially described as the Signal Data Recording Set A/A24U-3, which will be placed on selected firstline aircraft for the remainder of the life of that aircraft. This instrument will be capable of recording 50 hours of flight time before the tape is replaced. This program involves extensive teamwork between the Operational Commands at the squadron level and the interested Air Materiel Areas of AMC. The groundwork is now laid for a large scale program which will be of considerable value in aircraft maintenance work. This program also gives airspeed, acceleration and altitude versus time or one degree of freedom type information as shown in Figure 16. The results of this program feed back, once again, into the fatigue tests and design criteria areas.

EIGHT CHANNEL SERVICE LOADS

The third part of the special instrumentation sub-program has been referred to as the "8 Channel Service Loads" program. Instead of acquiring VGH information alone (Figure 16), additional data is obtained, as shown in Figure 19, which will give us, in terms of six degrees of freedom, the exact load history which will be encountered by an aircraft from the point of take-off roll to the point of touchdown following completion of the flight mission. The amount of data which will result from this program is large. At the present time the Air Force expects to instrument only a 10% sample of those aircraft involved in the life history program.

As Figure 20 clearly indicates, this new 8-channel recorder program is, and has been, a jointly cooperative one with the Navy and National Aeronautics and Space Administration. In fact, the 8-channel recorder program dates back to 1954. At the present time it is expected that large quantity production of this instrument will start in the near future.

Insofar as ARDC is concerned all three phases of this instrumention program, as shown in Figure 21, will be vital to improving both fatigue and static test work and these efforts will certainly result in the improvement of design criteria.

SONIC FATIGUE

Many of you have heard of the problem of "Sonic Fatigue." This is the next phase of our program which I want to describe. It is an increasingly serious one as performance limits of modern aircraft are moved upward and engine powers are increased.

The decibel limits reached in current aircraft from jet engine sources are high. Limits of 164-170 decibels are typical on B-52, B-58 and KC-135 type aircraft.

To refresh your memories from engineering school days let me point out, as shown in Figure 22, that for each increase of 20 db the actual pressure level is increased by a factor of 10 in pounds per square inch. 160 db is close to jet engine take-off power. 180 db is in the area of noise emanating from large boost rocket sources. Note that somewhere between 180 and 200 db the pressure level appears to increase beyond a possible 14.7 pounds per square inch at sea level. What happens to noise, in its traditional sense, at that point is not clear, except that the wave peaks appear to crumble into the wave troughs. To date and to my knowledge such sound levels have not been achieved experimentally for laboratory examination.

The second factor important in acoustic fatigue is the amount of acoustic power available to react on airframe structure. In the last decade, acoustic power output has increased by a factor of 100 - from the days of the F-80 to those of the B-70.

The cost to the Air Force of Engineering Change Procurement Orders to correct sonic fatigue difficulties has risen rapidly in the last several years. For example, the B-52 first encountered this problem in its early days. Structural changes to correct this sonic fatigue problem were time consuming and costly. The problem was considered solved; however on increasing the take-off thrust in the later models of the airplane, new sonic fatigue problems were encountered which had to be corrected. The same type of annual cost increase curve is true for the B-66. Even our new KC-135 aircraft are not immune and have already required expensive maintenance due to sonic fatigue.

Of increasing interest to the designer is the fact that, if one plots rocket engine thrust against sound pressure level in db, as shown in Figure 23, it is already clear that, as rocket engine thrust continues to increase, extremely high noise levels will be reached. Even current rockets yield average sound pressure levels near the nozzle of around 170 db. If it is possible to continue the linear extrapolation of the lines shown in Figure 23, it is interesting to note that sound pressure levels, near the nozzles of large size rockets now under development, should result in the immediate failure of traditional structural configurations. Even fifty feet forward of the nozzle, sound levels will be encountered which could result in all types of structural and equipment failures unless the effects of sonic noise sources are taken into consideration by the designer in the development of future equipment and structure.

Another type of noise, commonly called "Boundary-Layer Noise" is one with which the future designer must cope. The plot of altitude versus velocity (Figure 24), includes the familar missile reentry and continuous flight corridor paths. The plot has superimposed on it the db levels which will be encountered due to boundary layer noise. Note the high levels involved.

In over simplified form, boundary layer noise results from the turbulent flow which occurs after the point of flow discontinuity has been reached on any flight vehicle surface as shown in Figure 25. The pressure variations between adjacent vortices of air, flowing over the flight vehicle surface at approximately 0.8 the velocity of flight speed, have the same general effect on structure as more traditional sources of noise.

In the case of one current high performance aircraft, the principal frequency encountered, from a point ten feet behind the point of airflow discontinuity to one sixteen feet aft, may vary as much as a thousand cycles per second.

If one considers that future structures may be exposed to pressure level variations of up to five pounds per square inch or more and thousands of times per second it is not difficult to see why the Air Force is concerned about sonic fatigue in future weapons systems.

In order to study sonic fatigue type phenomena the Air Research and Development Command plans a large scale sonic test facility which will include a test chamber approximately 80 x 80 x 40 feet in size, with a siren bank noise source capable of producing up to one million watts of acoustic energy at a pressure level up to 175 db.

In order to reduce construction costs to a minimum the 20 foot, sub-sonic Massie Wind Tunnel at Wright Air Development Center is being dismantled and,

wherever possible, existing bricks, mortar and equipment are being used in the new construction program. This facility has recently received House and Senate Appropriations Committee approval.

A great deal of high priority effort must be forthcoming in this field in years to come if the sonic fatigue problem is not to prove a bottleneck in future flight vehicle operation. Figure 26 outlines how the sonic fatigue area reflects back into the total structural integrity program.

HIGH TEMPERATURE

You are all familiar with the problem of high temperature. It is included in our program because of its impact on the fatigue problem.

Figure 27 is a chart of temperature in degrees F plotted against what, in effect, is the "efficiency" of the material shown. Included are several of the different types of materials which were considered for use in the B-70. Note in particular how the traditional 24 ST and 75 ST materials drop in usefulness for aircraft structure after a temperature of 200 degrees F has been reached. AM350 was selected for certain portions of the structure of the B-70. It is interesting to note, in Figure 28, how rapidly even AM350 decreases in usefulness at the temperatures which will be encountered in future flight vehicles.

It is currently felt that a 10 hour life for leading edge structure in a boost-glide type vehicle is about all that the designer can expect. Materials for such use are shown in Figure 29.

In order to remedy this situation a great deal of materials research is under way throughout the country. It is hoped that practical material operating temperature limits available to structural component designers can be increased in the 5 year time period as outlined in Figure 30.

Design criteria which result from high temperature studies, (Figure 31), of both material and structural configurations must be considered by the flight vehicle designer in future.

DESIGN CRITERIA

The extensive and expensive program already described does not achieve optimum value unless the results are made available at the earliest possible date to the aircraft designer. The Air Research and Development Command is making special efforts in this connection to insure that the lessons learned from the total Structural Integrity Program described will be published as soon as possible.

It may interest you to note that the Wright Air Development Center is not satisfied with developing design criteria in the traditional sense. As shown in Figure 32, research efforts are under way in various phases of design criteria work to improve their validity and to get them to you at the earliest possible date.

As a result of recent cooperative effort between the National Aeronautics and Space Administration, the Department of Navy and the Department of Air Force criteria experts working together have evolved proposed engineering service life values for different types of aircraft. It is clearly realized that service life desired in any class of vehicle is dependent on many other factors such as actual

operational life required by the Operating Command involved. The times listed in Figure 33 have been made available to Air Force Headquarters as engineering estimates. In due course the Air Force will make available to industry desired aircraft service life values in order that the industrial designer will know better how to design future weapon systems.

This joint AMC-ARDC program to improve service life estimates, reduce fatigue incidence, develop better design criteria, etc., as just described, has been presented to higher authority and has received the personal approval of the Vice Chief of Staff, USAF, General Curtis E. LeMay. The General has assigned the program an extremely high priority. Thus, the basic program deficiencies originally mentioned have been corrected

In funding a part of this program, the Air Force is protecting the investment in SAC aircraft with a research and development type insurance policy. This will not only assist in protecting our first line fleet of today and tomorrow but will also aid in reducing the expensive "Fixit" program which could evolve over the next several years.

MATERIALS FATIGUE PROBLEMS

To be certain that flight vehicles of the future, using the typical double wall structural configurations in Figure 34, do not run into the same or similar structural integrity problems that exist in our high performance aircraft fleets of today, it became necessary to delve ever deeper into the basic causes of fatigue. This search made it clear that we do not know enough about the fundamentals of fatigue in materials and structure, both research and engineering wise. For example, the few application theories that do exist are challenged by experts in the field and sources of environmental materials fatigue have increased rapidly in terms of aero-space vehicle designs of tomorrow. It is clear that henceforth the fatigue problem must become not only an integral part of the structural design problem but also a part of operational utilization planning.

Let us look for a moment at the basic materials fatigue problem. As shown in outline sketch form in Figure 35 the mechanism of crack nucleation has received detailed study at the visual and microscopic levels. It has also been investigated at the atomic level. We may find it necessary to investigate fatigue at the sub-atomic level. We must have a better understanding of the process of crack nucleation. There is no completely satisfactory explanation of this mechanism today.

In another way we must now consider the relation of crack nucleation to the types of applied loading and environment to which future flight vehicles will be subjected (Figure 36). Certainly, the factors of radiation, particle bombardment, vacuum and corrosive atmospheres must be studied individually and collectively with that of increasing higher and lower temperature limits. Although some individual empirical relationships are available, we do not have a unified understanding of all the principles and fundamentals involved.

Further, such material properties, as outlined in Figure 37, affect the mechanism of crack nucleation. We are familiar to some extent with the effect of impurities encountered in large production runs of materials. We know much about the mechanical and magnetic properties of materials. But, if we combine

these with increasingly severe thermal limits, the effects of chemical environment, etc., we see an increasing need for a unified understanding of all these interactions that will work concurrently on materials and structure.

Further, we must be able to take the results of precisely controlled laboratory sine wave loading conditions on selected specimens and extrapolate these to the random loadings which occur in practice, as shown in Figure 38.

FUTURE GOALS

You have heard in brief form the ARDC portion of the joint AMC-ARDC program of "Structural Integrity for High Performance Aircraft" and have seen a glimpse of our future planning. But this in itself is not enough, we must also have a set of goals toward which to strive. Here then are several such goals we must reach at the earliest possible date.

We must learn how to emphasize and integrate the concept of structural integrity into the design mission of the flight vehicle. Structural integrity requirements must receive equal consideration with performance parameters as the military mission of the flight vehicle is evolved or changed.

We must learn how to profit more quickly from the structural knowledge gained in the design, fabrication, inspection and service operation of aircraft and missiles, if these data are to be made available to the aerospace vehicle designer in time to do him any real good.

We must learn how to obtain sufficient information from one test article which when supplemented with improved fatigue analysis methods will provide substantiation of structural strength and fatigue life of the flight article.

We must learn how to extend the application and interpretation of static and fatigue component tests through improved analysis methods to provide reliable indications of structural integrity of the full scale article.

We must learn in a technical as well as in a management sense how to speed the transition and control of the instrumentation and data handling phases of our program in the research and development areas to that of service engineering and maintenance.

We must learn how to compress the time it takes to acquire, reduce and release flight load data in final design criteria form as well as to rework it in a form useful to service engineering and maintenance managers and the operational planner.

We must learn how to attain increased selectivity in the application of instrumentation for recording flight load experience of all types without sacrificing the statistical reliability of the data obtained.

These seven goals can be, in my opinion, seven league boots to a successful resolution of our mutual structural integrity problem - now and in future.

My time is up so let me summarize by saying that:

The impact of fatigue on aircraft service life is a problem now and will continue to be in the foreseeable future. No modern high performance flight vehicle is immune. The technical approach recommended is mandatory if a rational basis for developing adequate structural integrity in current and future aerospace vehicles is to be achieved -- in time.

To do so, state of the art and frontiers of knowledge in this area must advance more rapidly than at present. Your support is urgently recommended.

FIGURE I

CONTRACTOR ESTIMATES OF SERVICE LIFE

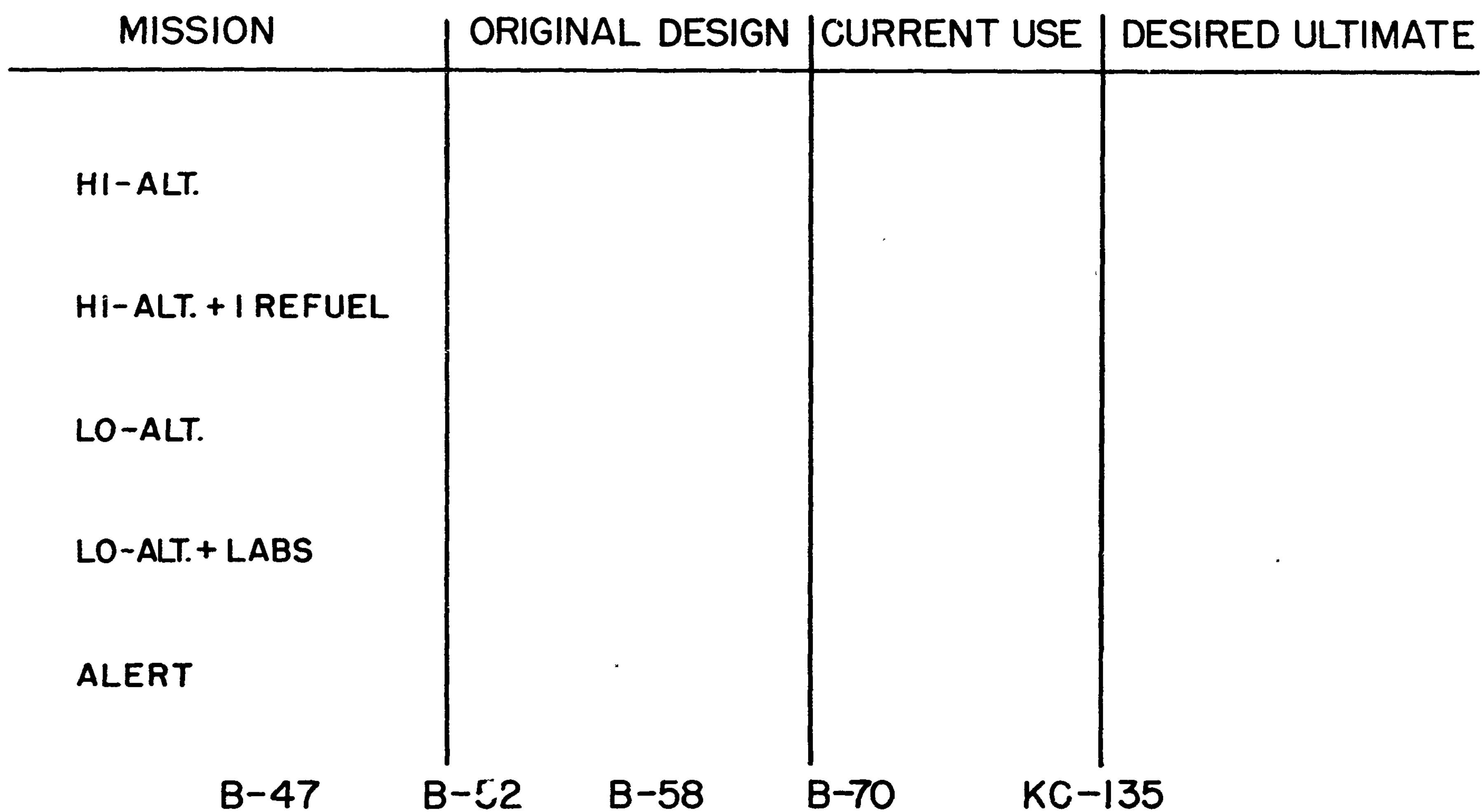


FIGURE 2

F-84F UNDERGOING STATIC TEST

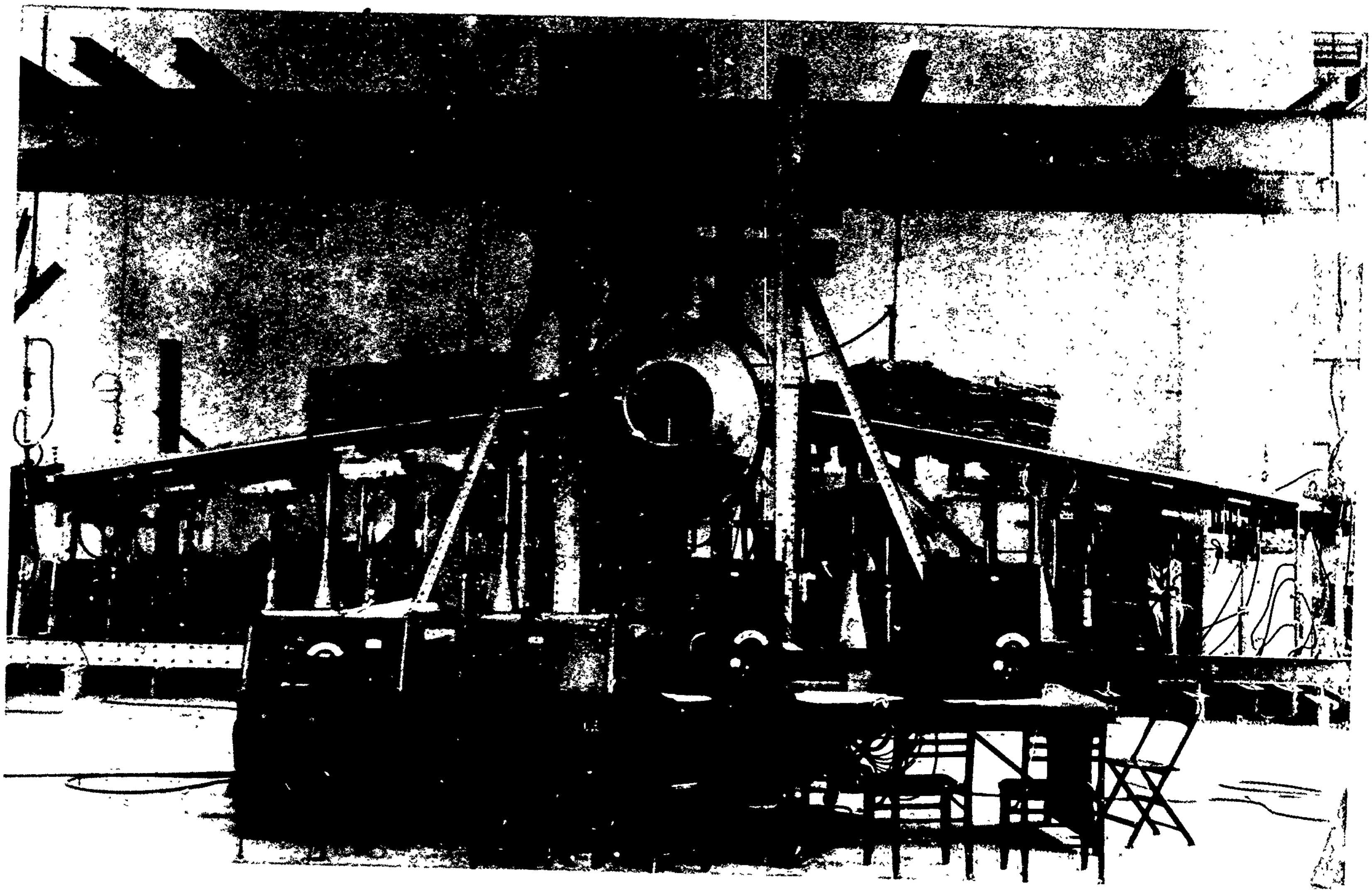


FIGURE 3

B-58 UNDERGOING STATIC TEST

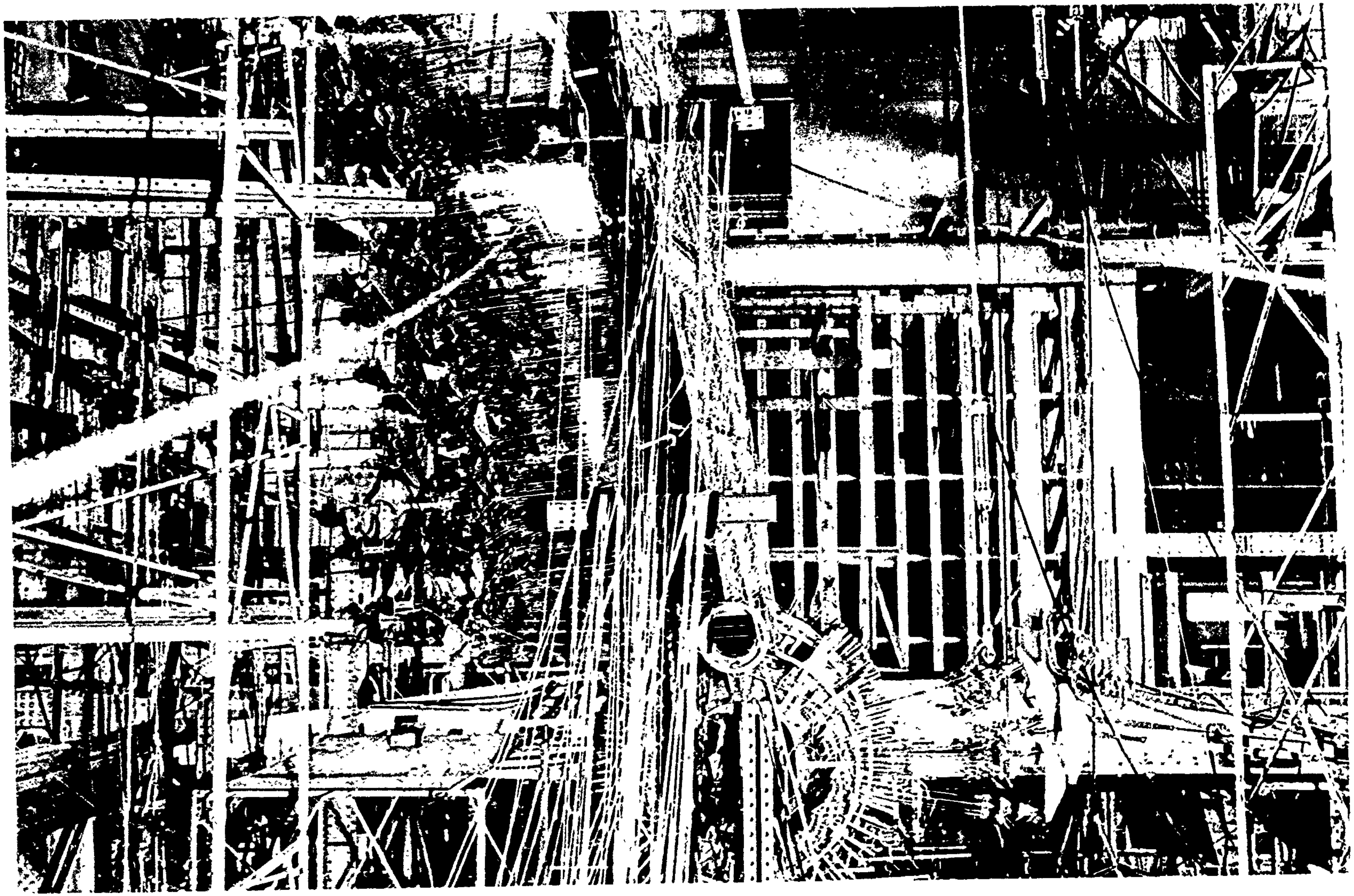


FIGURE 4

STATIC TEST

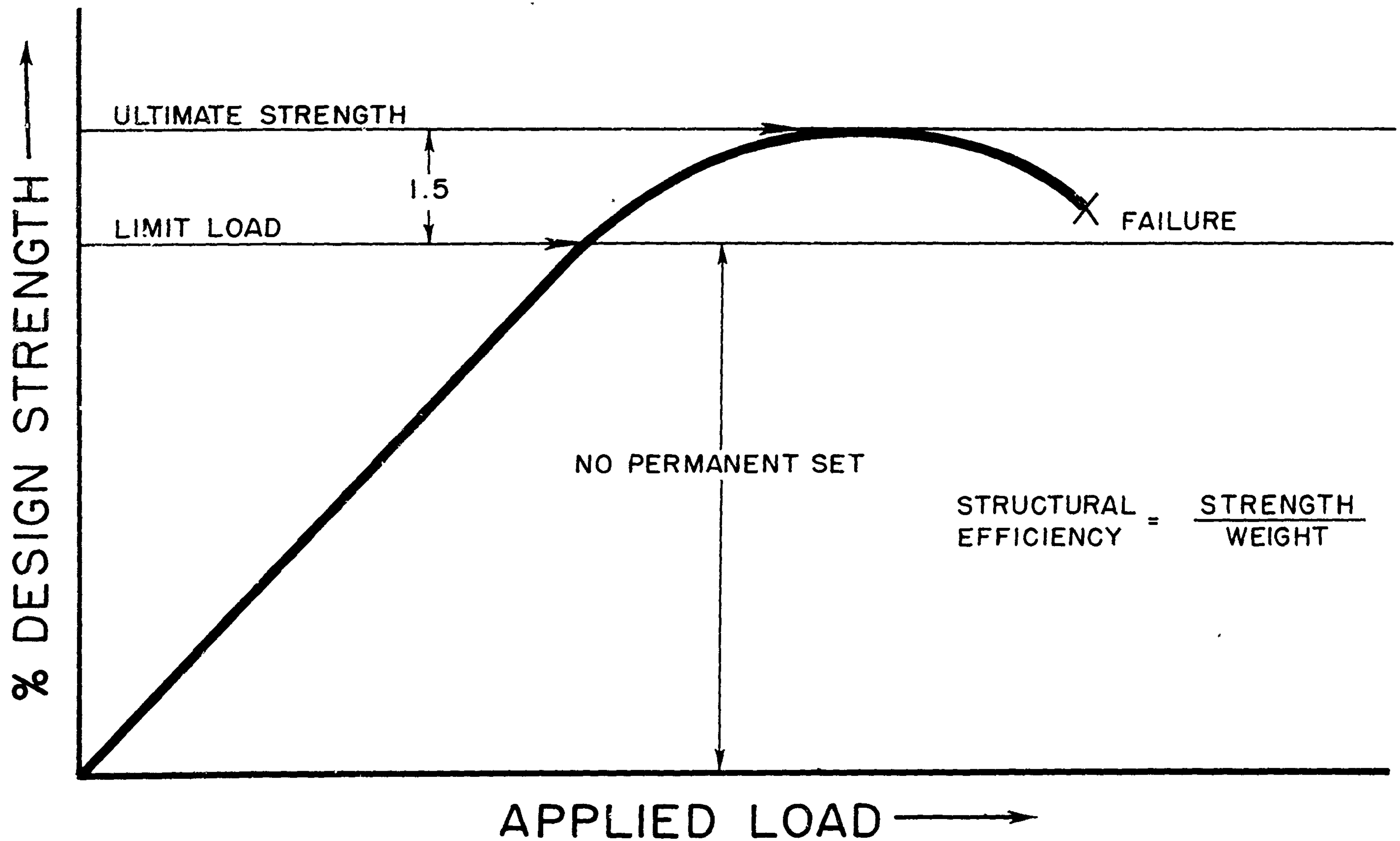


FIGURE 5

STEPS TO IMPROVE STRUCTURAL INTEGRITY OF HIGH PERFORMANCE AIRCRAFT

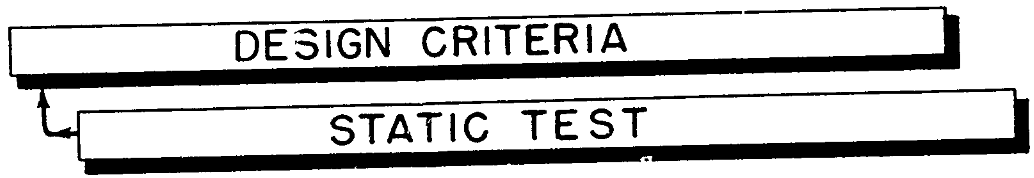


FIGURE 6

TYPICAL FLIGHT LOADS TEST PROGRAM

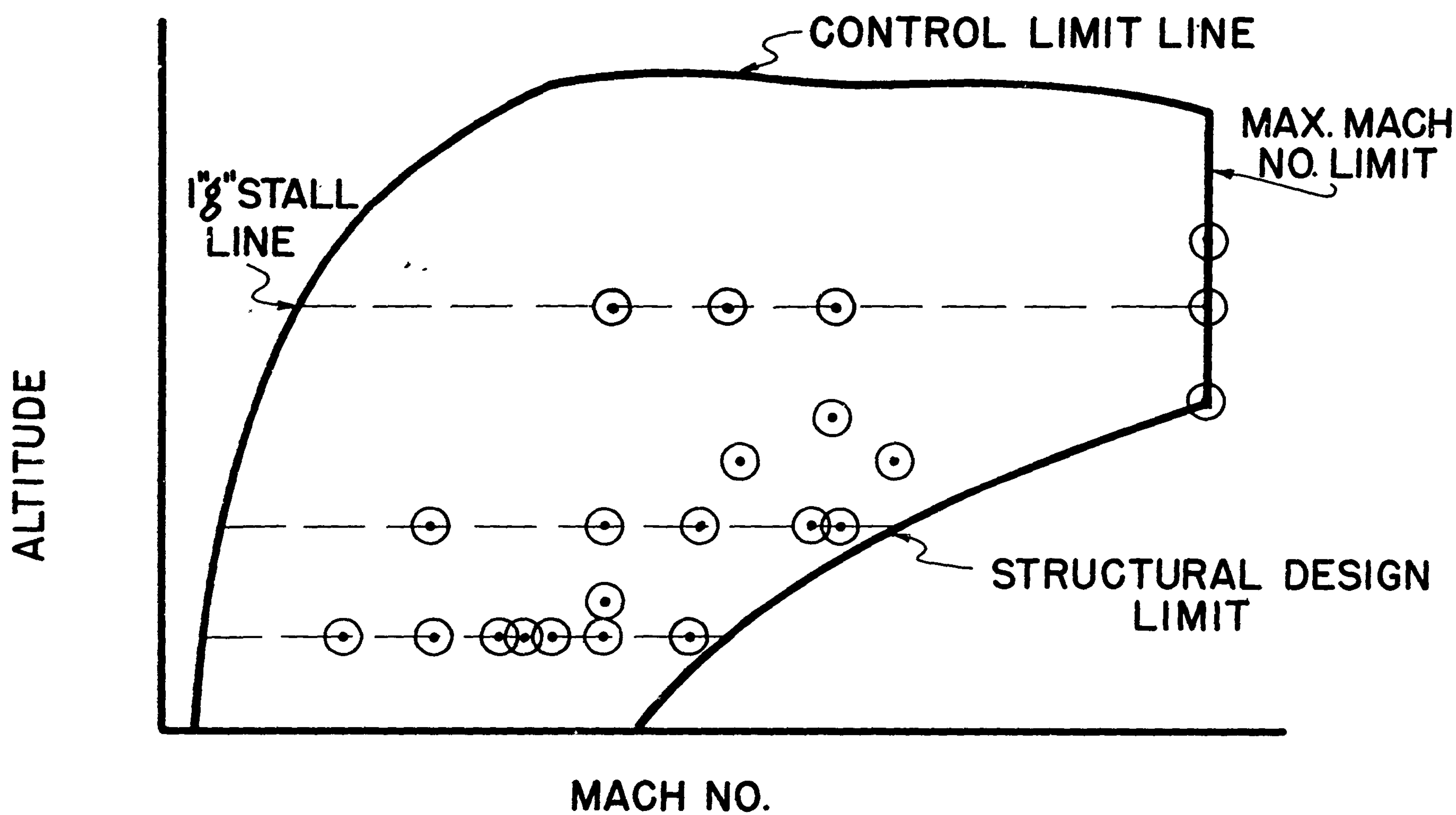


FIGURE 7

STEPS TO IMPROVE STRUCTURAL INTEGRITY OF HIGH PERFORMANCE AIRCRAFT

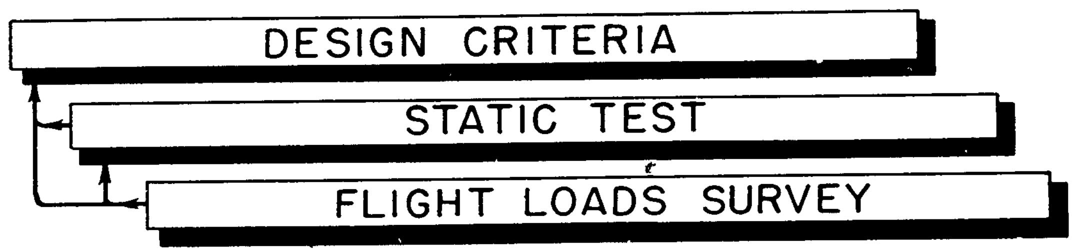


FIGURE 8

B-47 WING STRUCTURE

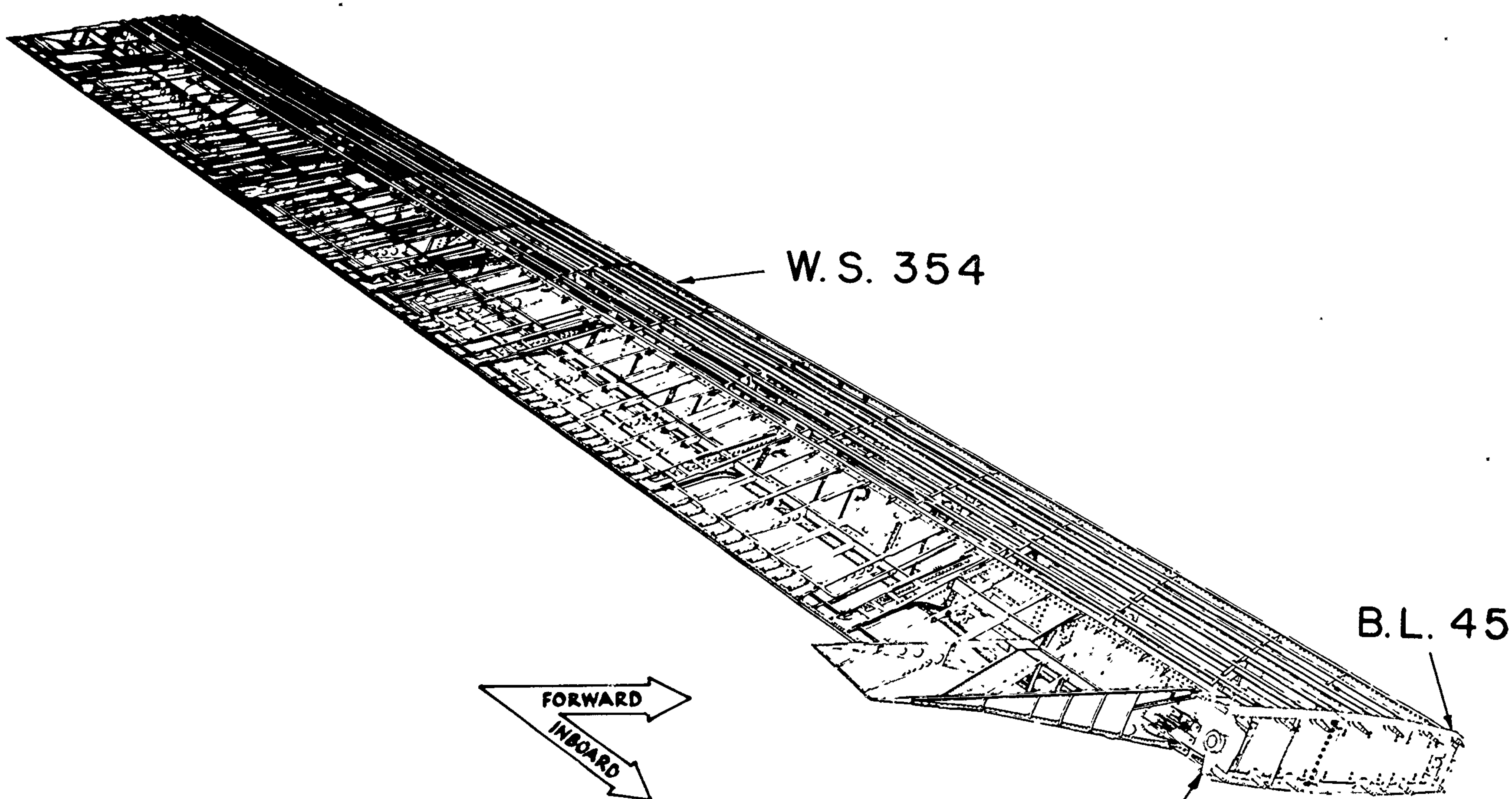


FIGURE 9

B-47 CYCLIC FATIGUE TEST SUMMARY

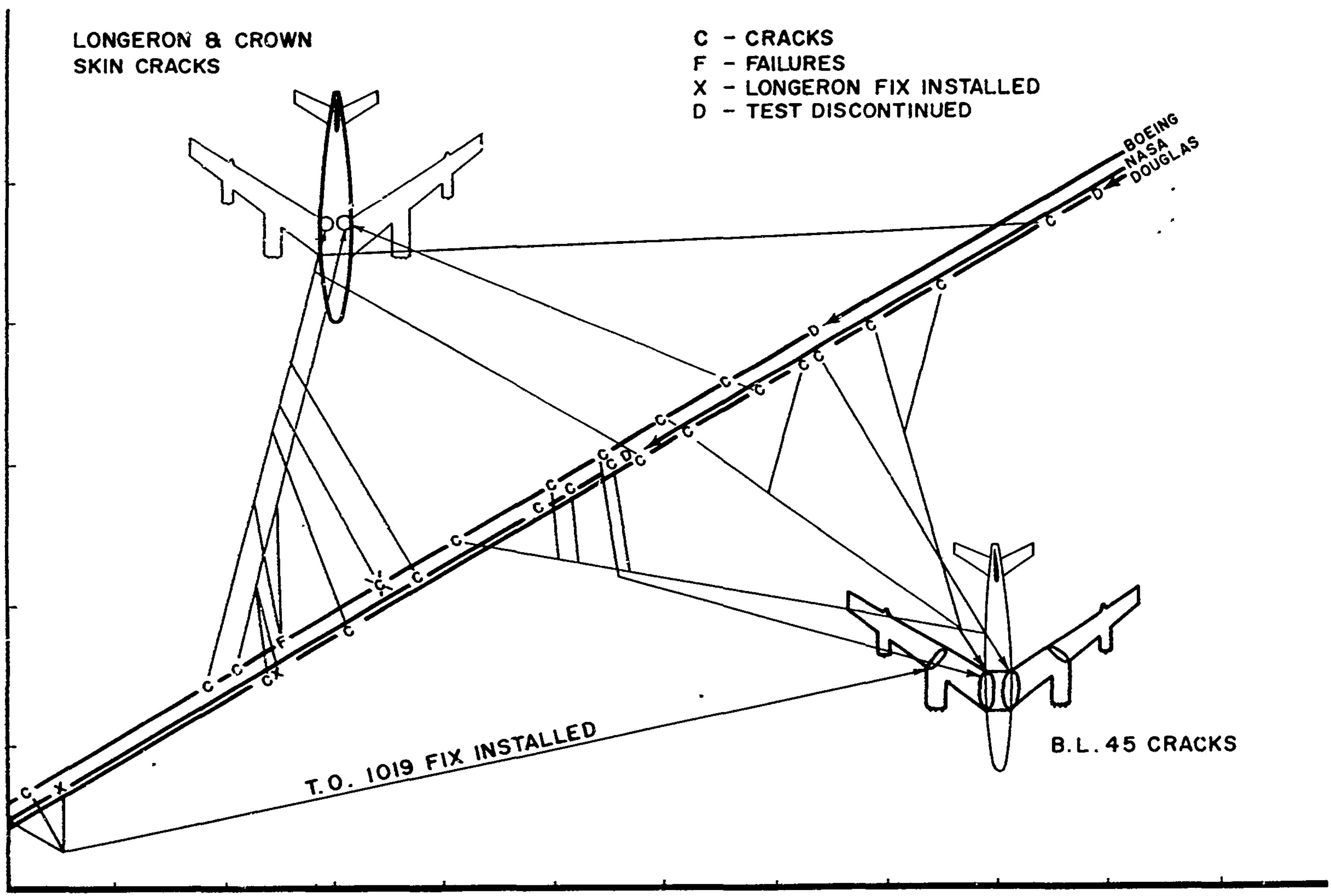
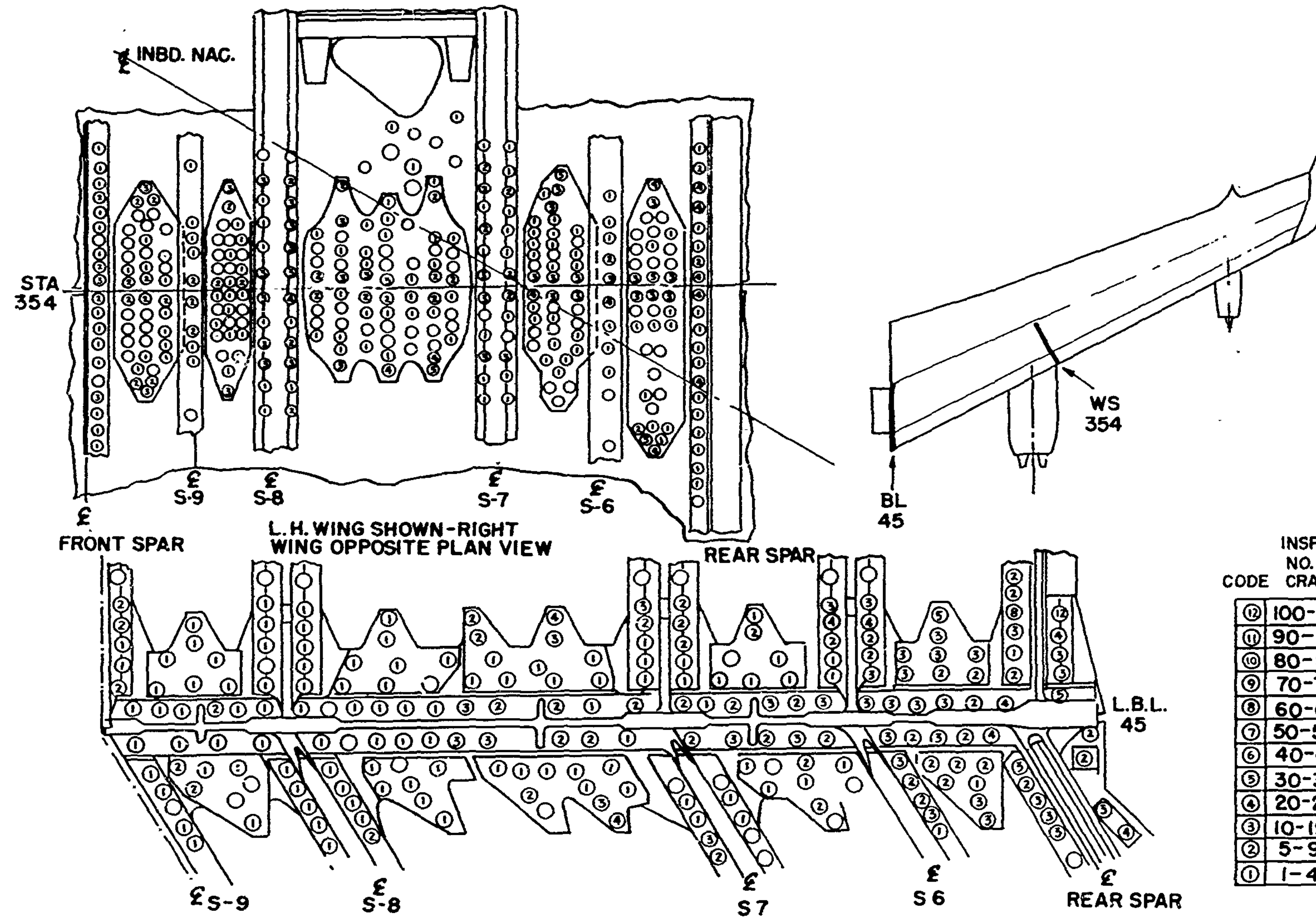


FIGURE 10

RESULTS OF B-47 T.O. INSPECTION



CODE	INSPECTION NO. OF CRACKS
12	100-109
11	90-99
10	80-89
9	70-79
8	60-69
7	50-59
6	40-49
5	30-39
4	20-29
3	10-19
2	5-9
1	1-4

FIGURE II

STEPS TO IMPROVE STRUCTURAL INTEGRITY OF HIGH PERFORMANCE AIRCRAFT

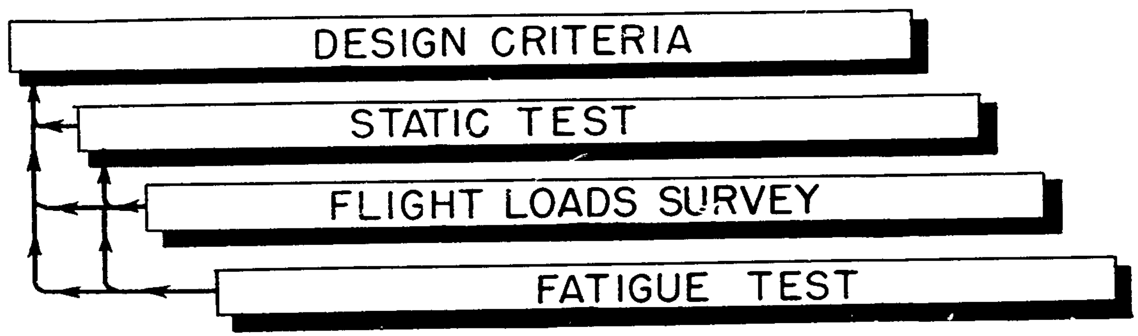
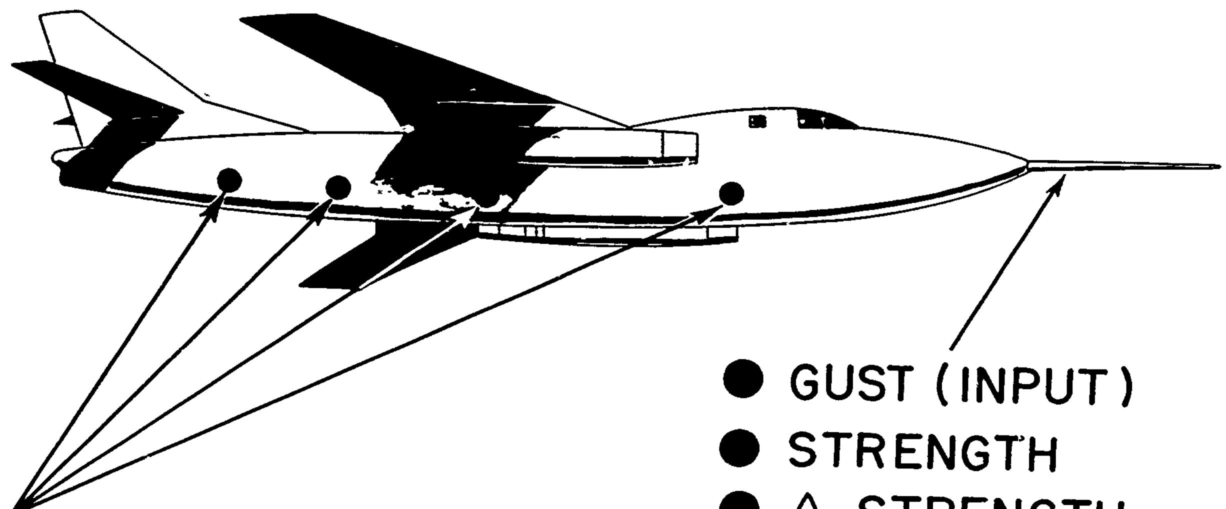


FIGURE 12

LO-ALT. GUST ENVIRONMENT

COMPLETE TIME VS:



- GUST (INPUT)
- STRENGTH
- △ STRENGTH
- FREQUENCY
- DIRECTION
- TERRAIN PHOTOS

- RESPONSE
- BENDING & SHEAR
- ACCELERATIONS
- PITCH, ROLL, YAW RATES
- PILOT REACTION

FIGURE 13

STEPS TO IMPROVE STRUCTURAL INTEGRITY OF HIGH PERFORMANCE AIRCRAFT

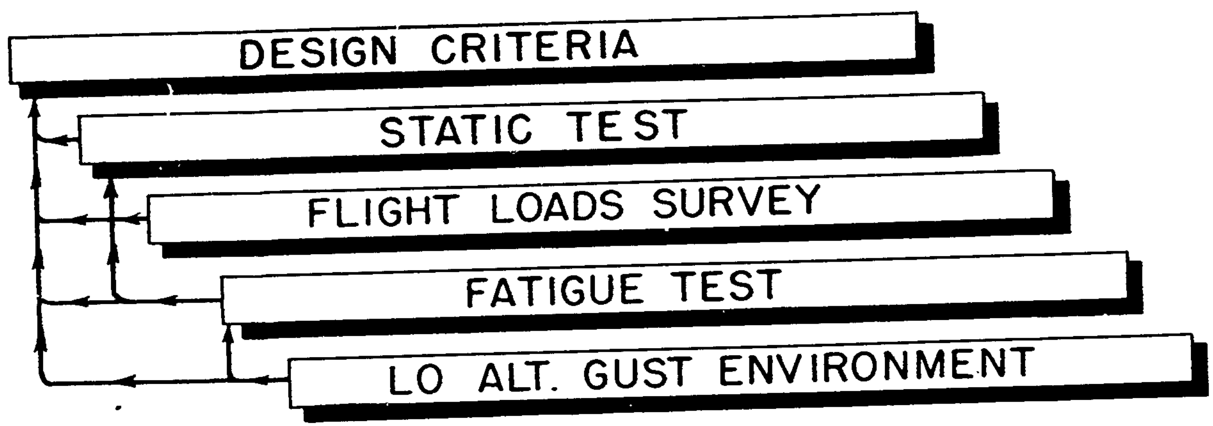
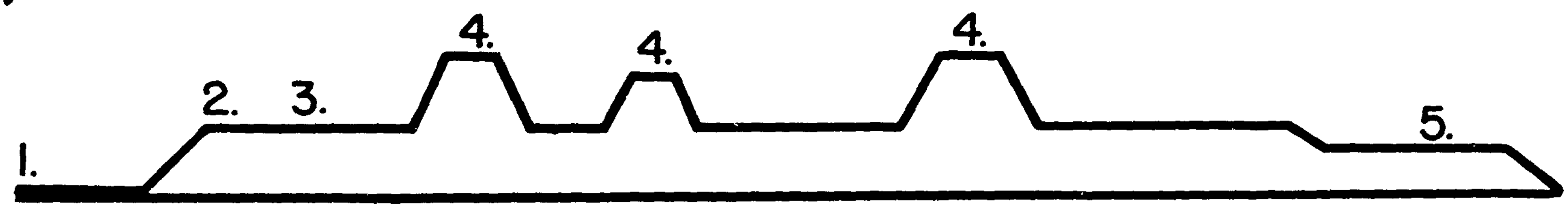


FIGURE 14

Typical B-47 LO-ALT. MISSION PROFILE

TANKS ON



CONDITIONS	GROSS WT.	TIME	ALT.	SPEED	REMARKS
1. TAKE-OFF & ACCELERATION					
2. INITIAL LEVEL-OFF					
3. LO-LEVEL NAVIGATION LEG					
4. BOMB-RUNS					
5. PILOT PROFICIENCY ITEMS INCLUDING TOUCH & GO's					

FY's 1959-'60-'61-'62 — NO. OF MISSIONS/YEAR/AIRCRAFT

FIGURE 15

STEPS TO IMPROVE STRUCTURAL INTEGRITY OF HIGH PERFORMANCE AIRCRAFT

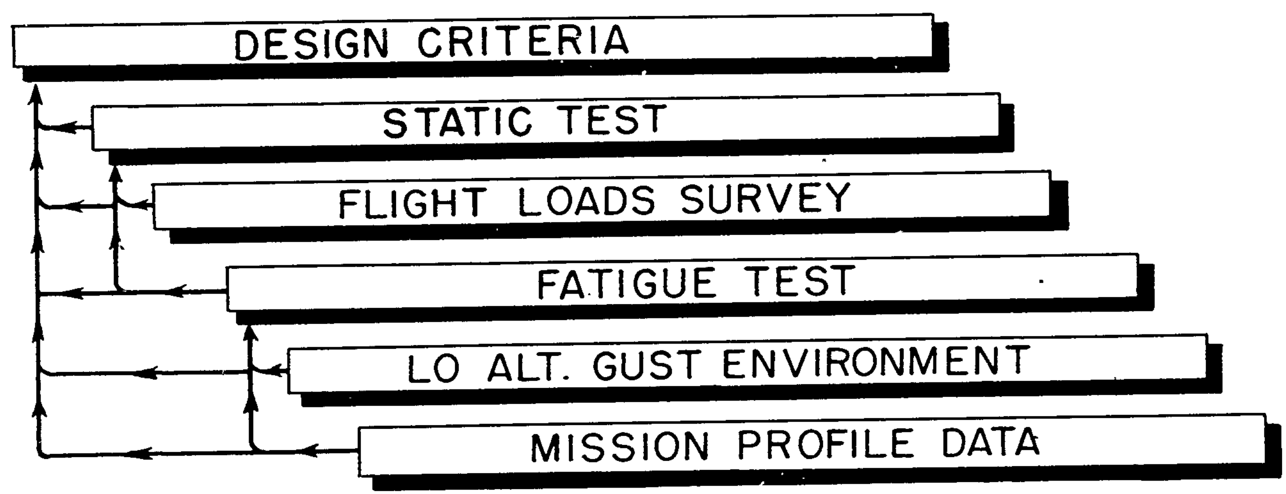


FIGURE 16

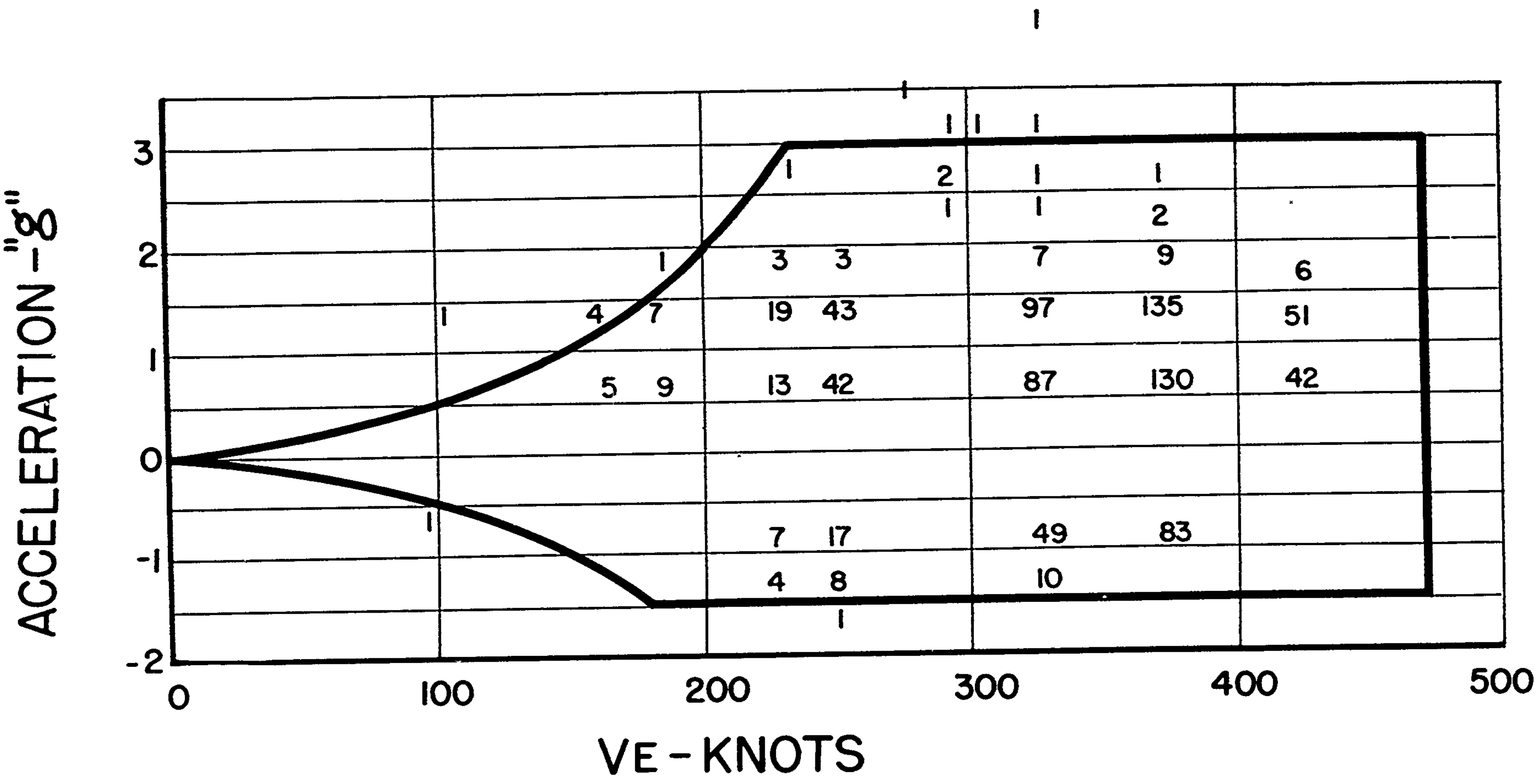
FLIGHT LOAD DATA

THREE CHANNEL RECORDER

DATA CHANNEL	PRESENTATION	"PAYOFF"
V-IMPACT PRESSURE	q_c	I. ONE DEGREE FREEDOM NORMAL ACCELERATION
G-NOR. ACCELERATION	n_z — ACCELERATION	
H-STATIC PRESSURE	q_D	II. BETTER DESIGN CRITERIA ON BASIS OF ONE FLIGHT PARAMETER
VS		
T-TIME	t	

FIGURE 17

TYPICAL V-G DIAGRAM



OLD STYLE 3-CHANNEL RECORDER:

10 ACCELERATION RANGE
5 AIRSPEED RANGES
5 ALTITUDE RANGES

FIGURE 18

STEPS TO IMPROVE STRUCTURAL INTEGRITY OF HIGH PERFORMANCE AIRCRAFT

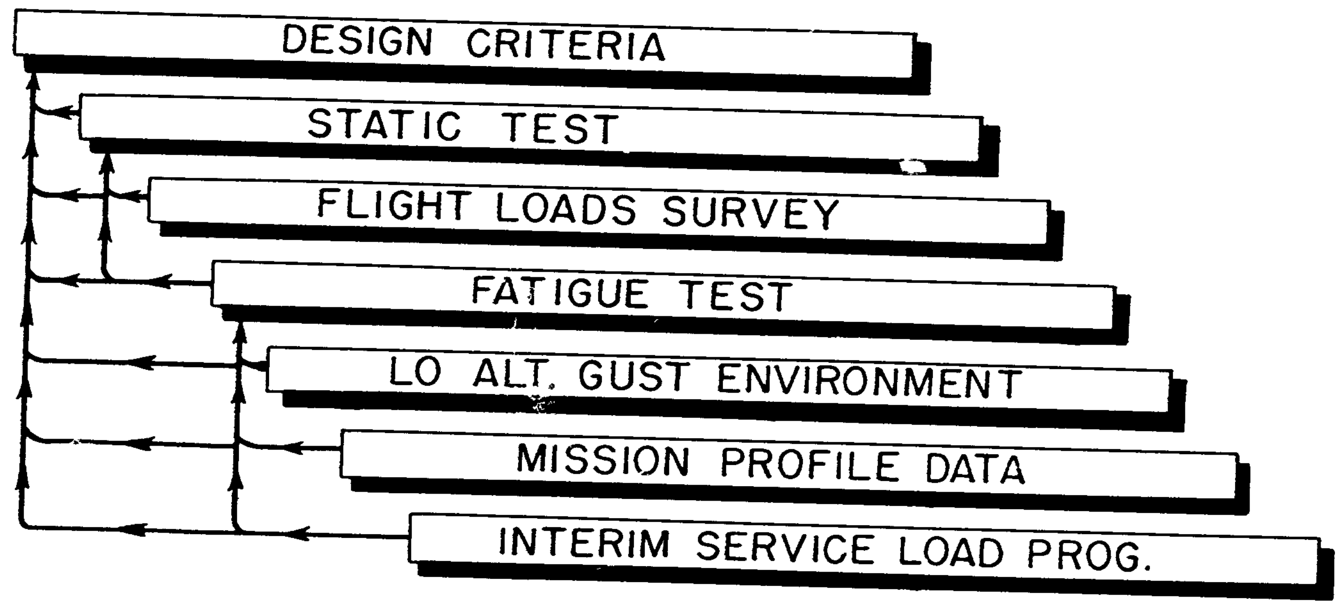


FIGURE 19

FLIGHT LOAD DATA

DATA CHANNEL		PRESENTATION	"PAYOFF"
		3 CHANNEL RECORDER	
V-IMPACT PRESSURE	q_c	ACCELERATION	I. ONE DEGREE OF FREEDOM – NORMAL ACCELERATION
G-NOR. ACCELERATION	n_z		II. BETTER DESIGN CRITERIA ON BASIS OF ONE PARAMETER
H-STATIC PRESSURE	q_D		
T-TIME	t		
		8 CHANNEL RECORDER	
LONGITUDINAL ACCEL.	n_x	LINEAR ACCEL.	I. SIX DEGREES OF FREEDOM A. PREDICTION & OCCURRENCE OF ABOVE
TRANSVERSE ACCEL.	n_Y	LINEAR ACCEL. VS ANGULAR VEL.	
ROLLING VELOCITY	p	CROSS COUPLING EFFECTS	II. BETTER DESIGN CRITERIA IN TERMS OF ALL CRITICAL DEGREES OF FREEDOM OF VEHICLE IN FLIGHT
PITCHING VELOCITY	q	ANGULAR MOTIONS	
YAWING VELOCITY	r	ANGULAR ACCEL.	

FIGURE 20

8 CHANNEL RECORDER PROGRAM

1942	3 CHANNEL VGH RECORDER PROGRAM INITIATED.
1954	NACA APPROVED CHANGEOVER FROM 3 TO 8 CHANNEL RECORDING SYSTEM.
1955	ARDC DEVELOPS 8 CHANNEL RECORDER IN COORDINATION WITH NAVY DEPT.
1956	THREE STUDY CONTRACTS AWARDED
1957	STUDY CONTRACTS COMPLETED
	CONTRACT AWARDED TO DEVELOP 8 CHANNEL RECORDER
1958	DELIVERY OF THREE PROTOTYPES FOR TESTING
	CONTRACT AWARDED FOR 20 SERVICE TEST ARTICLES
1959	PROTOTYPE TESTING OF 8 CHANNEL RECORDER PROGRESSING

FIGURE 21

STEPS TO IMPROVE STRUCTURAL INTEGRITY OF HIGH PERFORMANCE AIRCRAFT

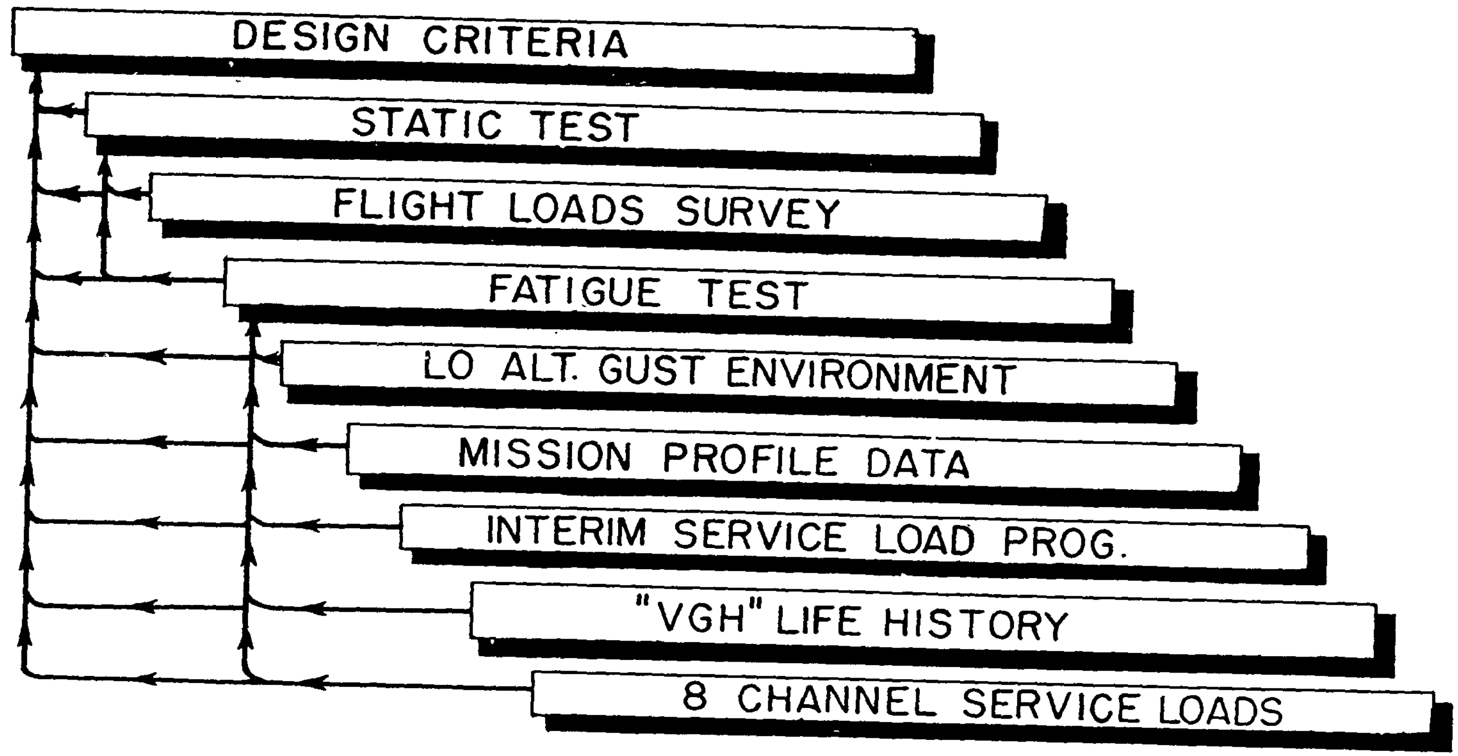


FIGURE 22

TYPICAL NOISE LEVELS

PRESSURE PSI	DB	
29	200	
2.9	180	LARGE BOOST ROCKET
.29	160	CLOSE TO JET ENG.@ T.O. POWER
.029	140	BEGINNING STRUCTURAL FATIGUE (THRESHOLD OF PAIN)
.0029	120	
.00029	100	
.000029	80	
.0000029	60	
.00000029	40	
.000000029	20	
.0000000029	0	THRESHOLD OF HEARING

FIGURE 23

EXTERNAL NOISE LEVELS OF MISSILE DURING LAUNCH

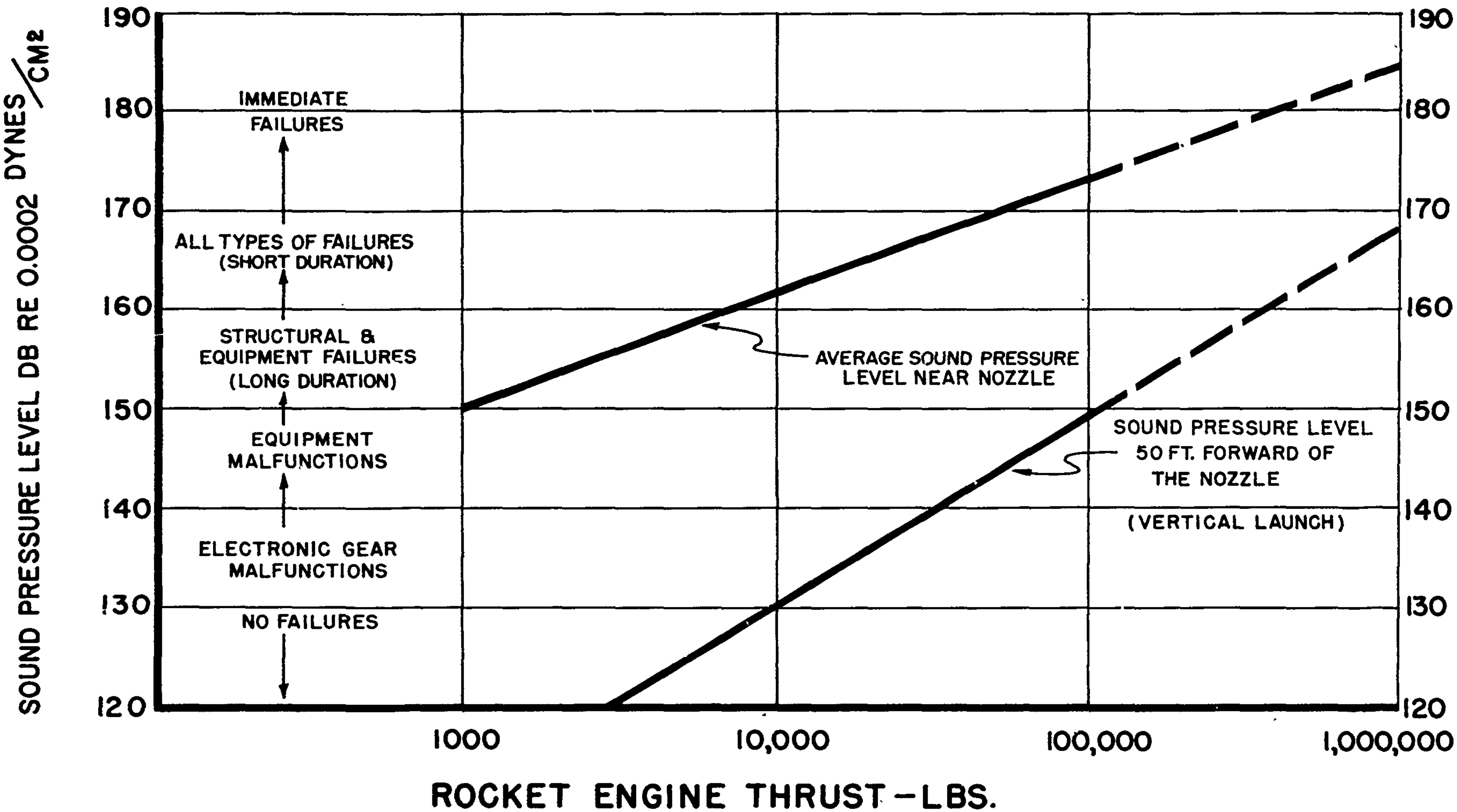


FIGURE 24

BOUNDARY LAYER NOISE

FUTURE FLIGHT REGIMES

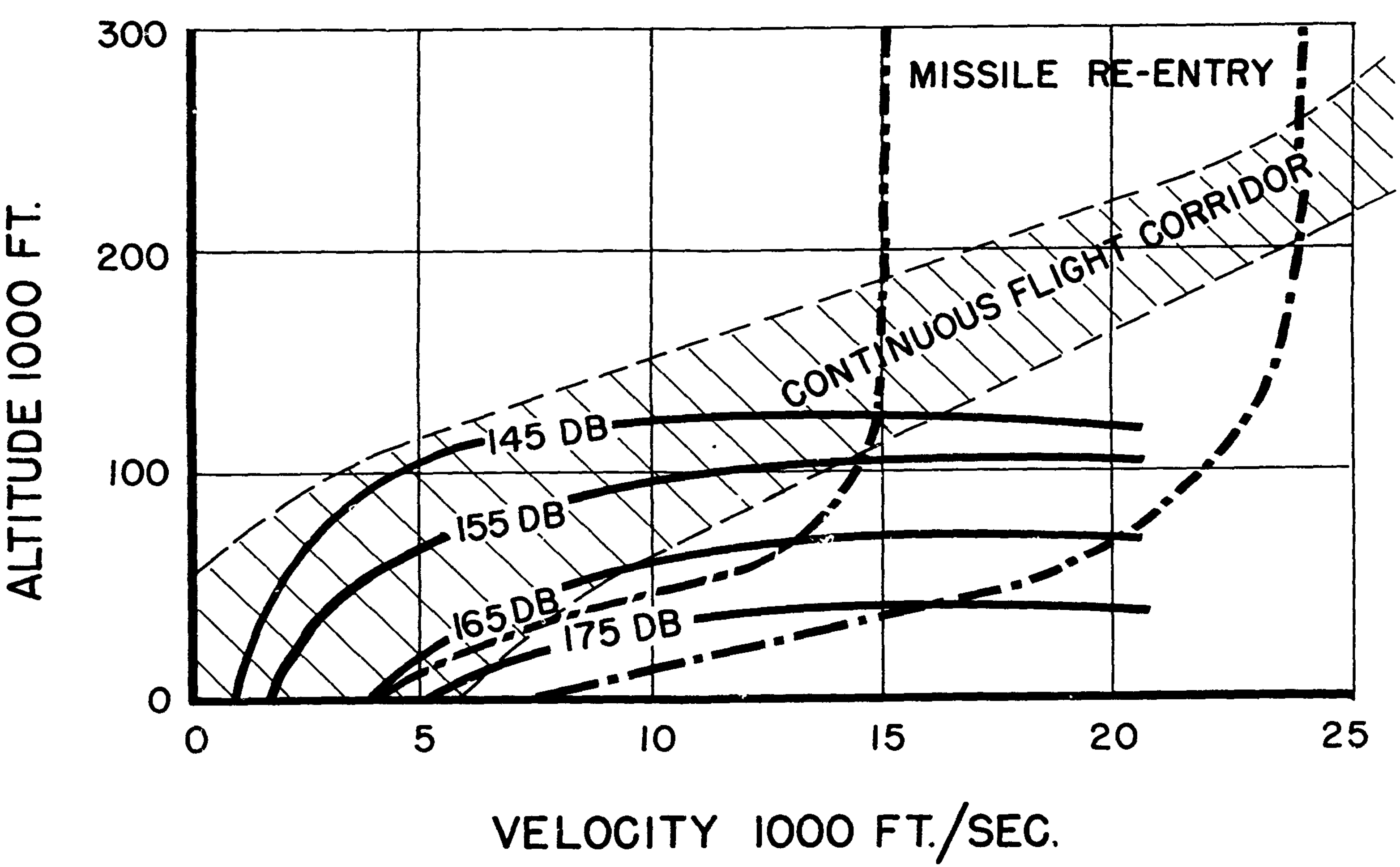


FIGURE 25

TURBULENT FLOW

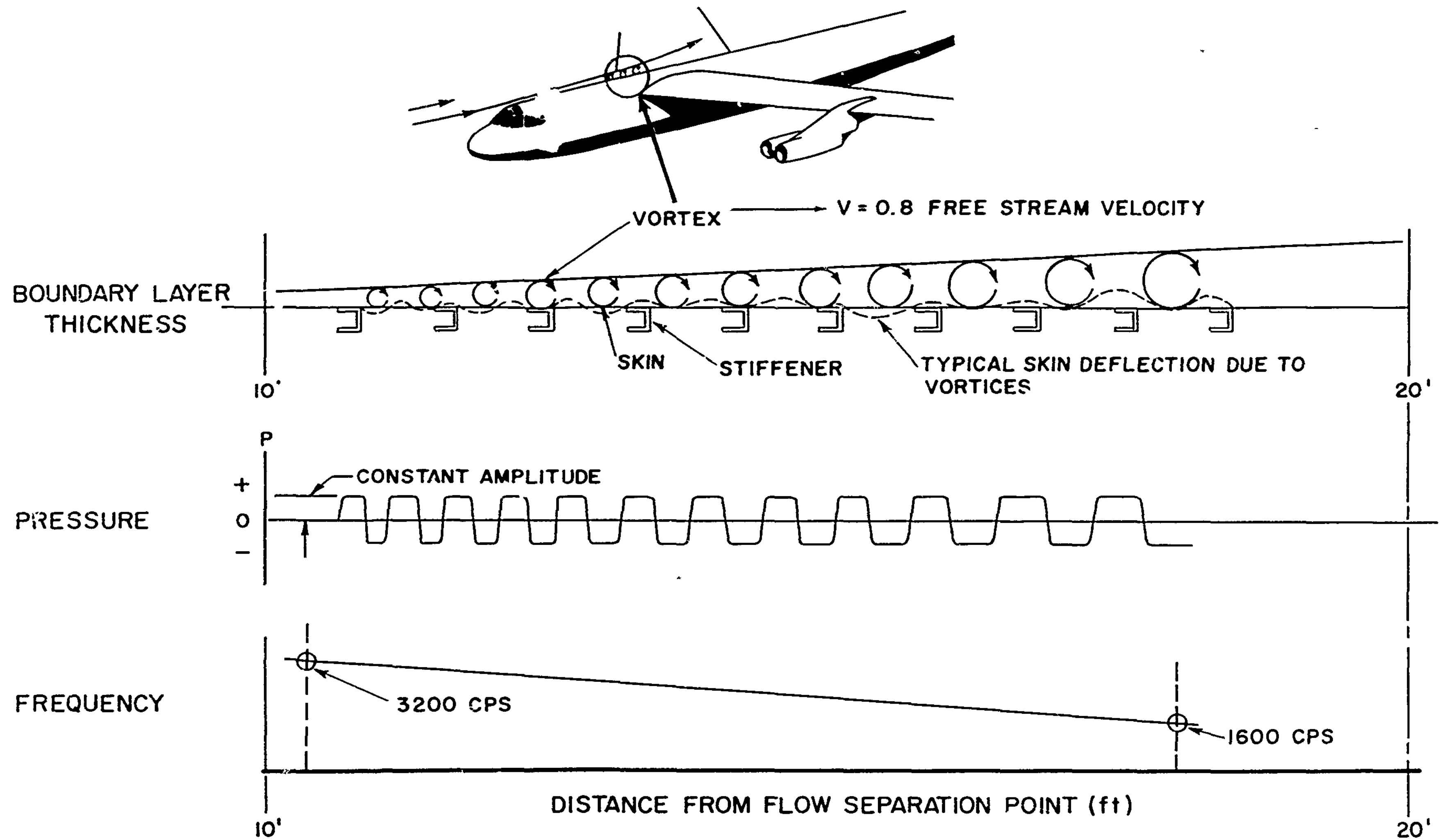


FIGURE 26

STEPS TO IMPROVE STRUCTURAL INTEGRITY OF HIGH PERFORMANCE AIRCRAFT

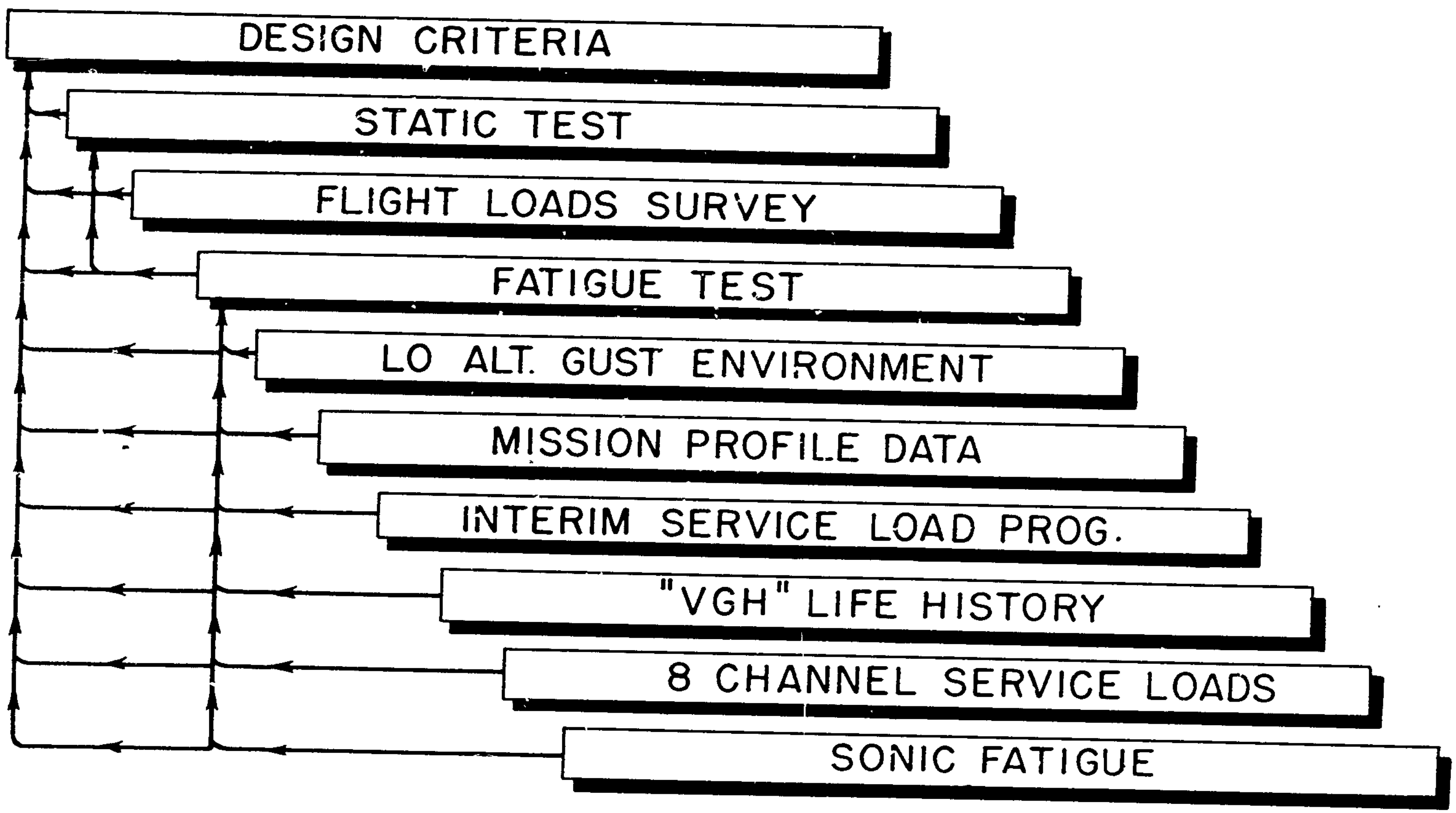


FIGURE 27

SPECIFIC STRENGTH vs TEMPERATURE

SHEET ALLOYS - ULTIMATE TENSILE

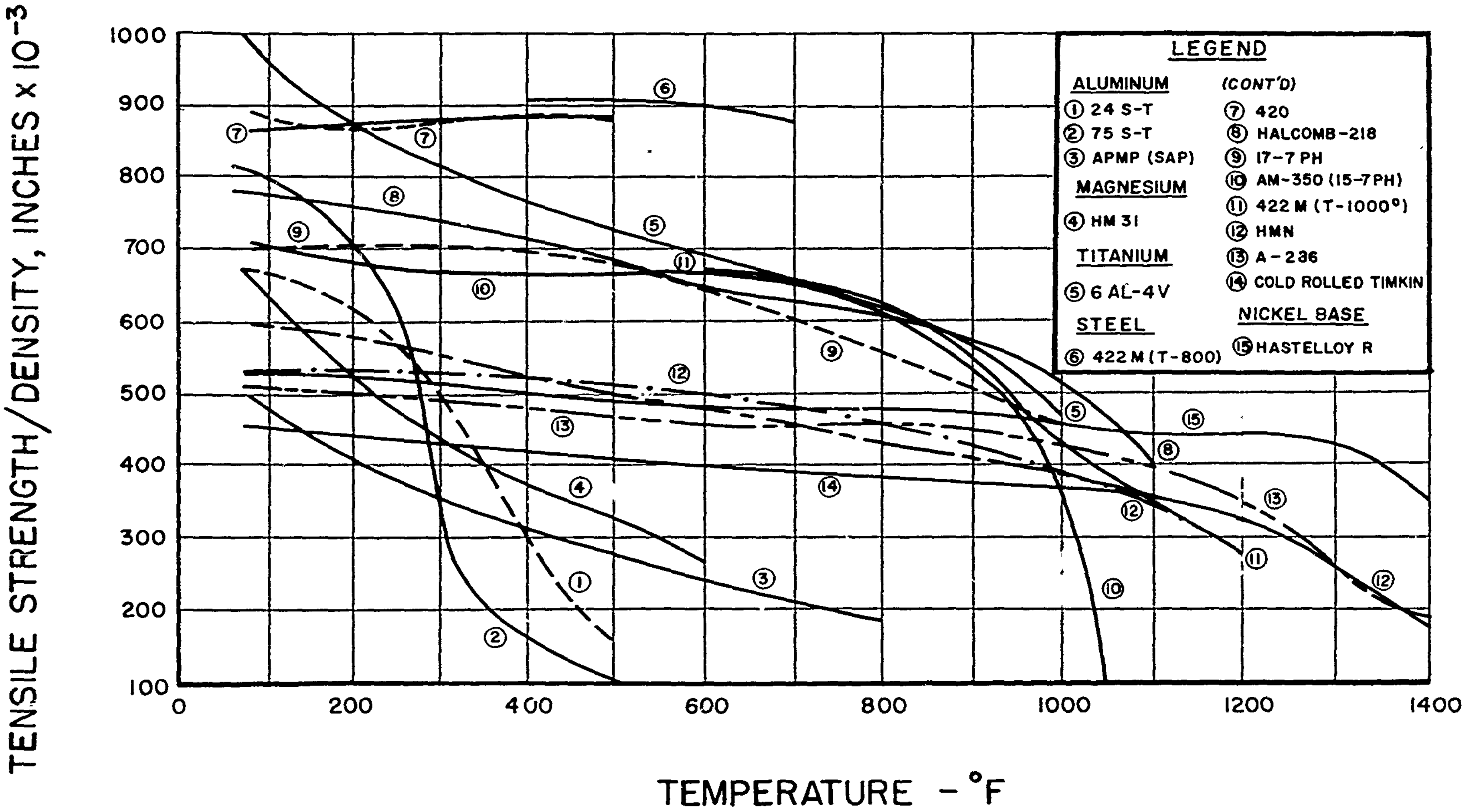


FIGURE 28

REFRACTORY METALS POTENTIAL

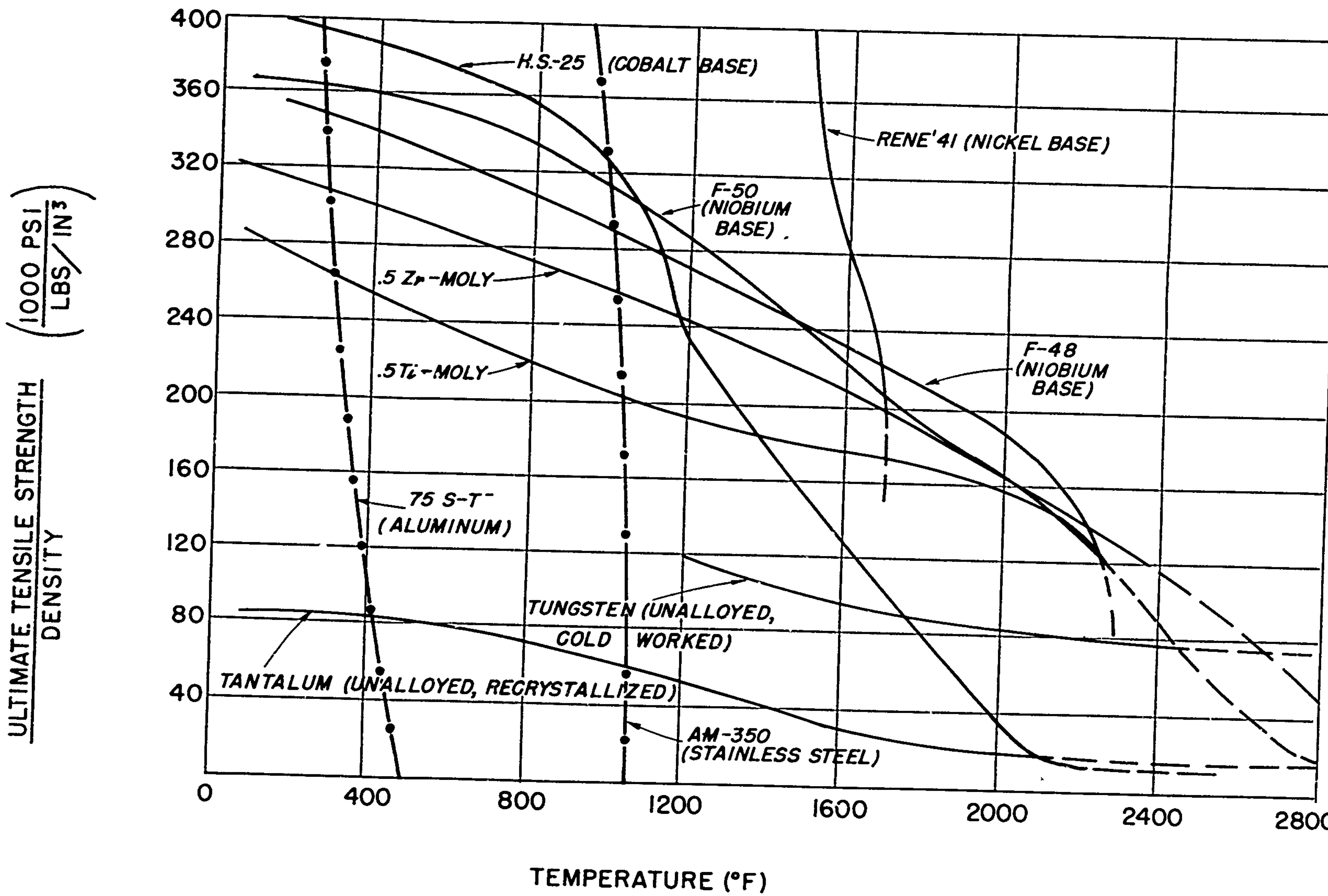


FIGURE 29

AVERAGE STRESS RUPTURE DATA
10 HOUR STRENGTH

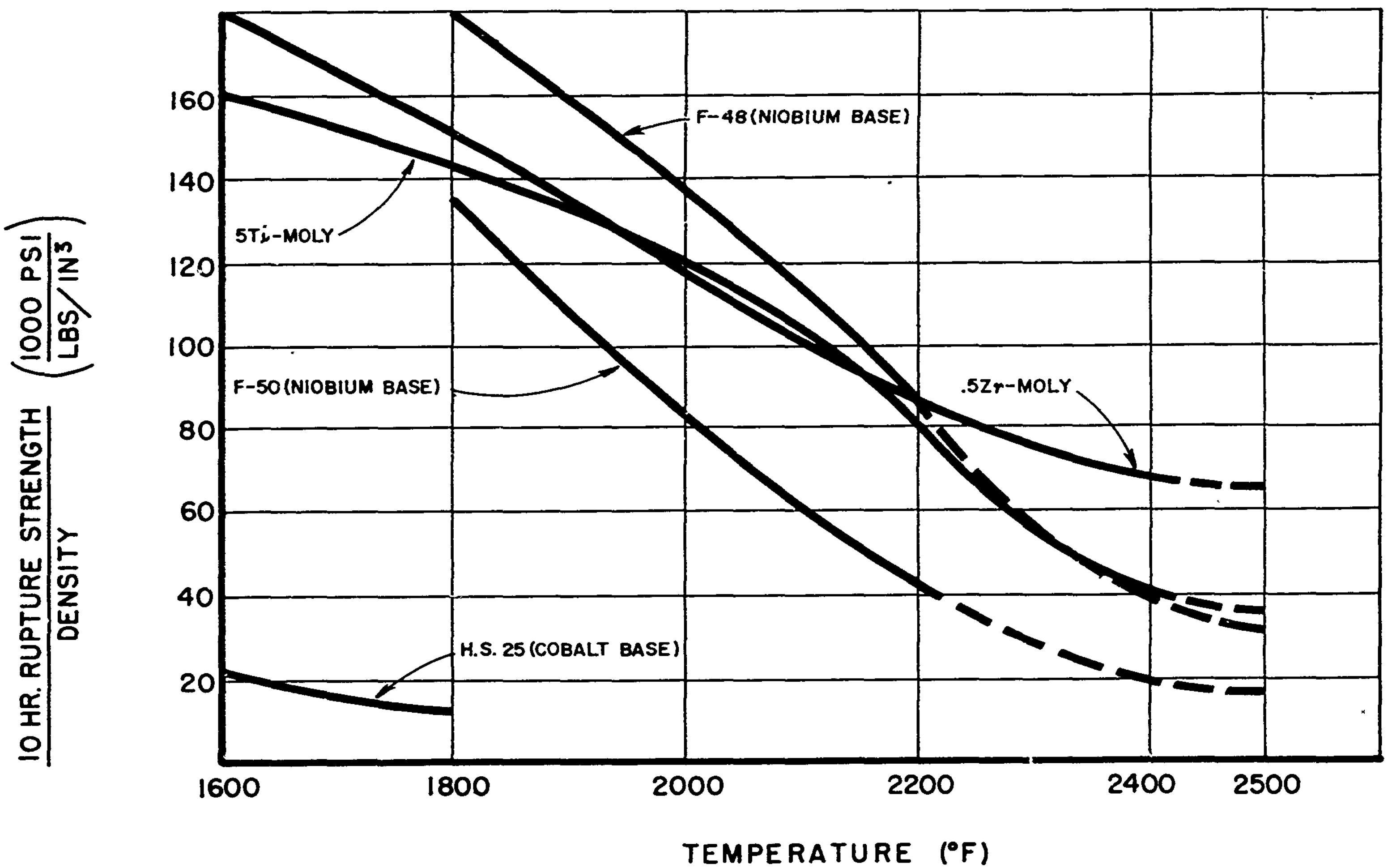


FIGURE 30

STRUCTURAL COMPONENTS DEVELOPMENT SPECTRUM
1959 thru 1965

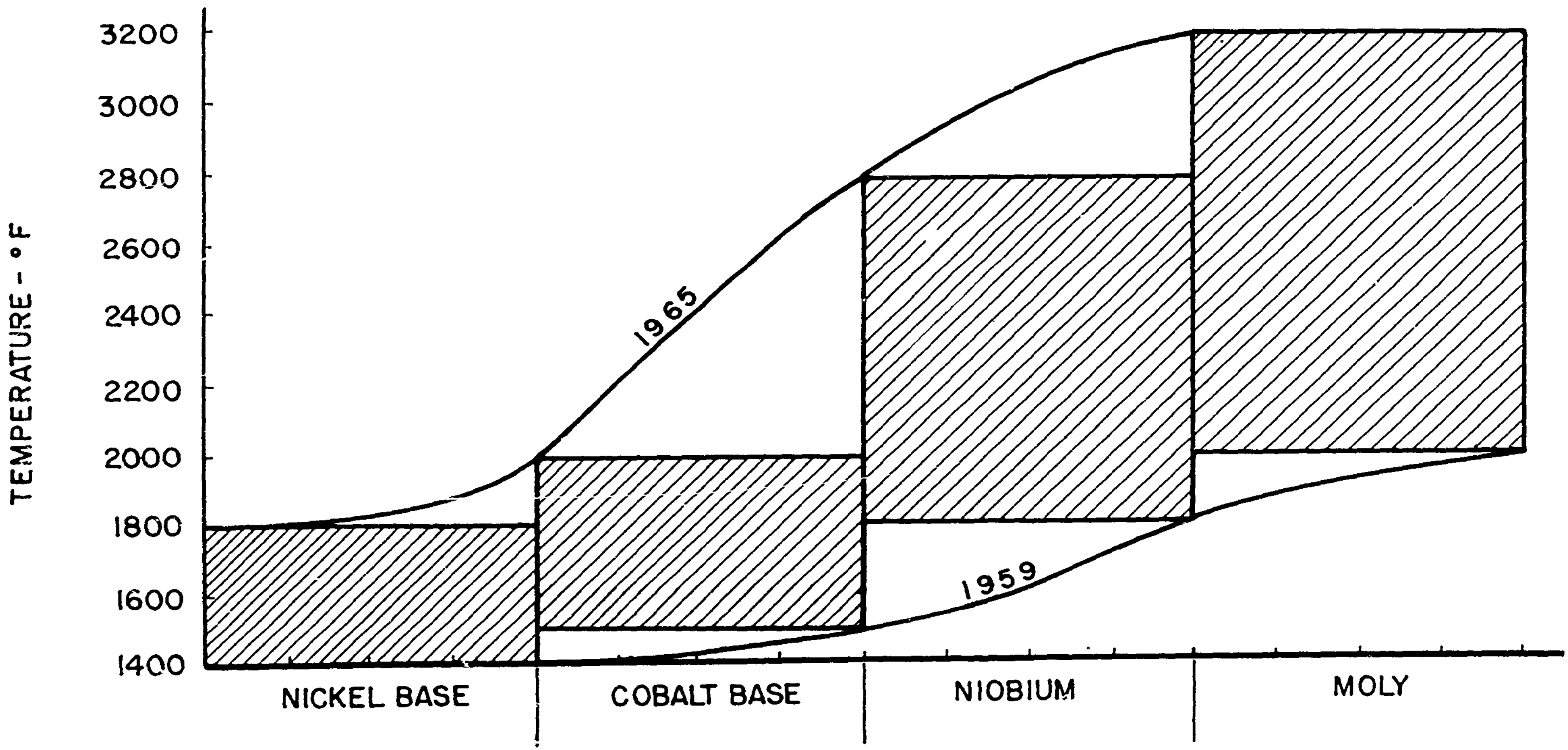


FIGURE 31

STEPS TO IMPROVE STRUCTURAL INTEGRITY OF HIGH PERFORMANCE AIRCRAFT

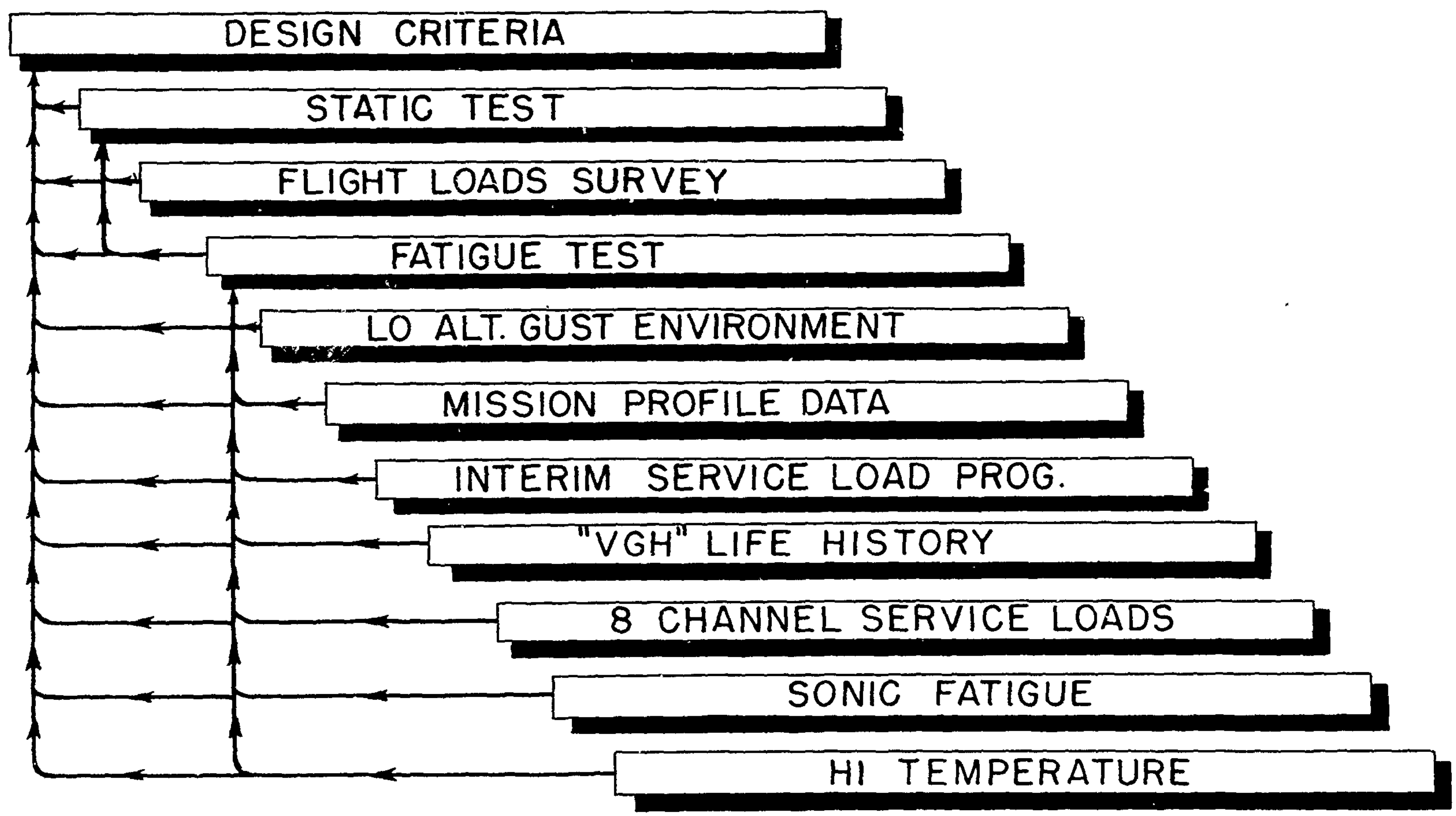


FIGURE 32

DESIGN CRITERIA RESEARCH ACTIVITY

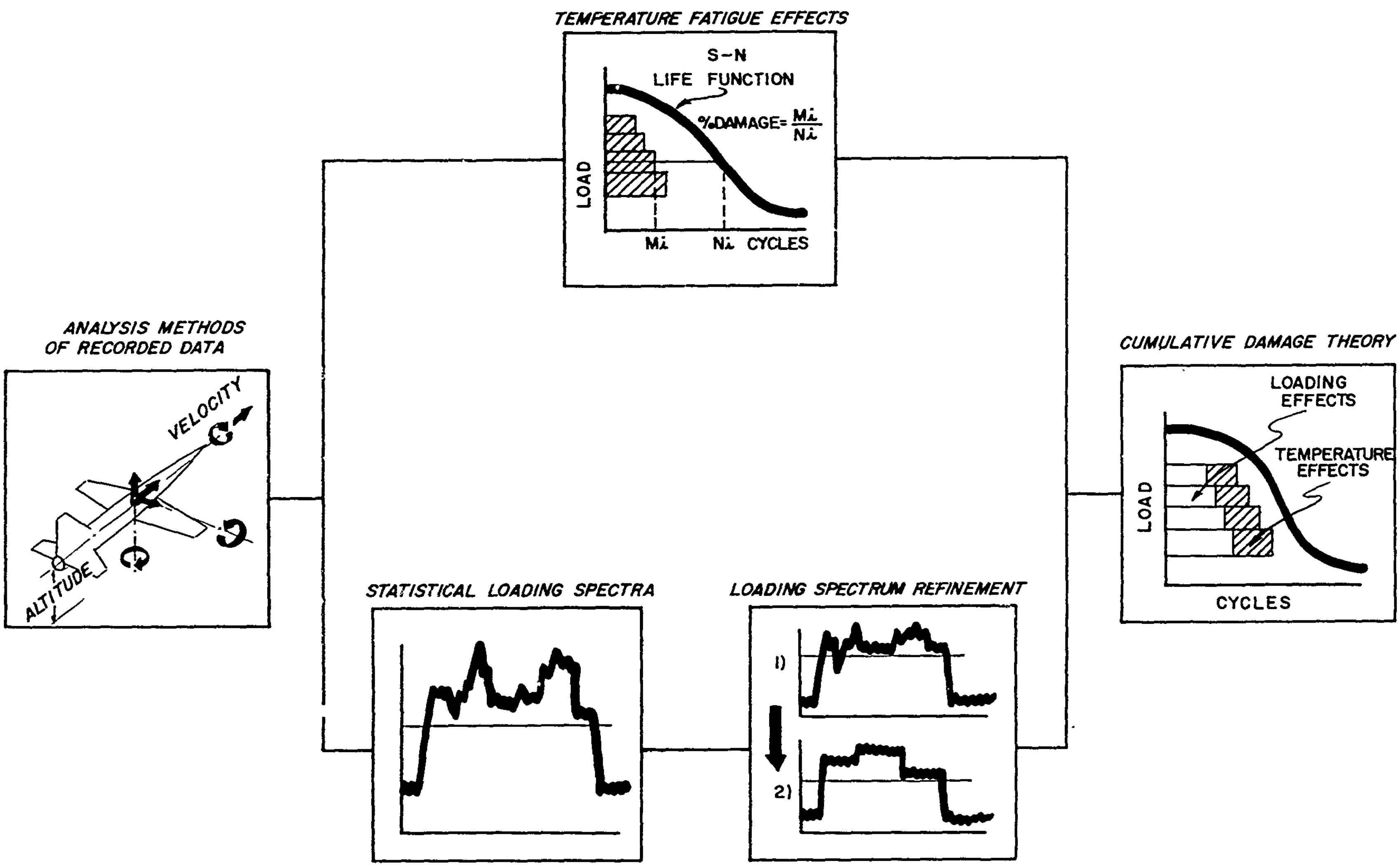


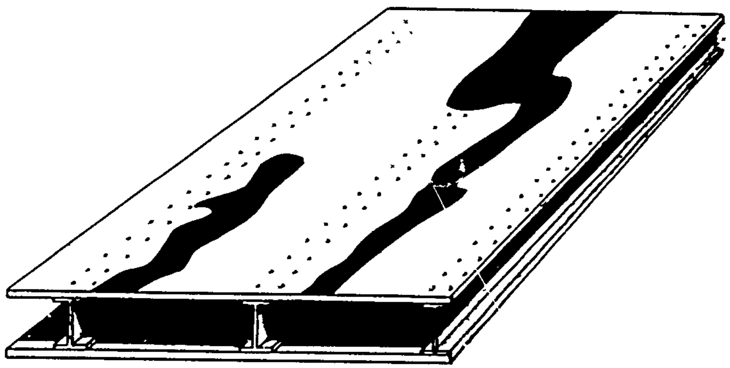
FIGURE 33

SERVICE LIFE PROPOSED FOR AIRCRAFT STRUCTURAL DESIGN

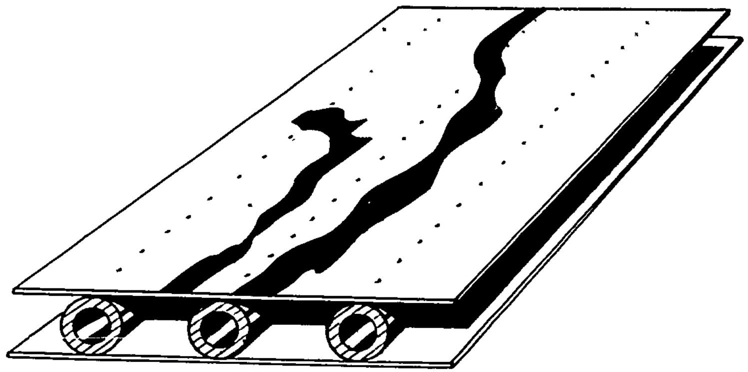
AIRCRAFT TYPE	FLYING HOURS	LANDINGS
FIGHTERS	4000	5200
BOMBER (ATTACK)	5000	2500
BOMBER (MED. HEAVY)	10000	5000
CARGO (TRANSPORT)	30000	15000
CARGO (ASSAULT)	10000	15000
LIAISION	8000	20000
TRAINER	8000	20000
TRAINER (FIGHTER)	4000	5200

FIGURE 34

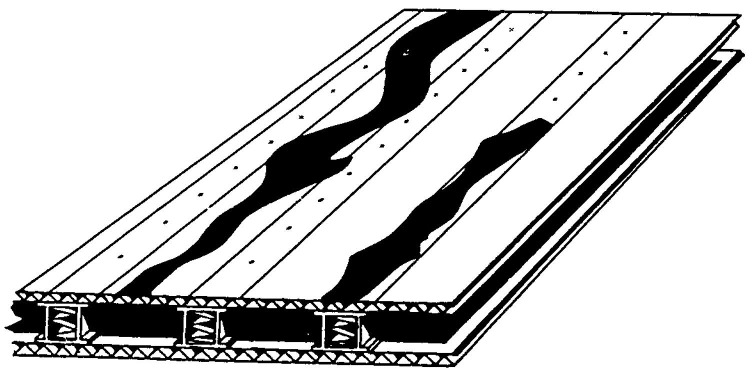
FUTURE AERO SPACE VEHICLES PROPOSED STRUCTURAL CONFIGURATIONS



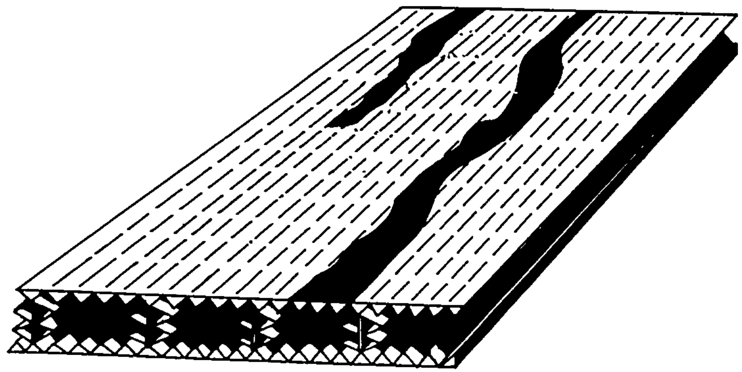
PRESSURE STABILIZATION



FLUID STOWAGE



FAIL SAFE DESIGN

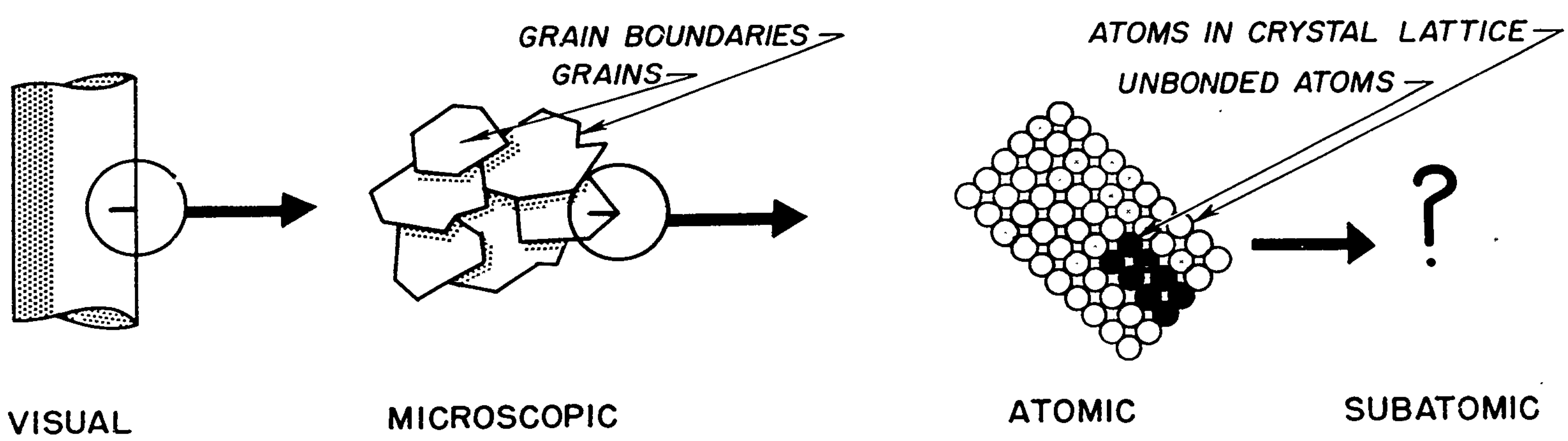


DESIGN FLEXIBILITY

FIGURE 35

FUNDAMENTAL QUESTION:

WHAT IS THE MECHANISM OF CRACK NUCLEATION ?

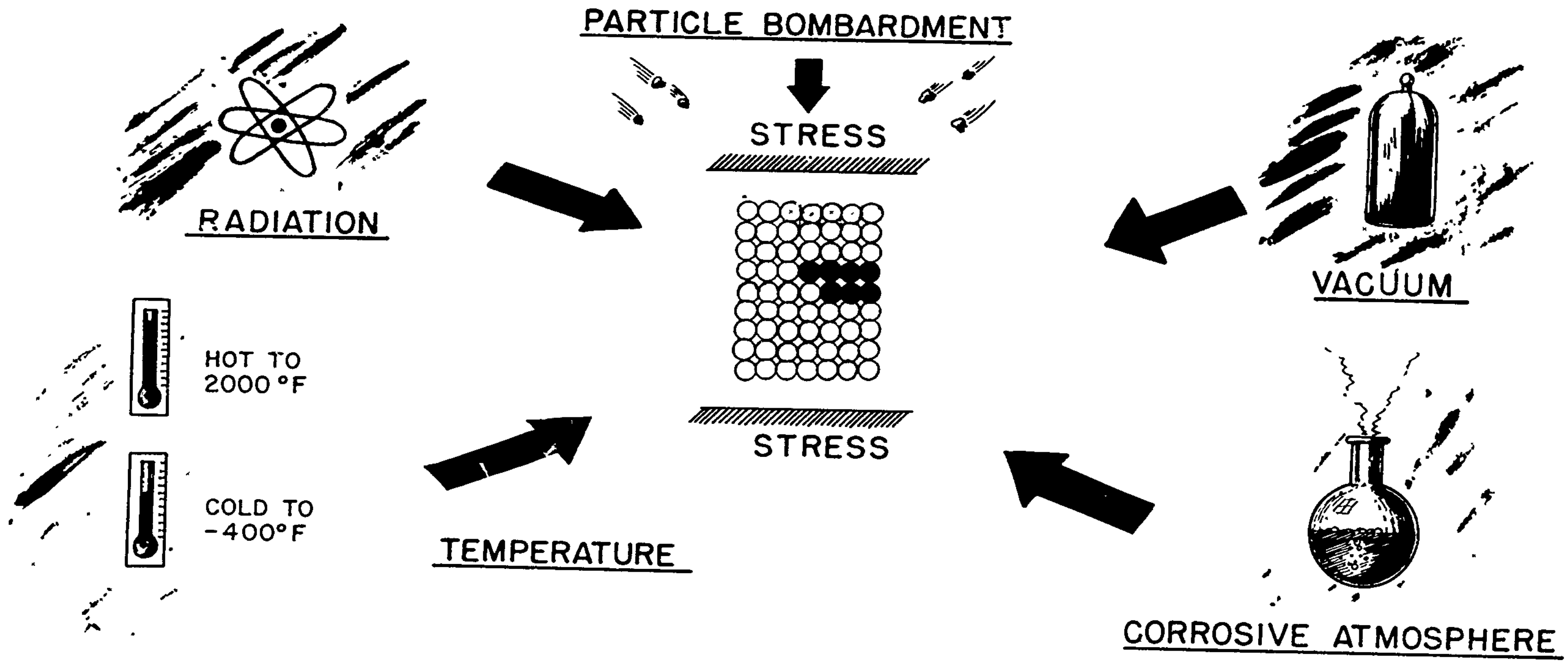


"SEVERAL THEORIES HAVE BEEN ADVANCED WHICH EXPLAIN MANY OF THE OBSERVED FEATURES OF CRACK NUCLEATION, BUT AS YET THERE IS NO COMPLETELY SATISFACTORY EXPLANATION OF THIS MECHANISM"

FIGURE 36

FUNDAMENTAL QUESTION:

WHAT IS THE RELATIONSHIP OF CRACK NUCLEATION TO THE APPLIED LOADING AND ENVIRONMENT ?

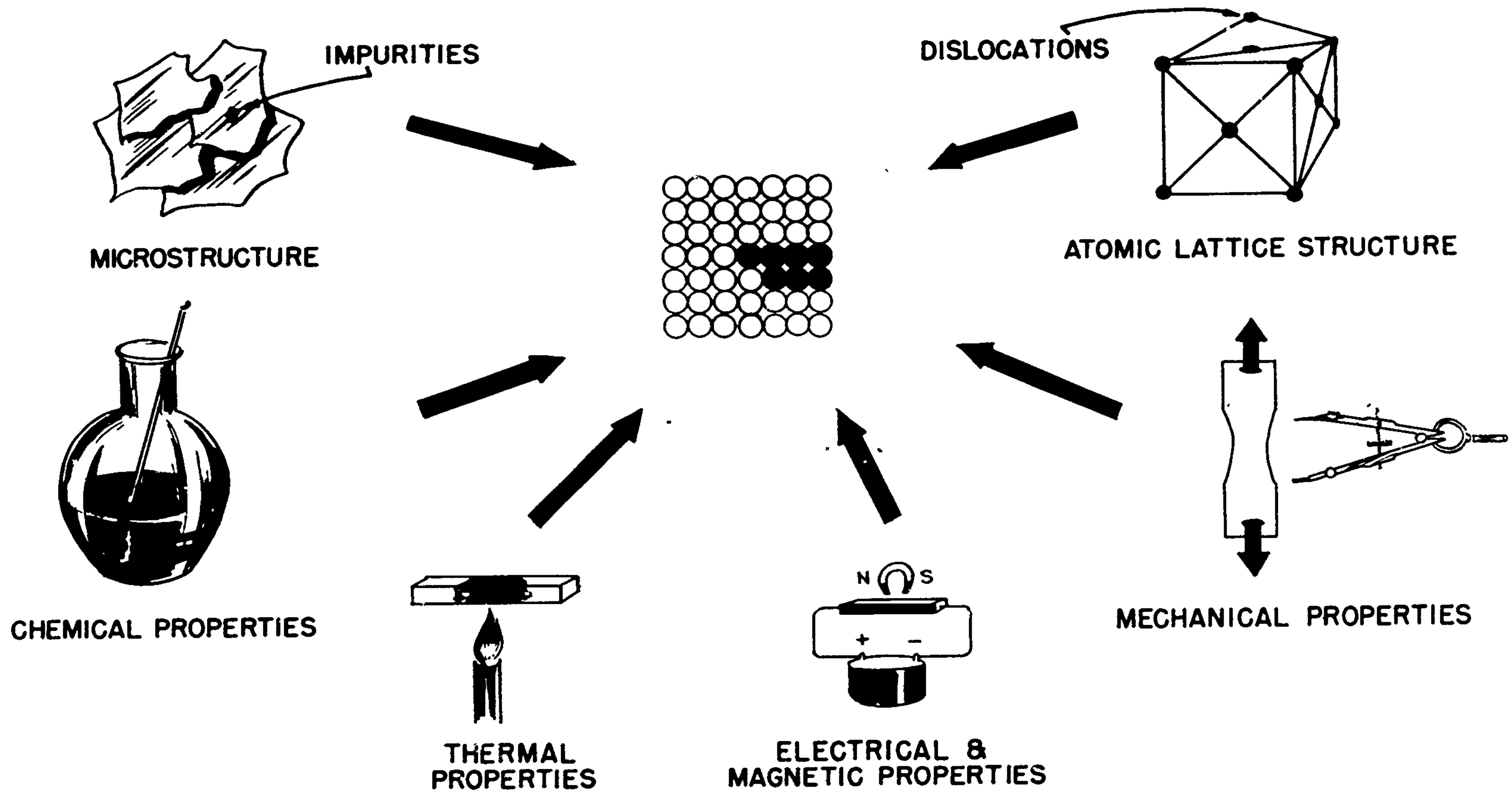


"MANY OF THE ABOVE EFFECTS HAVE BEEN STUDIED AND SOME EMPIRICAL RELATIONSHIPS ARE AVAILABLE, BUT THERE IS NO UNIFIED UNDERSTANDING OF THE FUNDAMENTAL PRINCIPLES INVOLVED."

FIGURE 37

FUNDAMENTAL QUESTION:

HOW DO MATERIAL PROPERTIES AFFECT THE MECHANISM OF CRACK NUCLEATION?



"EMPIRICAL RELATIONSHIPS HAVE BEEN DEVISED BY ENGINEERING AD-HOC TESTING, BUT NO UNIFIED UNDERSTANDING IS AVAILABLE."

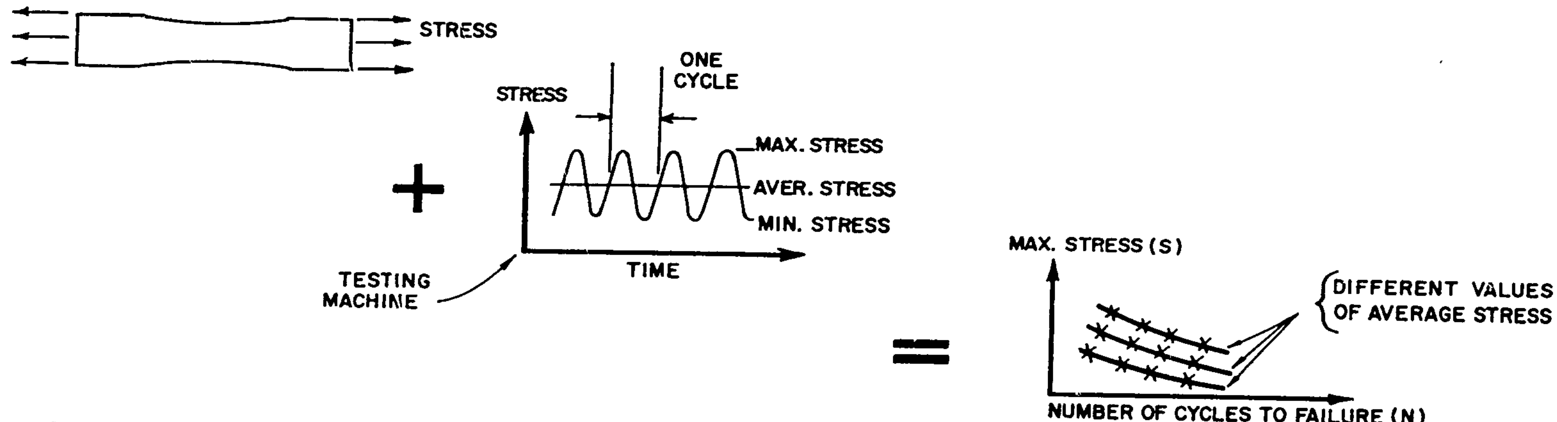
FIGURE 38

A BASIC PROBLEM:

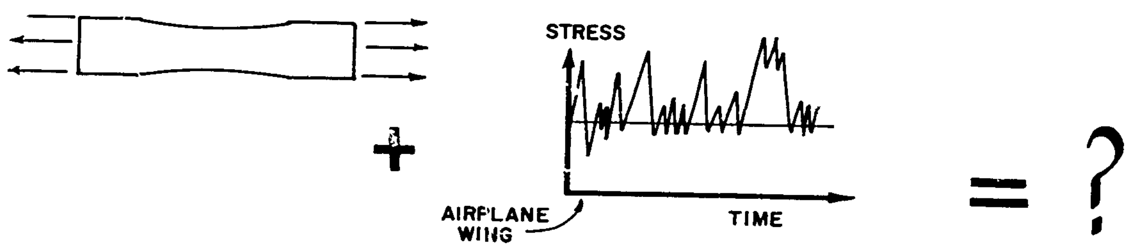
WHAT IS THE CUMULATIVE EFFECTS OF RANDOM LOADING?

FOR EXAMPLE:

KNOWING THE RESULTS OF THIS PROBLEM



WHAT IS THE ANSWER TO THIS PROBLEM?



"THIS IS THE CUMULATIVE DAMAGE PROBLEM FOR LOADING---IT IS COMPLICATED BY THE ADDITION OF MANY OTHER PARAMETERS IN THE TOTAL FATIGUE PROBLEM."

MAINTENANCE AND LOGISTIC ASPECTS OF STRUCTURAL FATIGUE PROBLEMS

By

MAJOR GENERAL T. P. GERRITY, COMMANDER
OKLAHOMA CITY AIR MATERIEL AREA

As I was listening to this story unfold this morning, it occurred to me that, as in most symposiums or meetings of this type, we tend to beat the problem to death before we get around to talking about the solution. So, as I listened to each speaker unfold his story, I gradually threw pages of my speech away. As you heard other speakers say, the Oklahoma City Air Materiel Area, or OCAMA for short, is involved in the B-47 program. We're involved very deeply, not only in the B-47 program but in the B-52 and KC-135 programs as well. Since we were so deeply involved in the B-47 program which kicked off the large effort that we are discussing today, I hope you'll forgive me if I beat this subject to death just a little bit more.

After the learned discussions you have heard here, I am sure that I can't add to the technical studies that have been made. I can, however, give you some insight from the service support viewpoint. First, let me explain our mission very simply. OCAMA, as a sub-command of Air Materiel Command, provides logistic support for the weapons for which we are responsible. It's an over-simplication to say our mission, basically, is to support SAC, but this will serve well for this discussion. Actually, we're supposed to get the supplies to the customer when they're needed. We're supposed to provide technical data as part of the Air Force team working with the contractor and ARDC on the maintenance and servicing of the products, both in the field and in the depot. We are also supposed to contract for that repair or modification in the depot or the contractor's plants. Occasionally we have the problem of providing area support. In these cases we send our people into the field to work on a technical basis, providing assistance to take care of problems that are beyond the manpower and know-how capabilities of the bases. This, very simply stated, is our mission.

Before I get into the story of the B-47, I would like to generalize for a moment. Do you know that approximately 90% of our logistic difficulties in supporting the customer involve engineering problems? Very few of our customer support headaches in the field stem from simple supply and maintenance problems. The basic difficulties are engineering deficiencies across the multitude of equipment that we support - not just aircraft - but their many thousands of components. At the base of practically any logistic support problem or supply shortage in the field, I can show you an engineering deficiency. Now why is this important to bring out in a session such as this? It is important simply because most of these engineering deficiencies involve fatigue or failure of components and of equipment. Many of these are metal or electronic components. These engineering deficiencies cause unpredictable failures which prevent the equipment in the field from achieving reliability. This creates terrific problems in the support area because we cannot provision, gentlemen, to support equipment that doesn't live out a reasonably predicted life between overhauls. Hence, the problem of vibration fatigue goes well beyond the B-47, or the B-52, or other major weapons that we are discussing. It goes right into the vitals of the equipment we use in these weapons and constitutes today's biggest AMC headache in obtaining the proper posture to support the customer.

Most of you in the audience who are contractors are very keenly aware of this point. But I am not so sure that all of the folks who are involved in this symposium are as aware of this as they might be. Therefore, I would like to ask you to bear in mind this basic problem as you labor to find solutions to these problems of structural fatigue and vibration failure. Here is an example of what we are facing in obtaining good equipment service life. For years we have been trying to determine a way of predicting the service life of turbine wheels for jet engines. We have not found a way yet except through destructive testing. When a jet engine comes back for overhaul, ordinarily, the wheels are condemned if there is any suspicion that they have been subject to over-temperature operation because we lack a method of non-destructive testing. Now this used to be a small cost -- about $1500.00 in a J-33 engine -- but in a J-79 we're talking about $15,000.00. These are important dollars and involve logistic support of the Air Force as well as funds for research and development leading to future weapons.

Now, the sonic fatigue problem which has been mentioned here several times this morning has been a source of difficulty ever since we got into high performance jet aircraft. We have had failure after failure of this type on the B-47 and the B-52. Fortunately, practically none of these resulted in a loss of aircraft, but they were catastrophic in the sense that we had to go into all-out modification problems to cure the source of the trouble. In most cases this has been a patchwork affair. We have not found, as yet, good cures for sonic vibration failures.

Towards the goal of doing a better job as a member of the team -- the Air Force-Industry Team responsible for the task of supporting combat aircraft in the field -- we have recently in the AMA's of AMC taken over what is known as in-service engineering responsibilities. Through building up the engineering capability to fix the weapons in the field today, we are developing better data to feed back to industry tomorrow. This, in turn, will help you in establishing future design criteria.

Before I discuss the B-47, I think I should say that the stratojet problems in my opinion, and in the opinion of the Air Force, were clearly not Boeing's responsibility. They were created by using an airplane far beyond the original design for which it was built. I'd like to turn this needle around a little bit (because it appears to have been a needle) and commend Boeing for having done an outstanding job, not only in providing a fix for the B-47 when these catastrophic failures occurred, but in carrying through a cyclic test program to develop criteria which is helping to pierce the frontiers of this very difficult fatigue problem.

Let me go, then, to the B-47 story. We know from what has been said here previously where the failures occurred on this aircraft. We also know that they occurred because the aircraft was used beyond its original design concepts. We in the AMA's of AMC found out, through these B-47 failures, that we could no longer rely on older methods of maintenance and inspection to find structural deterioriation of this type. Fatigue failures such as occurred were not evident, in the incipient stages, to our normal visual inspection systems. Furthermore, these could occur in portions of aircraft which were inaccessible visually. It became apparent to us that we had to develop a much more precise method of determining and fixing weak points in structures caused by fatigue, and to do this early in the service life of high performance aircraft, that is, in sufficient time to permit correction through modification without encountering catastrophic service failure and the bad effects that are created by these failures: loss of

life, loss of confidence in a weapon, and the terrific impact upon the logistics system throughout the nation. When we had to go into Project "Milk Bottle," which was the B-47 structural repair effort, overnight we involved some five agencies; three contractors, and two Air Materiel Areas. Headquarters USAF wanted an all out effort. We worked on this effort for a period of six months modifying some 1800 B-47 airplanes. Some of the first were fixed on an interim basis because we didn't know what the final fix was -- or what we hoped the final fix was -- but the Strategic Air Command's urgent need for these aircraft precluded a wait.

This was just a part of the impact. Of course, whenever you work all-out, seven days a week, twenty-four hours a day, you realize that you are somewhat wasteful of money and resources, yet we had no alternative in this case. Another impact which was less evident, but nevertheless real, was the curtailment of other projects of importance in order to put our whole energy and limited resources into the B-47 project. As a matter of fact, we have a problem in SAC today because we stopped repairing the KC-97 tankers during this period.

Now, there are a couple of things that we encountered in the "Milk Bottle" modification at all facilities that bear brief mention. First, we found out that our inspection processes were inadequate to the task of examining this high performance, complex jet airplane. We found that we needed to develop new tools such as borescopes of a special design to allow examination of the bolt holes of the highly stressed center section of the wing of the airplane. Once we got the scopes, we were not exactly sure of what we were looking for or how to find it, so we had to develop techniques to help us do that. Now, gentlemen, it's one thing to look at a "clean" structure and determine whether it is deteriorating down to the point of being unsound, but it is another thing to look at a structure that has been out in the field subject to a certain amount of dirt and corrosion. You cannot examine this sort of structure very clearly nor can you determine if there are cracks in it without first etching it with acid. However, we found out in some of our inspections that the etching with acid led to a question of whether we were doing good by making the inspection. It led to some doubt as to the quality of our job. To be very frank with you, gentlemen, although I think on the whole our job was done very well, we're not certain whether or not we have left a condition which, combined with moisture, may produce corrosion under stress conditions in the future. This makes it mandatory that we watch closely the continued operation of these airplanes and perform sample tear-downs as they age further.

Another thing we discovered during Project "Milk Bottle" was that we needed much better equipment than the old hand tools and drills to re-drill and ream holes in this highly stressed structural area. We needed better tools all around to do the job with repetitive quality. We've learned this in the production processes of some of our newer airplanes too. I think we have still more to learn because one of our most serious problems in some of our highly stressed and highly complex equipment today is the problem of obtaining repetitive quality.

Of course we want to avoid more project "Milk Bottles." Therefore, based on what we have learned, we have embarked on the program which Colonel Taylor described to you of extensive cyclic testing of B-47 aircraft, the cyclic testing of B-52's, and we intend also to go into the cyclic testing of the KC-135. This latter program will enable us to find out early in the service life of an airplane what its life expectancy might be. This can be done by timing the flight

load surveys with the cyclic test spectrum, correlating the two together, and thus predicting a life based on actually cycling through an estimated life period. Using this method on the B-47, as Colonel Taylor mentioned to you earlier, we solved the longeron problem before we had catastrophic failures in the field. This was of utmost importance. It didn't preclude an expensive retrofix, but it did avoid sudden failure and the loss of aircraft and life.

Looking ahead, what we need to know more clearly is the expected life of an airplane with inspection and repair, and its ultimate life without repair. We hope that through the efforts of this group and others, we will find some good answers to this question. We need to correlate our flight test and load spectrum programs with actual flight experience. We will do this through the VGH recorder program. Colonel Watkins mentioned that this program, once it got to OCAMA, was somewhat obscure. I don't believe there's anything obscure about it except that we still haven't been able to start. However, we intend to work closely with SAC and with ARDC to provide data reduction information which will be useful for design purposes, as well as useful for SAC operational purposes. We intend also to use the data, correlated with information based on the cyclic tests, to schedule the amount of IRAN needed.

Now, in addition to rescheduling and refining our aircraft inspection program, we have decided that we need complete structural tear-down of sample quantities of aircraft, in spite of the fact that we are learning much from these cyclic tests. Through these actions, I'm satisfied that on aircraft like the B-47, the B-52, and the KC-135, we will be able to counter most of our structural problems before they become catastrophic. I don't think we will be able to avoid emergency type repairs to prevent structural deficiencies from becoming catastrophic in the future, but I think we'll be able to catch them in time. But it is a very great disappointment to me that our lack of know-how is so great that the only way I can assure myself, or my people can assure me at OCAMA, that the B-47 or the B-52 continues to be sound structurally is by a complete tear-down examination. I believe that this lack of information should spur us on to try to learn more about the fundamental characteristics of metals and materials, vibration, and stress corrosion so that we can, with much more accuracy, predict the structural life of equipment or weapons without encountering catastrophic failures. I think this is very important not only for major weapons like the B-52 and the KC-135, but also for the less dramatic areas of aircraft equipment and components. Because we have no sound, predictable life for these, we are faced today with serious problems of support.

If we can solve some of these problems, then indeed we will have reliable equipment in the future. Reliability is an absolute essential to the success of hypersonic aircraft, missiles, and the spacecraft of the future. We talked about the money we spent on the KC-135, the B-52 and the B-47 and what we expect to spend on the SAC fleet in the next five years or so. I think this is small potatoes when you consider what we might have to spend in the future if we don't find some fundamental answers to the problem of fatigue and stress corrosion. From my point of view, the Air Force, the aircraft industry, and the research industries which have made such a great contribution to the advance of aerospace science have failed miserably in this one area. I think we have contributed to that failure by failing to foresee the need for a research and development money to enable industry to solve this problem in the immediate future. In addition, I urge the colleges and universities in their research not to wait for us to finance all of it but to get after the problem themselves.

PROBLEMS AND PROGRESS IN MATERIALS RESEARCH AND DEVELOPMENT

By

Richard R. Kennedy

Materials Laboratory

Wright Air Development Center

Fatigue is defined in the 1948 edition of the Metals Handbook as the tendency for a metal to break under conditions of repeated cyclic stressing considerably below the ultimate tensile strength. That is a clear and simple definition, but the details of the "conditions" leading to fatigue failure are difficult to describe or determine. It has long been recognized that conditions leading to the fatigue failure of a structure are often extremely complex and include many variables. Therefore we must differentiate between the study of structural fatigue failure and the study of the phenomenon of fatigue where variables can be isolated and controlled.

One of the early investigators in the field of fatigue was Woehler, the German scientist. He worked on the subject around 1840, and discovered many things which are useful today. He devised a fatigue machine so good that it was used well into the present century. I will not attempt, however, to trace the long history of fatigue down to more recent times.

When the airplane was invented, it soon became evident that, due to its very nature, all parts were highly stressed and were subjected to cyclic loading. It was also recognized that failure of an important part was disastrous, so fatigue was a subject of primary importance. For more than forty years, the Materials Laboratory has had an active program on both the phenomenon of fatigue and structural fatigue failure.

R. R. Moore was a member of the Materials Laboratory staff when he developed the rotating beam fatigue machine which bears his name. It has been the most widely used fatigue machine in this country.

The work of Johnson and Oberg, carried out over a period of many years, won national recognition and contributed much to our knowledge of fatigue.

One of the older problem areas is that of the endurance limit. The endurance limit is defined by the Handbook as the maximum stress that a metal will withstand without failure during a large number of cycles of stress. On a S-N curve, it is the point where the curve approaches the horizontal, so it would be expected that no very great change in load carrying ability would take place as the number of cycles is extended.

It is generally considered that non-ferrous metals do not have a true endurance limit, so, in fatigue testing, they are subjected to, say, 100 million or 500 million cycles, and the endurance limit is reported as the stress which that particular alloy will withstand at that number of cycles. It is assumed that the endurance will continue to drop slowly as the number of cycles increases. For the strong aluminum alloys, the endurance limit at 500 million cycles is frequently reported. To determine the effect of much longer periods of cycling, it was demonstrated,

in a 9 billion cycles test program, that there is no significant change in the endurance limit beyond 500 million cycles, indicating that current practice gives a safe figure for such alloys.

It has been postulated that steel has a true endurance limit of about 50% of its ultimate tensile strength. The endurance limit at 10 million cycles is considered to be adequate for any number of cycles. This has been supported by many years of service experience with steels heat-treated to 180,000 psi or less.

When the ultra-high-strength steels were introduced, it was discovered that, at these high strength levels, the endurance limit was much lower than 50% of the ultimate tensile strength. A long and careful study, conducted on both internal and sponsored bases, showed that the endurance limit in ultra-high-strength steels is influenced by the number, size and distribution of inclusions. Steels have been produced during the course of the investigation, usually by vacuum melting and using sophisticated deoxidation methods, in which the endurance limit is 50% or more of the tensile strength at strength levels as high as 280,000 psi. These steels are substantially free from large inclusions, and the inclusions present are below a certain critical size.

Work was conducted by the Materials Laboratory to determine whether the ten million cycle endurance limit is safe for these pure, ultra-high-strength steels. In this investigation, specimens failed after as many as 485 million cycles, demonstrating that the 10 million cycle endurance limit is not reliable if a part is to be subjected to a very large number of cycles.

One of the variables which can be controlled in a fatigue test is surface finish; so fatigue specimens are finished with great precision. Aircraft parts cannot be finished to the same degree of perfection. It is known that notches and other surface irregularities have an important influence on the fatigue life of a part. This makes it difficult to apply the results of laboratory tests to the design of parts and components. A study of the role of notches in fatigue has been conducted over a period of years. It has been shown that the geometry and sharpness of a notch have a profound effect on specimen life. We now have a much clearer understanding of notch behavior to aid designers in arriving at safe configurations for aircraft and missile parts.

Inspection of a typical S-N curve will show that a metal will withstand a relatively high stress, above the endurance limit, for a limited number of cycles without failure. The behavior of materials, under repeated application of loads of varying amplitude, is an important problem in materials application to aircraft and missiles. Most structures and components are subjected to a random fluctuation of loads, whereas conventional fatigue tests are conducted under constant load. In order to take the effect of such load spectra into consideration, a linear cumulative damage theory has been widely used for years which does not present a sufficiently close approximation of the real, complex process of fatigue damage under varying stress amplitudes.

Theoretical and experimental efforts to improve the cumulative damage rule have led to the development of a semi-empirical, non-linear cumulative damage rule. The validity of this new rule has been proved by random spectrum loading experiments, based on actual flight records, for several structural materials, demonstrating a considerable improvement, which will allow more efficient design in respect to fatigue considerations.

The final goal of this work is to design a constant stress or simple spectrum fatigue experiment which will, with sufficient accuracy, generate representative cumulative damage initiated by a complex exponential or other type spectrum encountered in actual service.

Efforts are made, also, to make possible the translation of cumulative damage fatigue information, gathered on materials specimens, into design criteria of structural components, for the purpose of simplifying and improving design procedures.

In the course of this work on cumulative damage, equipment has been developed which allows the simulation of actual service load spectra with random distribution, including ground-air-ground cycle and single overloads on axial loading materials specimens.

A cumulative damage theory has been formulated, by Prof. Freudenthal of Columbia University, in which heat flashes and associated thermal gradients, resulting from the conversion into heat of the work in slip of the resolved shear stresses, are found to be of sufficient severity to account for the initiation of microcracks, as found in fatigue. This theory has contributed greatly to the understanding of fatigue mechanisms.

It is well known that there is often a wide variation in the number of cycles to failure of apparently identical specimens subjected to the same stress. Various statistical methods for evaluation of fatigue data are in existence, but a relatively large number of samples is always necessary. In order to satisfy a great need, especially in the fatigue evaluation of structures, a method has been developed which allows the statistical evaluation of fatigue data from small test series. This method which is based on regression theory, yields a reliable statistical evaluation in cases where small populations are distributed over a wide range of the stress-cycle diagram and where only a few samples are, or even one sample is, available at the individual stress levels. The method has been proved sufficiently reliable by experimenting with a large number of different test series. It appears particularly valuable for fatigue tests on flight vehicle components which are not easily available in sufficient numbers to conduct a conventional large sample statistical evaluation.

The effect of finishing processes has been studied. Repeated tensile loads are necessary to produce fatigue failure. Any finishing or other manufacturing process, which leaves the surface in tension, tends to lower the fatigue strength. Conversely, a process which leaves the surface in compression raises the fatigue strength. Shot peening appears to be especially effective in this respect.

Under a basic research program with the Midwest Research Institute, the micro-physical changes under cyclic loading are investigated by small angle X-ray scattering, electronmicroscope and other microphysical techniques for the purpose of determining mechanism of fatigue. The mechanism of crack initiation and propagation, atmospheric effects, subgrain formation and polygonization, and dislocation movement are studied. Valuable information has been found, and it is anticipated that this work will contribute greatly to an improvement of fatigue properties in bulk material and structures in the future.

Other efforts in the Air Force, on the fundamentals of fatigue, are being expended by the Aeronautical Research Laboratory, WADC, and the Office of Scientific Research. The Aeronautical Research Laboratory is conducting research

programs concerned with the atomic mechanism of the fatigue limit, the mechanisms of fatigue crack propagation and the effects of neutron radiation on the fatigue behavior of metals. The latter work is being conducted in a cooperative program with the Materials Laboratory. The Office of Scientific Research is sponsoring work on the effects of vibration on the movement of atomic lattice imperfections and on early detection of fatigue microcracks.

Formerly, materials, which were cyclically loaded in service, operated in a fairly narrow temperature range. This range, in which fatigue information is needed, has been rapidly expanding and now extends from the very low temperature of certain liquified gases to the highest temperatures a material will withstand. Other environmental conditions, such as vacuum, gases other than air and corrosive conditions, are receiving attention.

The work of Dr. Lazan, of the University of Minnesota, in the field of high temperature fatigue is noteworthy. He demonstrated that, under axial loading, a metal may fail either in creep or fatigue according to the temperature and the magnitude of the tensile pre-load component of the cyclic stress. Work, at the Materials Laboratory, on superalloys at engine operating temperatures, is being conducted. Increasing service life of engines has made it important to determine any change which may take place in endurance limit with an increasing number of cycles.

One of the newer and more perplexing problems is that of sonic fatigue; failure of a part due to cyclic stresses generated by high intensity sound. The Materials Laboratory has realized the importance of this problem and has initiated an active and comprehensive program.

Work has been sponsored at the University of Minnesota for many years. Certain phases of this work are subcontracted in order to insure that the best qualified people in each area are working on our problems. Theories have been developed for predicting large deflection flutter behavior of structural panels. The theory is being extended to include structural damping, as well as, high tension stresses, which are responsible for the nonlinear behavior of panels.

Theories have been developed for the application of interface damping to structural panels. Designs have been conceived, on the basis of these theories, which introduce practical viscoelastic interface damping as a means of sound energy dissipation. Panels have been constructed, and are under investigation, which exhibit a very high energy dissipation capability.

A theory has been developed, for the energy dissipation caused by slip of rounded contacts, for the purpose of introducing dynamic damping devices into honeycomb panels.

Response, damping and correlation studies of structural panels and other configurations are being made which are essential for the solution of the acoustic fatigue problem.

A mathematical analysis of cumulative damage, under various acoustic loading conditions, is being made by the University of Southampton, and the theory is being collated with the latest test results obtained by the RAE Farnborough. Fractographic examination of crack propagation is being carried out to interpret the program load sequence in terms of deformation bands and laminations.

Damping, stiffness and fatigue investigations are being conducted on riveted joints with viscoelastic interfaces under harmonic and random loading. Studies of panel vibrations and response are being carried out jointly with the British Ministry of Supply and RAE Farnborough. Here the auto-correlation technique is being used to give an effective narrow band frequency analysis of the strain in panels subjected to jet noise. The panels are part of a representative section of an aircraft structure.

Cyclic thermal stresses and, therewith, thermal fatigue play an important role in flight vehicle components and propulsion systems. The difficulty of assessing actual stresses, and predicting thermal fatigue damage, lies in the complex elastic-plastic action taking place. Relatively little attention has been paid in the past to this low cycle fatigue phenomenon merely due to the lack of a reliable stress analysis method. Work is being performed by Neuber, for the Materials Laboratory, to develop a thermal stress analysis for the determination of a thermal fatigue criterion.

An important and unending task is the determination of the behavior of the vast number of materials in cyclic stressing under most, or all, of the conditions I have mentioned. Reliable design data are essential, in order to use materials efficiently.

As an aid in data collection, the Parsons Corporation has developed a system for coding materials property information on IBM cards for subsequent automatic processing of the data. The card system will permit coding not only the property information (stress vs. lifetime for fatigue information, stress and time-deformation data for creep, etc.) but also pertinent information describing: (1) the basic material and fabrication; (2) short time mechanical properties of the material; (3) specimen design and methods of preparation; and (4) type of test and associated testing procedure.

The data coding system has been completely developed for fatigue information. The system is useful in coding information from basic fatigue studies of materials, as well as, component and structural fatigue studies. Approximately 30,000 fatigue data points have been entered on cards up to the present time. A data coding system for creep data is being developed currently.

In order to make the results of any work program available to the scientific community, it is necessary to communicate with others. During the calendar year 1958, some 19 technical reports, covering various phases of the fatigue problem, were printed and distributed.

An International Conference on Small Angle X-Ray Scattering, sponsored by the National Science Foundation, the Air Force Office of Scientific Research, and the Materials Laboratory, WADC, was held in September 1958. International authorities were gathered to discuss the use of X-ray techniques for submicroscopic research on dislocation movement, subgrain formation and similar micro physical phenomena. The proceedings of this conference were published in the May 1959 issue of the Journal of Applied Physics.

An International Air Force Conference on Acoustic Fatigue, sponsored by the Materials Laboratory, WADC, at the University of Minnesota, will be held in October 1959. The principal purpose of the conference is to assemble scientists in the fields of acoustics, applied mechanics and field engineering, and discuss

problems of common interest as related to the acoustic fatigue problem. Papers will be presented by authors from the United States, and other NATO nations, and panel discussions will be conducted.

I have thus far discussed work on the phenomenon of fatigue. The Materials Laboratory has also been very active in the field of structural fatigue failure. Over the years we have examined thousands of parts and structures which have failed in service. The majority of these failures have been due to fatigue. Years of work in this field has enabled the Laboratory to contribute materially to the solution of many difficult problems.

I have tried to show that fatigue is a big problem. We have learned, and continue to learn a lot about it, but much work remains to be done in many areas. As the understanding of the phenomenon of fatigue increases, designers must utilize the latest advances, so that parts and components can be constructed which will withstand the highest possible stresses with the required safety.

In conclusion the most important areas in which more work is needed are believed to be those listed below.

1. Continuation, to a logical conclusion, of work now in progress.

2. Translation of basic and state-of-the-art results into terms more useful to designers.

3. Continued development of design criteria for new operating conditions.

4. Expansion of non-linear cumulative damage rules to apply, first, to simple components and then to complicated structural configurations.

5. Development of analytical methods for determining thermal-mechanical interactions.

The assistance of Mr. W. J. Trapp and Mr. D. M. Forney of the Materials Laboratory in preparing this paper is gratefully acknowledged.

AN APPROACH TO IMPROVEMENT OF THE FATIGUE SITUATION

By

Richard V. Rhode

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INTRODUCTION

For a long time, now, the literature on fatigue has grown by some 300 papers per year. Most of these papers have been thoroughly discussed. Literally millions of words have, therefore, been written and spoken on the subject in recent years. Now here we are again. Why are we here and what do we hope to accomplish?

The answer to this question is, of course, obvious. Despite the millions of words that have been written and spoken, the fatigue problem still remains a most important unsolved problem in aircraft structures. We are here to try again to find a solution to it. But what is the problem? There are actually several of them and they may, I think, be put this way in order of urgency:

1. How to live with today's airplanes that were built on yesterday's state of the art;

2. How to build tomorrow's aircraft and other flight vehicles on today's state of the art; and

3. How to improve the state of the art.

As a research administrator, I suppose that I should concern myself here with the business of improving the state of the art. I shall not, however, do so. The relative urgency of the first two parts of the fatigue problem lead me to step somewhat out of character and to assume the role of observer and commentator on matters which do not come strictly within my own professional and organizational area of responsibility. I shall, therefore, concern myself with the application of the state of the art. In particular, I want to talk about what I think has been and still is one of the weakest points in the fatigue situation. That weak point is the lack of adequate control of design stress, or, more specifically, the lack of adequate requirements for selection of design stress in the light of known fatigue phenomena.

In talking about this matter, I am under no illusions about the difficulty of the problem. It is a difficult one. Whether it can be solved will remain to be established by others. Nevertheless, I shall make so bold as to suggest a course of action.

Let us first, however, consider briefly some of the past approaches to control of the fatigue problem and some germane aspects of the present situation.

PAST APPROACHES

For many years, airplanes were designed solely for static strength, with generally satisfactory results. The most elementary knowledge of fatigue and obedience to a few simple qualitative rules relating to detail design served adequately to prevent fatigue difficulties. In short, with the earlier airplanes there really was no fatigue problem, and if a fatigue failure occurred, it was a sign of someone's gross ignorance or carelessness.

As time went on, fatigue failures of airplane structures cropped up with increasing frequency--even when the old rules were obeyed. Gradually, a general awareness of the existence of a fatigue problem developed.

Stimulated by large quantities of research results on notch fatigue in machine design and by advances in the theory of stress concentration, concerted efforts were started to establish a basis for the prediction of fatigue life of airplane parts and structures. There were some who were bold enough to say, "Give us the repeated loads and we will give you the fatigue life." In the United States and in other countries, over a period of many years, massive quantities of data on repeated loads were collected, materials and structures laboratories tested thousands of specimens, theories of fatigue were pursued, and schemes for predicting fatigue life were devised. The concept of prediction of fatigue life reached a noteworthy point in the writing of specific fatigue life requirements in England. Under these requirements, a crack-free or so-called "safe life" is established for a given specific structure or structural component by a few laboratory tests, and the structure is retired from service when the service life becomes equal to the safe-life.

The considerable experience with fatigue that many workers eventually acquired caused some to have grave doubts about the validity of these safe-life predictions. These doubts were based in part on difficulties in forecasting repeated load spectra, on large scatter in fatigue test data, and on differences of environment between the structural laboratory and the service arena. It was also recognized that not all fatigue cracks were equally lethal. Some were highly malignant--small and virtually undetectable cracks resulting in large loss in static strength. Others were more benign, showing themselves plainly before serious loss in strength occurred.

Recognition of these facts and difficulties led to application of the principle of fail-safe design. Under this principle, the designer recognizes that fatigue cracks and other damage are practically unavoidable, but he minimizes the danger of catastrophic failure by providing alternate load paths and choosing materials and configurations that result in a minimum rate of crack propagation and in maximum inspectability.

The oldest, purely qualitative approach of using only a few simple rules governing the detail design is adequate if the design stresses are very low. However, in the quest for improved structural efficiency, engineers have pushed the design stresses to their physical limits. Thus, the simple qualitative approach has been outmoded by progress in design for static strength.

The safe-life approach is fully rational in principle. In practice, however, its efficacy is very severely limited. There are too many facets of the problem on which our present knowledge is too scanty to obtain reliable results for the complete structure by a combination of component testing and analysis. On the other hand, to obtain reliable results by full-scale tests on the complete structure would be economically prohibitive because of the large number of specimens required to obtain a modicum of statistical confidence.

Fail-safe design, if adequately executed, gives a high degree of assurance against catastrophic failure. But, by itself, it gives no guarantee against costly maintenance. Therefore, even in a fail-safe design, adequate attention must be given to the question of fatigue life.

THE PRESENT SITUATION

Let us now take a brief look at the present situation and how it came about.

Figure 1, taken from a paper I gave in 1953, shows how the design stress for aluminum alloy structures of transport airplanes increased over a period of about 12 years. In this period, the design stress increased from 20,000 psi to 60,000 psi, that is, by a factor of three. The increase was due largely to improved knowledge of static design, except that a part of the last jump was due to the introduction of higher-strength alloys. Although the figure does not show it, this curve has not continued to rise in the ensuing years, but has reached a plateau as a consequence of recognition of the fatigue problem.

Figure 2 shows the estimated reduction in fatigue life for transport type airplanes that results from the increase in design stress shown in the previous figure. As can be seen, the fatigue life was reduced by several orders of magnitude, or by a factor of about 10 thousand, resulting in fatigue lives that were inadequate, in some cases grossly inadequate. To put it bluntly, the state of the art had been pushed too hard. Serious trouble was experienced first on transports, then a few years later on fighters By now, it has become an open secret that all types of airplane (but not necessarily all airplanes) may have serious trouble with fatigue, and that fatigue must be a dominant consideration in all stages of design for all types of aircraft.

It is true, of course, that some types of airplane flying now--and experiencing troubles--were designed at a time when relatively little attention was given to fatigue in the design stage. However, even very late airplanes which have had the benefit of very careful attention to fatigue in the design stage as well as the benefit of extensive development testing appear to have a disconcerting amount of

TREND IN DESIGN STRESS

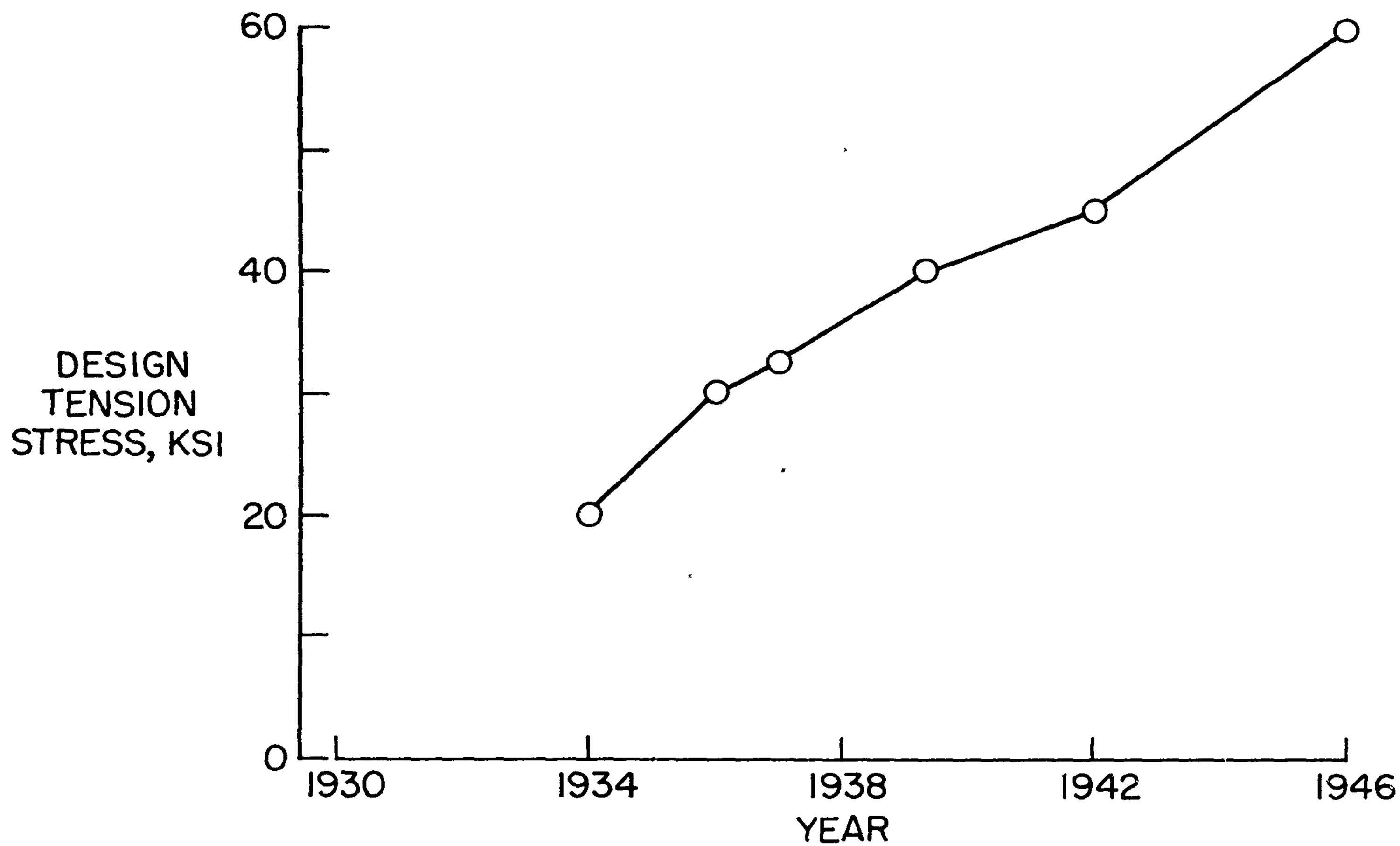

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FIGURE 1

TREND IN FATIGUE LIFE OF PRIMARY TENSION MATERIAL

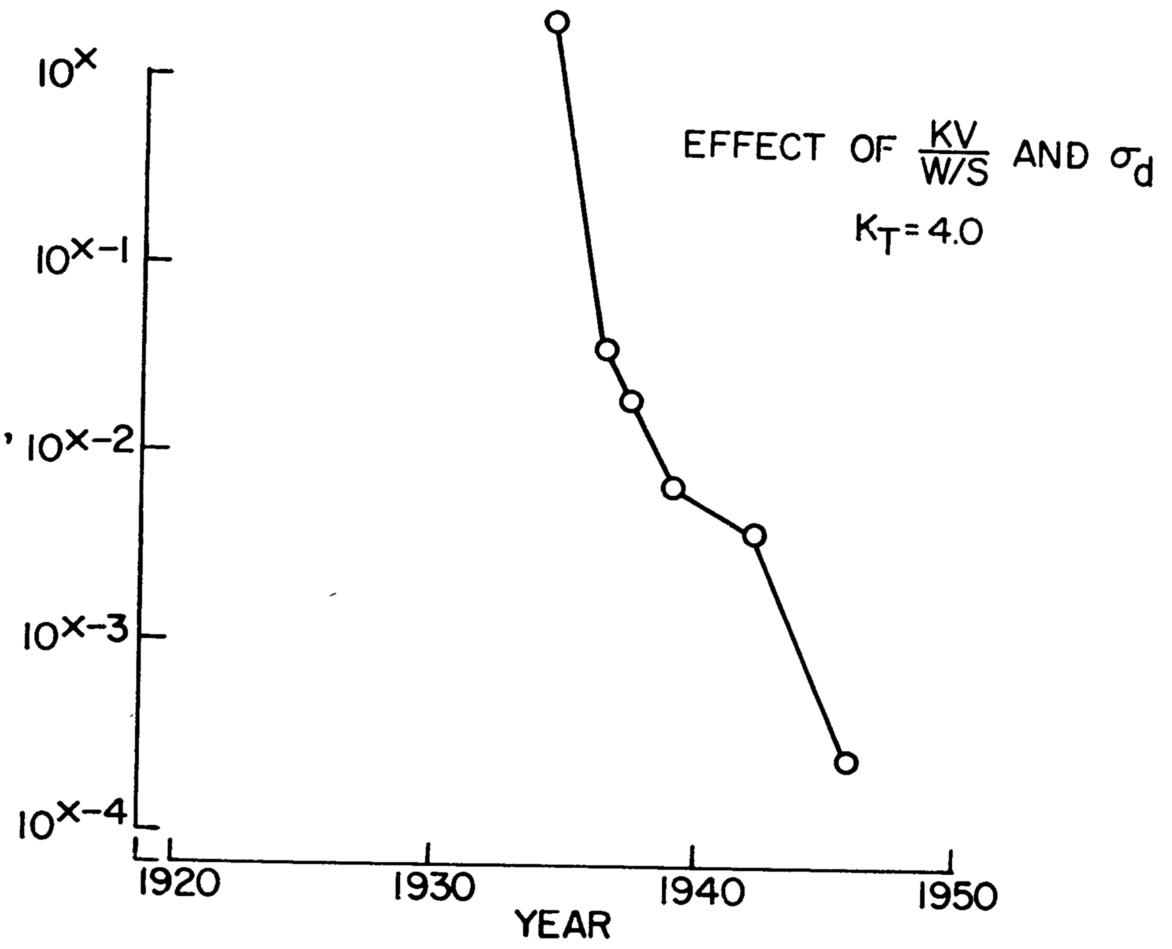

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FIGURE 2

fatigue troubles, often beginning soon after the airplane is put into service. It appears that the state of the art is still being pushed too hard.

Who is doing this pushing? Who owns the state of the art? The answer is that all of us are doing some pushing, all of us own the state of the art. The research scientist or engineer becomes enthusiastic about the importance of his findings and may seek, for example, to see them employed in a safe-life analysis The operator of aircraft, whether military or civilian, continually presses for higher performance and efficiency. He often also seeks to use the airplane beyond the conditions for which it was designed. The builder of aircraft in our free enterprise system shaves weight to the utmost in an effort to win the contract. All of this pushing is designed, of course, to achieve new pinnacles of success, but it can easily result in toppling over the brink of disaster.

Obviously, all of us--research scientist, operator, designer, whether in industry or in government, have a grave joint responsibility to minimize this risk of toppling over the brink.

RESEARCH VERSUS SPECIFICATION

In the field we are considering here, government plays two roles--it conducts research and it promulgates design and safety requirements.

Research in fatigue is being conducted by the government on an extensive scale in its own laboratories, and through contract, in others. This research must be continued in order to improve the efficiency of design and to increase our ability to control fatigue. It is important to recognize, however, that the results of research do not, in general, guarantee freedom from fatigue difficulties. To the extent that fatigue strength can be improved by research, advantage may be taken of these gains to justify higher allowable stress, in which case there is no increase in fatigue resistance or life, but only the ever desirable reduction in structural weight. The only conceivable product of research that will solve the fatigue problem once and for all is knowledge of how to design an otherwise suitable structural material with an endurance limit approaching the static strength. There is no basis, at present, for believing that such a result is possible. Nevertheless, research in appropriate branches of physics of solids must be pursued, as herein lies the only hope of a truly satisfactory solution of the fatigue problem.

The second role of government is to promulgate airworthiness requirements, including specifications relating to structural fatigue. The establishment of such specifications has been and continues to be a most difficult problem. Early attempts to do so resulted in vague stipulations that had little meaning. For example: "The effects of repeated loads upon the strength of the material shall be considered in selecting allowable strengths for design." Or again: "The design shall be such as to minimize the probability of disastrous fatigue failure." Such specifications probably accomplished little that would have been accomplished without them, although I say this in no spirit of criticism of their authors who were simply stuck with a state of the art that was characterized by little or no theory and

large masses of undigested data, much of which seemingly contradicted the rest. These early specifications did have the merit of requiring some review of the design for fatigue. A major difficulty here, however, resides in the fact that very few people have the broad familiarity with the state of the art required to assess designs adequately.

The proposed new Military Specifications represent a determined and very commendable effort to establish meaningful basic rules for fatigue design. They specify the service life required of each type of airplane, and they specify in fair detail the load history that each type of aircraft shall be capable of withstanding. Thus, in contrast with the generalities of previous specifications, the proposed specifications contain numbers which impart a definite meaning to them.

However, the proposed specifications still fall short of what is done in the specifications for static strength design. For the static strength requirements, the allowable design stresses are laid down in such documents as the ANC-5 volume. The allowable stresses given in this document are based on comprehensive tests, checked and cross-checked, reviewed in detail by the entire industry as well as by the procuring agencies, and finally become a uniform code binding on all. By contrast, the fatigue requirements leave the door wide open for establishing allowable stresses in specific cases by a small number of tests, without requirement of cross-checks and reviews.

Demonstration of the weakness of this procedure would require that we involve ourselves in lengthy, complex and abstruse considerations of probability theory as well as in masses of test data. I shall content myself, therefore, with two relatively simple statements followed by a brief discussion of them.

1. Neither a safe life nor a safe design stress can be adequately determined from the results of a few laboratory tests of structural components, including tests of complete structures, even if the laboratory environment is identical with the service environment.

2. Neither a safe life, nor a safe design stress can be adequately determined from a statistically satisfactory large number of structural fatigue tests, if the tests are made under ordinary, practicable laboratory test conditions.

These sound like strong statements but they are true. The truth of the first one follows from the facts that fatigue test data are statistical in nature (i. e., they always display large scatter with indefinite boundaries) and that statistical methods are incapable of defining probability curves with reasonable certainty when only a few test data are available. The author of a British paper on fatigue, having examined this question in some detail, commented wryly, "This is very discouraging."

The truth of the second statement follows from a great many considerations only some of which are listed here:

1. The service loads are imperfectly known.

2. The stress distribution in the structure is imperfectly known.

3. The test loading conditions are simplified approximations of the assumed actual loading conditions.

4. The test specimens usually represent the actual structure only nominally as to material and geometry and do not represent service or production-line structures adequately; that is, imperfections in machining, workmanship and other processing effects, some of which occur in service, often alter the material properties or structural geometry in small, but critical respects.

5. The usual, practical, accelerated laboratory fatigue test does not account for time-dependent effects, such as corrosion, found in service.

In view of all of these matters, it is felt that something better than reliance on a few laboratory tests is needed to control the choice of design stress.

AN APPROACH TO CONTROL OF DESIGN STRESS

Aircraft engineers have always relied heavily on past experience. If this experience is sufficient and germane to a new design, it can be used as a basis or starting point for the new design. For example, the design stress used on an aircraft with ample service experience might be established. Comparative fatigue life calculations might then be made for the old aircraft and the new one to account

for changes in certain design and operating parameters such as external configuration, weight, speed and flight altitude, and these comparative calculations then be used to establish the design stress for the new case under consideration.

The approach has some rather major and obvious limitations: the material used in both designs would have to be the same; manufacturing processes would have to be essentially identical; the load experiences and structural geometry would have to be reasonably similar, et cetera. Nevertheless, if such essential conditions are met, the use of a suitable datum stress based on experience coupled with analysis of the effects of the changed parameters is about as sound a procedure as can be. New designs, however, often involve such pronounced changes in material, configuration and environment, that previous experience on a single machine or on a series of similar machines, cannot be considered as applicable. Consequently, more basic procedures must be used and means must be found for employing a broader range of experience including available and applicable laboratory test data.

One way in which this might be done will now be outlined. There is no intention here of recommending a specific procedure, but only to illustrate in a general way how the problem might be approached.

The object is to establish the design stress for a certain region or component of the structure, say a transverse splice in a wing cover. Specimens are manufactured which represent the region in question. A load schedule is prepared which gives, in accordance with the proposed Military Specifications, the loads for a 100-hour block. Two arbitrary stress levels are chosen, one higher than the expected design stress, one lower, and the corresponding test loading schedules are prepared, as shown in figure 3. The abscissa is the number of individual load cycles in each "step". A repeated-block fatigue test is run on one specimen at each of the two stress levels chosen, resulting in failure after N_B repetitions of the load block (or 100-hr. unit).

The two values of N_B are now plotted on a diagram of design stress vs N_B (Figure 4), and a line 1-2 is drawn between the two points. From this line, a first rough approximation (S_1) to the design stress can now be read for the required life N_{Breq}.

At this first-approximation design stress level, several more specimens are tested. The lives of these specimens are plotted on the diagram of Figure 4 and a line parallel to the line 1-2 is drawn through the point representing the mean of the additional test points. From this line, a second-approximation design stress S_2 is read at the required life N_{Breq}.

As an alternative to these first two steps of this procedure, each of the total

LOADING SCHEDULES FOR PRELIMINARY TESTS
100 HOUR BLOCK

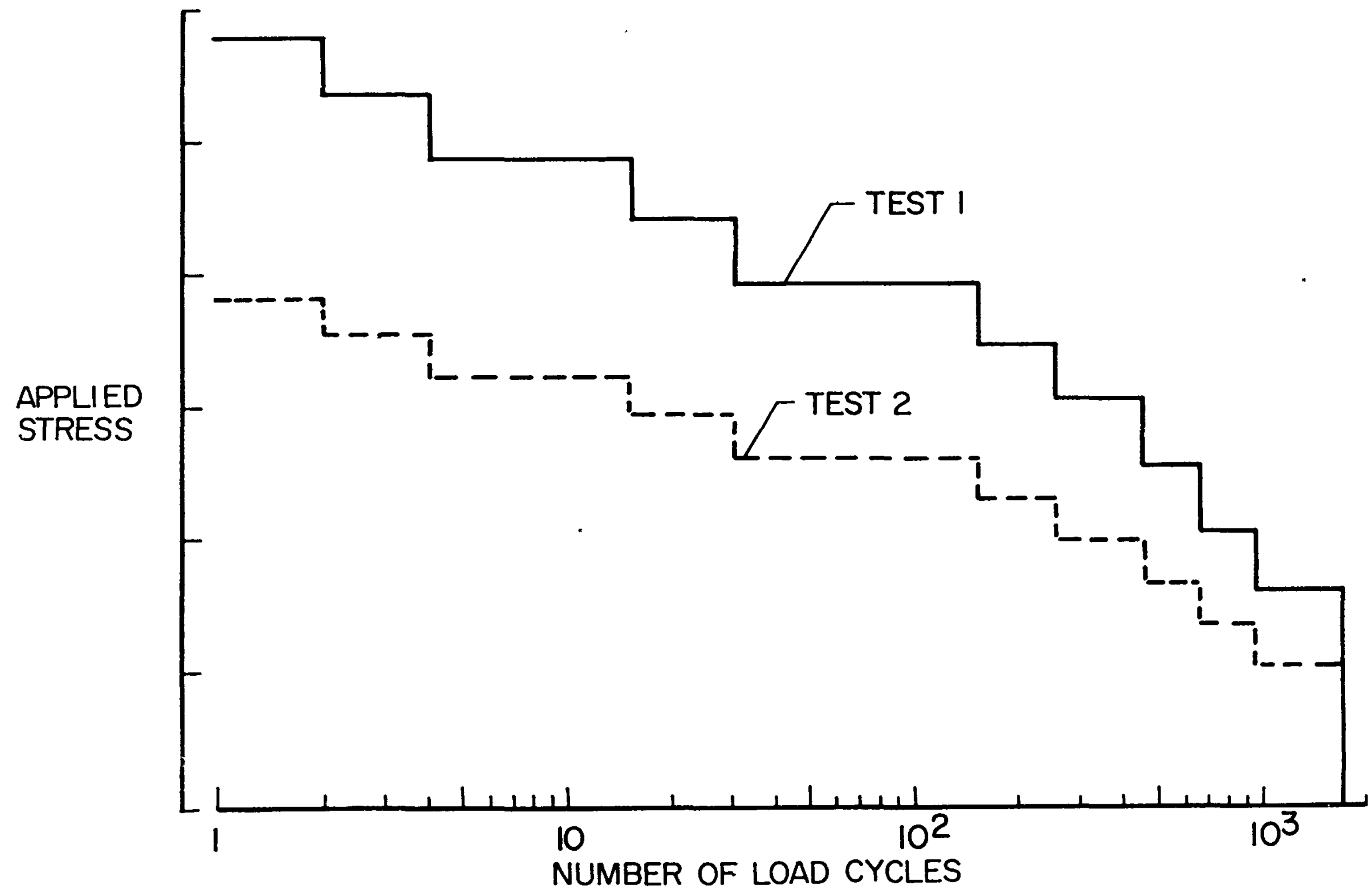

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FIGURE 3

PROCEDURE ESTABLISHING ROUGH VALUE OF DESIGN STRESS

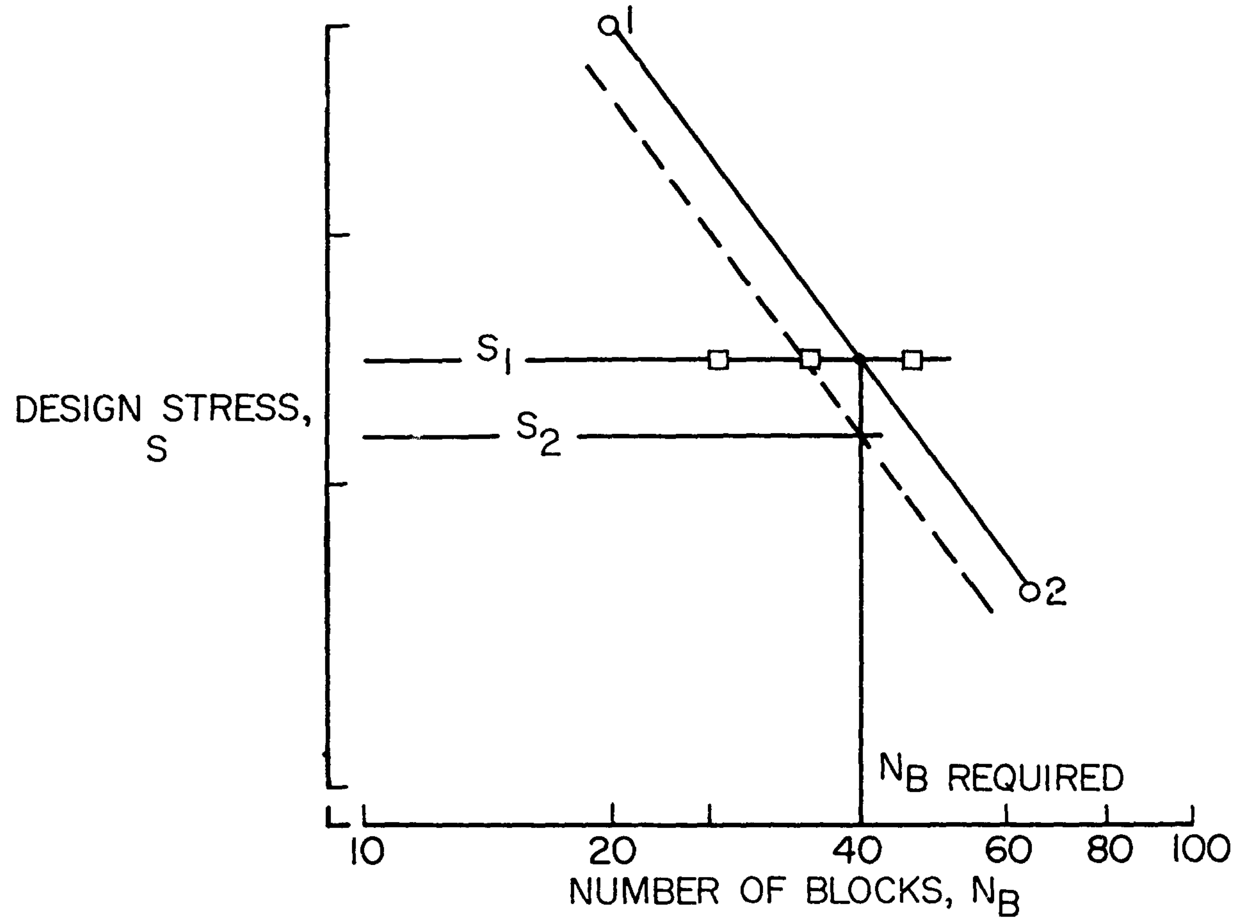

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FIGURE 4

number of specimens might be tested at different stress levels between the originally selected high and low values. The mean line through these test points would then, in effect, establish S_2 directly.

The next diagram (Figure 5) is a plot of probability of failure vs design stress. The stress S_2 is indicated on the right; only the central portion of the associated probability curve is shown in dashed form, because the limited number of tests made is sufficient to establish only an approximate mean value, but not other properties of the probability curve.

The design stress is now lowered or shifted to the left from the value S_2 to the value S_3 to account for effects present on the airplane, but not in the laboratory tests made. The main effects considered in this shift are exposure to atmospheric corrosion and imperfections in machining and workmanship.

The "true" probability curve of the laboratory specimens is not well established, as mentioned; it might look like curve A in Figure 5. Due to the small number of specimens, it is likely that all the material used was from one heat, all the machining was done by one man, the heat treatment was made simultaneously--in other words, a considerable degree of consistency and a consequently low scatter probably existed. In the full-scale production, more scatter, due to all the reasons mentioned and others, is likely; therefore, a broader probability curve B is used. This curve, together with an appropriately chosen percentage n of tolerable failures, establishes the final design stress S_4.

Two key items in the approach just outlined are the shift from S_2 to S_3 and the flattening of the probability curve. These items are so important that they should not be left to the arbitrary discretion of a single man, actually or effectively. The best experts available should be rounded up to combine their experience, weigh the evidence and decide. The decision on the tolerable percentage of failures, the third key item, is obviously a matter to be decided by regulatory authority in consultation with experts.

Those who have lived with the fatigue problem for many years may understandably feel that there is nothing really new here. Basically there is not. The only essential procedural difference between this and other similar-looking approaches is that it attempts to bring to bear the whole state of the art in a simple workable way in contrast to relying completely on a few tests or else attempting to bring the state of the art to bear in a complex, unworkable way. Further, it takes away from designers of varied experience the responsibility for setting maximum limits on design stress and places this responsibility on the regulatory agencies working through the collective experience and judgment of the nation's best fatigue experts in industrial, governmental and academic circles.

CONCLUDING REMARKS

In conclusion, I should like to restate in summary form some of the points I have attempted to develop.

PROCEDURE FOR ESTABLISHING FINAL DESIGN STRESS

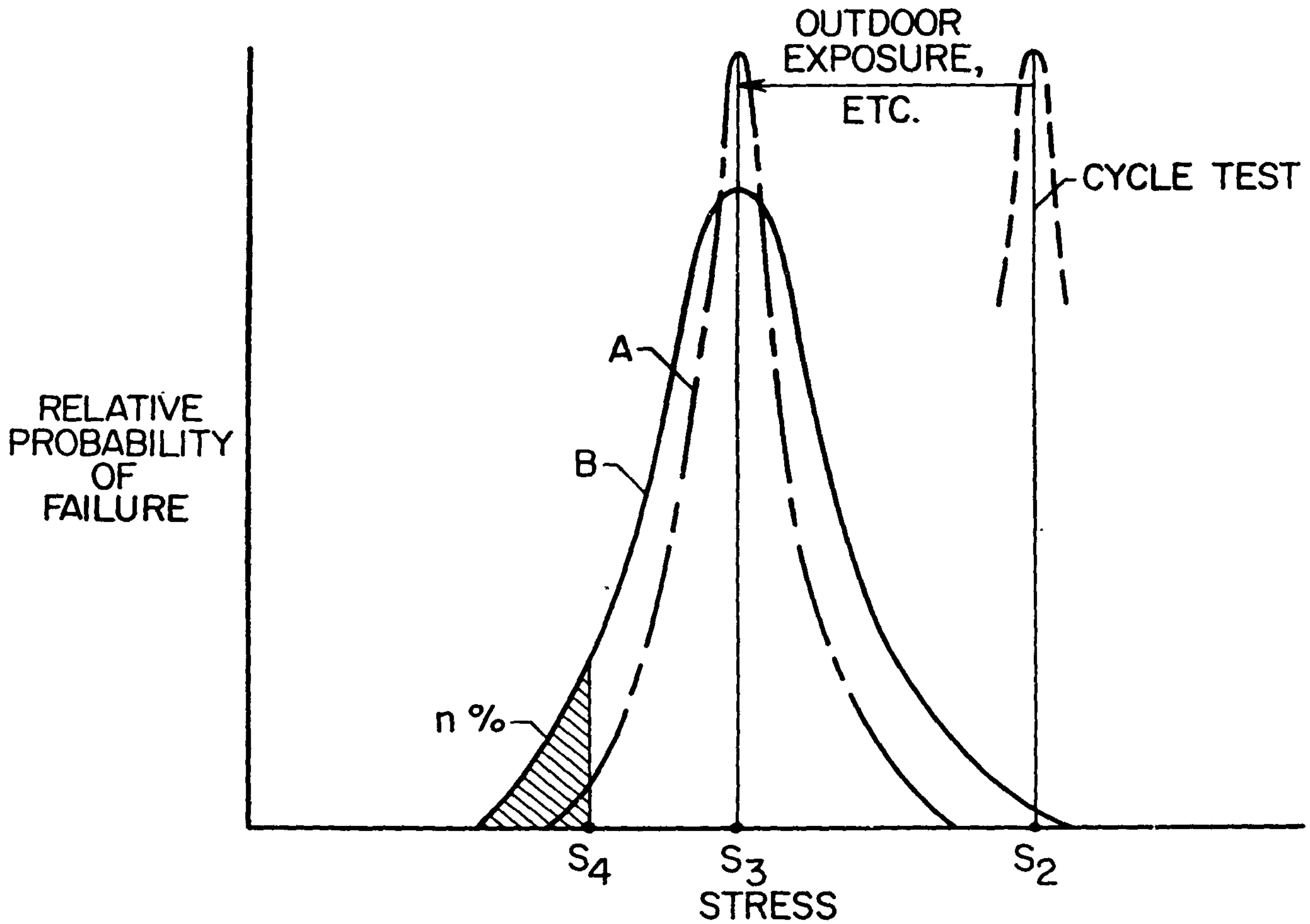

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FIGURE 5

1. One of the weakest points in the present fatigue situation appears to me to be the lack of adequate control of design stress.

2. A safe design stress cannot be adequately determined solely from the results of a few laboratory tests of structural components or even of complete structures.

3. Structural fatigue tests of components of particular designs are necessary, but the results of such tests have to be supplemented by other inputs derived from the whole state of the art.

4. In order for this process to be practicable, there must exist a simple, clearly defined framework of action or procedure.

5. Because individuals have limited experience, knowledge and freedom of action, the whole state of the art can be brought to bear only by making use of the collective experience, knowledge and judgment of a representative group of the nation's best fatigue experts.

6. The responsibility for such course of action resides in the regulatory agencies.

FAA PHILOSOPHY ON THE PROBLEMS OF FATIGUE IN RELATION TO CIVIL AIRCRAFT

By

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My objective today is to reflect briefly on the matter of structural fatigue as viewed by the Federal Aviation Agency. Our interests in this matter involve prescribing minimum safety standards applicable to the design of civil aircraft as well as insuring that aircraft designs actually meet such standards. This activity encompasses a wide variety of aircraft configurations classified as transports, personal aircraft, and rotorcraft. The uses and missions of these aircraft vary and therefore the means of demonstrating compliance with the fatigue standards will also vary. At this session I will direct most of the observations to the transport category field.

THE CIVIL AIR REGULATIONS

As you may know, all civil aircraft of U. S. registry are required by law to be certificated by the Federal Government as airworthy. The Federal Aviation Agency, by Act of Congress in 1958, (and the predecessor agency, the Civil Aeronautics Administration, by Act of Congress in 1938) is required to establish minimum safety standards governing the design of aircraft. The Administrator is empowered to grant type certificates on aircraft designs only after he finds compliance to exist with such standards. One such standard would obviously involve fatigue; present CAR 4b. 270(b), reference 1, applicable to commercial transports is a most important example of a standard in this area. However, the mere fact that a regulation exists on the subject is hardly sufficient since many questions are raised thereby requiring attention.

A reading of CAR 4b. 270 is indeed thought provoking. Quoting from 4b. 270(a) Fatigue Strength, "the structure shall be capable of withstanding repeated loads of variable magnitude expected in service" and paraphrasing 4b. 270(b) Fail-Safe Strength, "Catastrophic failures are not possible," how is it possible for one to translate such scientific phraseology in terms of human life values In this respect is it reasonable to expect any one individual to specify "an acceptable failure probability rate" as a criterion governing aircraft structural failures? And yet these are common terms, the pros and cons of which are universally debated today wherever discussions are held relative to aircraft structural fatigue. The FAA is in the position of having to find that an aircraft to be type certificated complies with standards couched in just these terms. Wisely, the Act permits the use of such elements as judgment, experience, and "state of the art" by the Administrator in exercising his powers and duties.

DEVELOPMENT OF STANDARDS

Until late in 1947, structural fatigue was not regarded as a major problem in the design of fixed wing aircraft. Catastrophic fatigue failures in civil aircraft rarely occurred without the contributing cause being a disregard of the classic design and manufacturing rules governing good design for fatigue prevention. Those fatigue failures which did occur were of such a nature as to be easily detected. They generally were amenable to control by inspection and replacement or repair. An excellent example of this technique is that old aerial work horse, the Douglas DC-3.

In 1948, a major fatal accident occurred to a newly introduced transport aircraft on an important domestic airline in the United States, caused by fatigue failure in the wing splice attachment fitting. Shortly prior to this case another newly introduced U. S. transport aircraft suffered a fatal accident due to fatigue in the vicinity of a fitting of its single spar wing. Other failures occurred on foreign aircraft. Further, in 1952, there developed a rash of fatigue difficulties on a number of U. S. transport aircraft. We are all, of course, aware of the much publicized events surrounding the two foreign jet transport accidents which occurred in 1954.

In approximately this same period, the aviation industry was entering into a world-wide race for turbine-powered, high altitude transport aircraft development. There was much pressure for a "rationalization" of civil requirements to eliminate hidden and overlapping factors of safety and to permit the use of more refined and analytical approaches to the problem of airplane design. Higher strength materials with little or no increase in fatigue resistance properties and somewhat more critical notch sensitivity were being introduced. The net result of these developments was in the direction of use of higher working stress levels in aircraft structures.

Realizing the seriousness and immediate nature of the rising problem of fatigue, our predecessor organization, the CAA, took its first official step in the form of a report which specifically highlighted concern for needed new fatigue safeguards, particularly for pressurized cabins. The report outlined the need for cyclic testing of pressurized cabin components, suggesting 10,000 cycles as a minimum criterion at that time. This proposal was followed, in the fall of 1954, by the first release for public consideration, prior to adoption in law, of the fatigue standards as now contained in Civil Air Regulation 4b.270. In their evolution it was only logical that complete coordination with interested industry and government segments be undertaken. This was done.

In April, 1957, the CAA released its proposed Appendix H to CAM 4b entitled "Fatigue Evaluation of Flight Structure." This contained guidance material of a non-regulatory nature to assist the applicant in establishing compliance with the objective regulations. After coordination with industry groups it was published in August 1958. Although somewhat brief and concise the regulations, coupled with Appendix H, provide a framework of minimum criteria for the guidance of aircraft designers today.

TRANSPORT AIRCRAFT FATIGUE STANDARDS

The current regulations allow applicants for type certificates the choice of either of two basic approaches to the problem of fatigue, concerning which you will hear much discussion during the next few days. These are: "Fatigue Life" (or "fatigue strength" method as it is called in the regulations) and the "Fail-Safe Strength Method." Both approaches are intended to ensure freedom from catastrophic fatigue failure.

The fatigue life method requires that by analysis and/or test the flight structure shall be shown capable of withstanding the repeated loads of variable magnitude expected in service. Finite lives are established for important structural elements requiring replacement thereof in accordance with established schedules. This determination necessitates the development of:

1. Typical loading spectra expected in service,

2. Identification of the principal structural elements and detail design points the fatigue failure of which could cause catastrophic failure of the aircraft; and

3. An analysis and/or repeated load tests of principal structural elements.

The fail-safe method provides for the possibility of occurrence of a fatigue failure of a structural element. This concept requires that it be shown by analysis and/or test that catastrophic failure or excessive structural deformation, which could adversely affect the flight characteristics of the airplane, are not probable after fatigue failure or obvious partial failure of a single principal structural element. Following such a failure, the remaining structure must be capable of withstanding essentially 80% of the limit design loads multiplied by a factor of 1.15 unless the dynamic effect of failure under this static load is determined.

Among several considerations, two main factors were taken into account in arriving at this value of 80% of limit design load. These were: (1) The actual failure experiences with U. S. transport type designs which had experienced major component failure in flight and yet had retained sufficient strength and rigidity to continue safe flight and landing at scheduled destinations, and (2) The probability of occurrence of accelerations of sufficient magnitude to cause catastrophic failure of the fatigue damaged member prior to detection of the initial fatigue damage.

RECENT EXPERIENCE

Most new U. S. transports have been certificated under the fail-safe method. This does not imply that fatigue testing is not being accomplished- - - -far from it. Fatigue testing is still quite necessary to develop the design information on the structure and for the development of inspection procedures and programs to insure that cracks can be detected before they become catastrophic in nature.

The success of the fail-safe design method in avoiding catastrophic failure depends largely on the effectiveness of inspection and its frequency in relation to the rate of growth of the fracture and also in relation to the operational conditions. Inspection programs must be such that any failure which could eventually become catastrophic must be found before it can propagate beyond safe limits and thus impair the structural integrity of the airplane.

The determination of the "principal structural elements" required by the fail-safe method is one of the key factors to the success of the method. This would seem most simple but in practice it is not. If one considers that all of the structure has some degree of importance in the over-all picture, the problem appears in its proper perspective. The type of damage expected and the means of testing presents further problems.

Certain of the foreign manufacturers of U. S. certificated imported transport aircraft have utilized the fatigue-life method. Also, some U. S. manufacturers have applied it to specific components. Necessary conservatism inherent in the method frequently dictates low initial life assessments which represents a serious drawback. The use of the fatigue-life method has not yet resulted in any reduction of aircraft inspection requirements, which one would think would be the case if one could truly rely upon this method of fatigue evaluation. Continuous monitoring by means of strain recorders has been necessary to obtain rational load data.

The choice of scatter factors in fatigue testing has been a problem. Values ranging from three to eight have been found acceptable when applied to various test programs. No hard and fast rules have been established for either load spectra or scatter factors. Each has been handled on more or less an "ad hoc" basis.

The use of either the fail-safe principle or the fatigue-life concept results in lengthy, laborious and expensive test programs to substantiate the design. The skeleton framework of the present CARs leaves much to be filled in by the designers and the FAA engineer who must certify to the results. The FAA has accepted both full scale testing and component testing in the various programs thus far. Analysis alone has not met with any great degree of acceptance in civil aircraft.

Civil experience in the U. S. is now unfolding as regards the application of the fatigue standards to transport aircraft. The indications to date speak in favor primarily of the fail-safe approach. Service experiences are being carefully followed by both FAA as well as the aviation industry. As experiences may dictate, adjustment of details of the methods and inspection programs for aircraft involved will be accomplished.

ROTORCRAFT AND PERSONAL AIRCRAFT

Up until now, this paper has been concerned primarily with transport category aircraft. As you know, the FAA is also concerned with the airworthiness of rotorcraft and personal type aircraft. The applicable regulations, Parts 3, 6 and 7 warrant every bit as much attention as do our transport requirements

contained in CAR 4b. Some brief generalized remarks regarding the pertinent Civil Air Regulations seem appropriate at this point.

Rotorcraft

Since fatigue failures on rotor systems and controls of rotorcraft are not likely to be forgiving and since fail-safe design is difficult to achieve, fatigue-life concepts have been long recognized as a necessary and fundamental approach to solving the fatigue problems of rotorcraft. The regulations specify that hubs, blades, blade attachments, and blade controls which are subject to alternating stresses shall be designed to withstand repeated loading conditions. For transport category rotorcraft the regulations are extended to include fuselage and rotor pylon structural components the sudden failure of which would threaten the structural integrity of the rotorcraft. The stresses for the critical parts shall be determined in flight at all altitudes appropriate to the type of rotorcraft and the service life of these parts shall be determined by fatigue tests or by other methods found acceptable to the Administrator. Recommended retirement lives of components are enforced by FAA as a mandatory requirement for airworthiness certification.

Personal Aircraft

In personal fixed-wing aircraft, fatigue failures have been predominantly of the nuisance type and have rarely shown up in the basic primary flight structure. No doubt this situation is the result of the relatively high design load factors employed with the resultant low effective normal operating stresses. However, violations of the classic design rules governing good detail design practice are still occurring. This situation cannot be ascribed to be a fault of CAR 3 because the standards presently contain provisions apropos of such practices. With the present trend towards higher speeds and pressurized cabins, these requirements will certainly require some amplification regarding fatigue, although pressurized cabins are already covered in a manner similar to transport aircraft.

FUTURE PROSPECTS - - SUMMARY

Drawing from the background of civil experience in fatigue of aircraft structures as just presented, the FAA must view the present situation as somewhat transitory. Recent surveys have shown that approximately one-half of all airworthiness directives issued to correct service problems in the area of airframe structures for aircraft are on the subject of fatigue. Changes and improvements of the Civil Air Regulations are in order, as the state of the art advances. In many aspects we are not yet in position to assess with positiveness their full degree of success; however, present indications are that some changes to the present civil codes may be warranted as follows:

A. <u>Part 4b Transport Aircraft, Including Projected Supersonic Transports</u>

1. There is a need to fill-in the present framework of regulations with enabling standards to give substantive meaning to generalized and objective terms now freely employed. For example: What constitutes the idealized loading spectrum or what is a single principle structural element etc., are typical criteria in need of defining in the standards.

2. Some indications to date support a need for expansions of the fatigue investigations to include other structure than flight structure, as now specified.

3. There is a need to introduce consideration of all essential environmental conditions as are of importance to proper evaluation of fatigue characteristics. The high temperature effects and stresses and sonic phenomena particularly for supersonic transports are definite considerations.

4. By carefully following service experiences of new transports, we may need to require improvements in methods and procedures now employed in the type design approval process as may be clearly necessary and proper.

B. <u>Part 3 Personal Aircraft</u>

1. We need to expand the present standards for fatigue prevention to provide for advanced airplane designs, particularly high-speed altitude personal aircraft designs.

2. For low speed airplanes, we may need to consider including in the standards provisions for a comparatively simple approach to fatigue substantiation; this approach would accept sufficiently low normal operating stress levels as a means for ensuring that the lives of important structural components are well beyond the airplane's operational life.

3. Appropriate steps should be taken to emphasize to airplane designers, (particularly the young engineers who may lack an appreciation of lessons learned from the past) the importance of good detail design practice. A designer's good practice handbook as an adjunct to the Civil Air Regulations would be of great help.

C. Parts 6 and 7 Rotorcraft

1. There seems to be a need for more specific regulatory criteria in lieu of the present objective regulations and the more comprehensive guidance material of a non-mandatory nature.

2. Changes in standards are needed consistent with the present state of the art and to reflect service experience to date. This experience is reaching a point where more definite conclusions can be reached in relation to fatigue requirements as they apply to rotorcraft.

CONCLUDING REMARKS - In behalf of the Federal Aviation Agency I wish to express our appreciation for having been invited by the United States Air Force to address this important gathering on our Philosophy on the Problems of Fatigue in Relation to Civil Aircraft. There is no question in our mind that the problems of fatigue are serious and demand our all-out effort. Much remains to be done in their peculiar solution to aircraft and missile construction. The FAA is looking forward to working together with the U.S. Air Force on this vital structural integrity problem by contributing its knowledge and experience. We believe that we are on the right track so to speak with our current designs. With the combined efforts of the designer, the operator and the government bodies, the FAA looks forward with confidence to the JET AGE.

REFERENCES

Reference 1, CAR Part 4b Airplane Airworthiness Transport Category

THE AIRCRAFT FATIGUE PROBLEM: BAREL Y UNDER CONTROL

By

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The title of this paper is not an inept attempt at frivolity. The fatigue problem is much too serious for anyone to treat it with any levity. Nor is it an indication that I shall spend the next twenty minutes "viewing with alarm." The situation that I shall describe is under control, but only through continuous vigilance and determined effort of everyone concerned. More than this will be needed in the future. Only by far better understanding of the nature of fatigue and methods of coping with it from design through test, evaluation and service use can we prevent this situation from getting out of hand. The cooperative effort represented by this symposium is one of the reasons I am not alarmed although gravely concerned by the difficulties ahead.

The examples I shall present of the operational threat of fatigue problems are all concerned with major structure of piloted airplanes. Individually these present the most immediate operational problems, but this should not be construed as minimizing the seriousness of the aggregate effect of difficulties with fatigue in other aircraft components. We currently have fatigue problems with power plants that are just as troublesome operationally as the structural "headaches" I shall cite. This discussion could also include a wide variety of circumstances where time-temperature effects add new dimensions to the consideration of repeated loads in the design of unmanned vehicles. Rather than survey the entire field, however, my remarks will be limited to the airplane airframe structural problem.

Every single one of the Navy's first line carrier aircraft has a fatigue or repeat load problem. On each our attention may not relax a moment lest an unacceptable operational situation develop. One example that I shall elaborate on illustrates particularly well the unfortunate consequences of assuming that we could take a short breathing spell. As we go through the list of some of these aircraft, the nature of the principal difficulties with each, and the narrow margin of control we have over each situation, you will share I am certain the sense of urgency which must be behind the structural integrity program. If I devote more attention to some aircraft than others, it is not to criticize their engineering development more than that of others. These cases either illustrate the operational effects of the problem more dramatically or the benefits of an intelligent determined attack on the problem when it occurs.

As I mentioned every single model of carrier aircraft is involved in this problem. I shall describe briefly the nature of the particular difficulty with the F8U, F3H, F4D, F11F, FJ4, A4D, and A3D. Other models of carrier aircraft in service and patrol aircraft could be added to the list. The major emphasis, however, will be on the influence on the attack potential of carrier forces, if the problem got out of control. There will be no attempt to go into technical details of the nature of failure involved. I shall not, in fact, try to distinguish between true fatigue and what might be better described generally as structural

fatigue under repeated loads. Our design requirements for structural integrity lump these together, for practical purposes, so we shall do the same in considering the operational implications of the problems.

The F8U problem affords an example that could have had very serious implications on the aircraft program, but fortunately has been scaled down to the level of a major scheduling nuisance with far less cost than earlier expected. In early repeated load tests of the F8U-1 wing, results indicated a threat of catastrophic in-flight failure close to the end of the first service tour. This situation would have required periodic replacement of complete center wing panels at a cost of about $50,000 per airplane and serious extra workload. Viewed pessimistically, this addition, on the order of $20,000,000 to the F8U program could only be met by buying fewer airplanes with a consequent reduction in fleet aircraft inventory. A successful fix, the usual external doubler, was evolved in time to avoid the necessity of ordering hundreds of extra wing panels. Installation of the change is now in process, eliminating the worst of the threat to the F8U program. It should be emphasized, however, that the wing with the doubler does not have an indefinite life; it is only sufficient for the currently planned service life of the airplane. The big uncertainty still confronting us is whether the repeated load test program has adequate correlation with service load experience. There is some evidence that the results of the laboratory program are conservative, but we must never relax our vigilance in keeping track of the service usage of the model.

The history of our difficulties with the F3H is probably well known to most structural engineers in the industry. The most urgent fatigue problems, with major wing fittings, were discovered almost accidentally. In the course of a general program of research in fatigue at the Aeronautical Structures Laboratory at the Naval Air Materiel Center, a dangerous fatigue life deficiency was discovered in the wing hinge fittings of some surplus F3H airframes being used in the research program. Other major deficiencies in wing carry-through structure were found. In spite of a determined cooperative effort of the contractor and the Structures Laboratory, the airplane went through a period of several years of service operation with operating restrictions that made it barely capable of performing its primary mission and severely limited its operational flexibility.

The wing fatigue problem of the F4D affords a good example of a situation that has not severely threatened the operational effectiveness of the airplane, but keeps the structures engineers from sleeping too soundly. The fatigue deficiencies that showed up during early repeated load tests appear to be of the type that will give some warning through simple inspections before the failure progresses far enough to be catastrophic. A fix is required, nevertheless, because a life less than the service tour is indicated. As long as the airplane is operated in high altitude missions, the load spectrum used for laboratory test is considered conservative. The principal effect of the fatigue problem with this model is then to add a maintenance burden -- and, of course, the cost of a change. One of the fatigue problems of this model had its lighter side. The wing hinge fitting, thought to have a good fatigue life, turned out to be susceptible to damage from unexpected load components resulting from manufacturing misalignment. An F4D showed up at Marine Corps Air Station, El Toro, after a cross country flight with one of the two lugs of the fitting missing. The lug was nowhere to be found. It must have dropped out when the wings were folded at some stop enroute. Naturally proper alignment of this fitting received immediate attention.

The F11F has had a repeated load problem of a different nature. As the weight of the airplane grew (weight growth is like death and taxes), the landing gear proved inadequate for repeated carrier landings. As you know, these involve sinking speeds way in excess of those for land operations. At first it appeared that a program to develop adequate strength for repeated carrier landings would involve millions of dollars and over a year in time. This posed a serious operational dilemma when considered with the projected program of the model. The major difficulty, fortunately, turned out to be the result of progressive deformation of the landing gear. A satisfactory solution was developed without the long expensive engineering program that had originally been expected. We still must watch the landing incident history of the F11F very closely.

The FJ-4 represents the typical airplane fatigue problem more completely than any other model. First, the operational history of the model exemplifies a principal cause of getting into fatigue troubles. The FJ-4 was considered to have adequate fatigue life for its original fighter mission. When it was converted into a fighter-bomber, or attack aircraft, "all bets were off." The extra operational weight plus the increase in severity of loads at low altitude changed the fatigue situation for the FJ-4 drastically. The most serious effect, however, turned out to be the increased difficulty in staying within the "v-g" envelope in low-level training exercises. After a fatal accident, conclusively determined to be caused by a major wing fatigue failure, it was evident that the service usage of the airplane had reduced the wing fatigue life from thousands (in its original mission) to only a few hundred hours. A crash program ensued. This occurred when the world situation was especially tense on both sides of the globe. For efficiency in maintenance and support, almost all of the FJ-4's were in one ocean. But this meant that a major part of the naval aircraft attack potential in that ocean was involved in the FJ-4, so the dire consequences of grounding or severe restriction of them was obvious. Good fortune prevailed again, because it turned out that incipient fatigue damage could be found by a borescope inspection of certain bolt holes. Reaming and bushing the holes restored the wing to its original fatigue integrity. This still left the serious problem of improving the fatigue life of the wings before the situation again became acute. Here another example of the necessity for eternal vigilance transpired. We were confident that we had a year to develop and get a fix into service based on expected usage of the airplanes. At the end of about six months, much to our consternation, we learned that some FJ-4's were being operated at about twice the hourly rate expected. Others were being flown hardly at all, it is true, so the average was about that planned. It was the old story of being trapped by statistics. There was no alternative but to set a limit on operating hours and try to accelerate the program to incorporate the fix. This is now going into the airplanes without too serious a reduction in availability, I hope. Once again, though, I must emphasize that the fix does not give us an indefinite fatigue life for the wing, but only enough for limited service life.

The A4D is included in this history primarily to illustrate the assertion that no airplane is free of some structural fatigue trouble. I must admit in this case that the troubles are of the variety that have been familiar to engineers since long before the current problems with relatively brittle materials used in critical single load-path members. The A4D problems have been confined to the fuselage and engine support structure.

The A3D, on the other hand, has been afflicted by a distinctly modern ailment, sonic fatigue. The primary operational effect has been the increase in maintenance effort

caused by cracks in the tail and aft part of the fuselage. The effect of sonic fatigue on the tail structure has presented a dilemma similar to that which may face the missile designer. Relatively short periods of ground operation at high power causes damage to the stabilizer structure. This has been very difficult to cope with even though the initial effect is not dangerous. The A3D has also encountered the problem involved in a change of mission. It has been necessary to develop changes to improve the fatigue life of the wing for loads encountered at low altitude. The A3D has not presented more serious difficulties to operating commands in this connection because information on the planned change in mission enabled responsible activities to program required service changes. It was necessary, however, to impose a requirement for periodic inspections prior to installation of these changes and to recommend to fleet commands that the amount of low altitude flying done be monitored to avoid accumulating excessive time on individual aircraft at low altitude.

While the foregoing recital has not been intended to paint a gloomy picture, the over-all effect should dispell any complacency. To return to my original figure of speech, a machine barely under control, there is a parallel to a very complicated dynamic system. It is known empirically to be unstable, but otherwise has unknown physical characteristics. The engineers have devised controls by trial and error that operate successfully but inefficiently within limited dynamic boundaries. They know that any change in environment or major disturbance to the system is likely to produce divergence. So eternal vigilance to keep all disturbances small is the only way to keep under control. Meanwhile a major program of research and study is essential to find out more about the mechanisms of the system and to be prepared for the changes in operating environment that must be anticipated.

In summary the operational aspects of the fatigue problem with Navy aircraft may be categorized as follows:

1. We have had to recommend or impose flight restrictions on some models which have hampered operating commands in training pilots for the intended missions of the airplane.

2. We have had to fund and schedule unanticipated modification programs on a priority basis.

3. We have been confronted with actual or probable usage of expensive spares at rates far in excess of normal or reasonable provisioning.

4. We have had to impose special, periodic, structural inspections on operating squadrons.

5. We have had to impose additional administrative workload on operating units in connection with timing the installation of changes in individual aircraft.

What is a sound approach to long term solutions to the fatigue problem? Solutions are certainly not satisfactory which impose operational burdens such as:

a. Replacing major structural sub-assemblies periodically, or

b. Establishing elaborate administrative procedures to monitor the operation of fleet

aircraft and rotate their assignment so as to achieve uniform usage, or

c. Requiring adherence to maximum flight operating restrictions which provide essentially no margin with respect to the minimum flight envelopes necessary for the airplane's missions.

In the long run fatigue must be handled as any other engineering problem. Attainment of adequate fatigue life requires recognition of this as a design objective just as performance, weapon compatibility, maintainability, and mission versatility are so recognized.

It is an obligation of the Navy to define this design objective adequately. We cannot expect a contractor to guarantee a minimum life for an aircraft, but we do expect contractors to agree to a design fatigue life. The Navy must produce a design fatigue life based on the relation of expected operational missions and associated loading spectra to the probable operational life desired.

Such a design fatigue life must then be introduced as a part of the requirements governing design proposals. We can see no way at present to accomplish proof-of-design other than by laboratory fatigue testing of articles as representative of production aircraft as possible.

A great deal remains to be done to achieve a satisfactory state of affairs in handling fatigue as an engineering design problem, reducing the operational complications to an acceptable level. The services in cooperation with the National Aeronautics and Space Administration and the Federal Aviation Agency are approaching their share of the responsibility by a joint effort. This cooperation is particularly important in applied research on flight loads and development of design requirements for structural integrity. A broad basic research program on the nature of fatigue is essential. The interim measures of an operational nature for coping with the fatigue problems of airplanes, barely satisfactory as they are, fail completely in application to engineering for missiles and space vehicles.

The cooperative effort required for real solutions to the engineering fatigue difficulties with airplanes has potential value throughout the whole space of transportation from the ground to the stars. This gives me confidence that we will not lose the motivation now behind our united attack on the problem.

PROBLEMS OF BALLISTIC MISSILE WEAR-OUT AND STRUCTURAL SERVICE LIFE

By

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I. INTRODUCTION

Operational requirements for ballistic missiles to date have not attempted to specify precise requirements for service life or wear-out period. The complicated relationships among factors which have bearing on the question, such as component maintenance and replacements, facilities and personnel requirements, logistics and supply, operational doctrines, and costs have not been satisfactorily evaluated; and since ballistic missiles are just now entering into operational capability, experience is lacking. No doubt some of the important issues related to the entire question have not yet been recognized.

The term "service life" applied to the current concept of a ballistic missile obviously does not imply an ability to perform a number of operational missions, for the number of missions possible per missile clearly is one. Service life in this respect must imply the ground experiences which must eventually require the retirement of the missile as an operational item without being fired.

Even though precise requirements may not exist initially, it would be foolhardy to ignore the problem of service life until such time that all requirements have been stated precisely. Common sense would indicate that a complicated and costly ballistic missile must be capable of surviving the operations which must be performed on the ground for many months, and perhaps years, before being launched. Clearly, the pre-launch history must not be allowed to lower the probability of success of the mission below acceptable limits. However, it is probably less important to be worried initially about the exact service life or wear-out period for a ballistic missile than it is to attempt to insure, through systematic design, development, and testing, that a certain minimum service life can reasonably be anticipated. If later experience or military doctrine should require a change in the assumed life requirement, the design studies, development, and testing already undertaken in this area will provide the simplest and surest guides for effecting the necessary changes.

Although the foregoing concept applies generally to the entire missile system, including ground support facilities as well as all subsystems of the vehicle, only the area of airframe structural design will be considered in the following remarks. Further, major consideration here will be limited to structures for liquid propellant ballistic missiles, which have achieved a more advanced stage of development than solid propellant ballistic missiles. Factors which must be considered in connection with structural service life include:

1. Susceptibility to structural fatigue for various repeated loading conditions expected in the air and on the ground;

2. Vulnerability of the structure to accidents, sabotage, or enemy action on the ground;

3. Resistance to atmospheric environmental factors, primarily corrosion.

II. DESIGN ENVIRONMENTS AND SERVICE LIFE

An extremely important objective of ballistic missile structural design is to achieve the lightest weight structure within practical limitations that is consistent with over-all system objectives. Clearly it is not desirable to penalize the structure size and weight by designing for non-flight conditions except in the over-all best interests of the system, and only then when it can be determined that the chances for the structure's successful survival of the flight environment will not have been impaired by its earthbound experiences.

The first and most important step in design - as well as one of the most difficult problems - is to decide which environments must be considered as design conditions, and how long a lifetime must be provided for exposure to or repetition of these environments. Having decided these things, it then would be desirable, ideally, to be able to predict exactly when any specific missile must be retired from service, or the last moment it may safely be launched, just by knowing its entire history of transportation, handling, training and operational exercising, exposure to the elements and damage by personnel. Actually, such an accomplishment is extremely unlikely, but one may approach this goal by careful consideration of environmental factors, proper design procedures, and adequate test programs, along with the application of information and judgment accumulated during the development of the weapon system.

It is convenient to consider the design environment in two parts: post-launch, or flight, and pre-launch.

Flight Environment. In the flight environment are a great number of conditions. After the missile leaves the launch pad, it is subjected to acoustic noise from the engines, engine thrust forces and vibration, aerodynamic heating, aerodynamic forces, engine swiveling control forces and vibration, propellant sloshing motions in the case of liquid propellant missiles, internal pressures and their fluctuations, and possibly panel flutter. If the missile is launched from an underground enclosure or "silo" a severe and concentrated transient environment will also develop, including acoustic pressures, aerodynamic forces, and exposure to the hot products of combustion from the rocket engine jets.

Considering the fatigue life of the structure, it is frequently remarked that since the powered flight duration is in the order of five minutes there should be no fatigue problem. True, it is generally believed at this time that the flight environment considered by itself has not been a governing factor in establishing service life, and that one does not consider fatigue in the most usual sense, i.e., many cycles of repeated stress at relatively low levels superimposed on moderate steady state stresses. However, in spite of the short duration and possible low cyclic stress levels, the in-flight environments must not be dismissed as unimportant from the fatigue standpoint. Conditions in flight most likely to induce fatigue are the vibratory stresses produced by aerodynamically excited panel flutter, by acoustic pressures, and by engine vibration, particularly in the thrust transmission path

and at equipment attachments. Frequencies in each case may be several hundreds of cycles per second, so large numbers of cycles can occur even in a few minutes of flight. Furthermore, since low design safety factors are employed, and since design conditions may be expected to occur in each flight, the cyclic stresses mentioned are in some cases superimposed on high static stress fields. In certain areas the same structure has also survived many stress cycles on the ground and may have accumulated considerable fatigue damage.

To date the flight conditions have been accounted for in design by using static stress allowables, making allowances for high temperature-short time properties when appropriate, but to my knowledge the repeated stresses from the flight environment for current ballistic missiles have not required fatigue life reductions of allowables. The degree of over-design present in existing structures is frequently unknown. It is fair to conclude, however, that with every effort made toward design improvements leading to structural weight reductions, conditions which may not now be critical will become increasingly important. Refinements in loads determination and statistical prediction, and better understanding of detailed structural capabilities will inevitably lead to this conclusion.

Pre-Launch Environment. For the pre-launch environment, it is more difficult to define either all the environmental factors or their durations and numbers of occurrences. Typical conditions to be accounted for include the following events:

1. Fabrication and acceptance proofing;
2. Transportation;
3. Handling to and from storage;
4. Erection for captive firing or launching;
5. Thermal and pressurization cycling of propellant tanks;
6. Captive firing;
7. Accidental damage due to personnel errors and equipment malfunctions;
8. Exposure to the elements while on an outside launcher, including wind induced oscillations;
9. Ground shock from earthquake or enemy attack.

Circumstances under which these events may occur include crew training exercises, operational checkouts, readiness alerts and holds, and enemy action. Static and dynamic loads, mechanical and acoustical vibrations, repeated temperature cycling, and repeated internal pressure cycling are variously encountered during the conditions described. Decisions must be made which govern the extent to which a missile is to be protected and to what extent it must be able to sustain all these conditions without seriously endangering the success of its mission or needlessly compromising the entire system. For this reason,

interaction studies must be undertaken to determine the over-all effects on the entire system which result from different basic missile capabilities and limitations imposed by environmental factors.

Even though in the interests of weight reductions it is desired to protect the missile airframe whenever feasible from the pre-launch environment, it has nevertheless developed that the pre-launch experiences of the missile are those which most strongly influence and limit service life. The various ground environments believed most significant in affecting structural service life are those which could induce structural fatigue failures. These include acoustic excitations due to static firing, vibratory inputs due to transportation, propellant tank pressurization and thermal cycling during repeated filling and draining operations conducted for many reasons, and bending excitations due to repeated exposure to ground winds. Additionally, of course, the general susceptibility to structural damage due to sabotage, accident, and corrosion, as well as the relative repairability of the airframe must always be limiting factors.

III. DESIGN CONSIDERATIONS

It should be recalled that fatigue is almost always a detail design problem. In order that fatigue damages be minimized three basic concepts must be followed, namely:

1. Use materials which have a long fatigue life, particularly at the low cycle-high stress end of the S-N diagram. Fatigue life must be determined under the ambient conditions present in the particular part of the missile, and include all important interactions;

2. Use detail design procedures for all structural components which minimize steep stress gradients and stress concentrations since these are inherent fatigue initiators;

3. Protect against stress corrosion by proper techniques.

Probably the most important of the items mentioned above is the second, since local stress raisers can initiate fatigue failures in even the best of materials and, once a fatigue crack starts, it is self-propagating with additional stress cycles due to the induced high stress concentration at the ends of the crack. It may be well to interpolate at this point that, whereas in aircraft design, fail-safe and crack-stopping concepts can be built into the structure so as to avoid, or minimize, the propagation of fatigue cracks, these techniques always lead to structural weight penalties. Such weight penalties while costly may be permissible in both military and commercial aircraft. On the other hand, similar excesses in weight might jeopardize the entire mission of a ballistic missile and could not therefore be tolerated. For example, in the case of the pressurized integral propellant tanks (which constitute much of the primary structure in large ballistic missiles) a leak which becomes a crack is in effect nearly always catastrophic, for not only are propellants lost that are vital to the mission, but also pressurization is lost which is vital to structural integrity. Crack stopper design techniques in the tankage cannot be substituted for crack prevention design features.

Three environmental problems may be mentioned as requiring special attention in design. They are the effect of acoustic pressure, the degradation of service life by transportation, and the ground exercising of missiles.

Acoustic Excitation. The acoustic field around a large rocket engine contains large amounts of pressure energy which can exceed values of 170 decibels and provide pressure levels of one pound per square inch. Careful consideration must be made during the design of the structure and components to provide a sufficient number of attachment points for hydraulic lines and cables as well as to provide sufficient strength and stiffness in panels particularly in the region of the engine compartment. It may be well to subject typical structural sections to comparable noise levels to establish failure criteria although the captive firing test of the complete system may frequently be considered adequate confirmation of the structure. It is interesting to note that during the development of the Atlas, Thor and Titan missiles, no failures have occurred which could be specifically attributed to acoustic effects.

Transportation Effects. Concurrent with the detail design of the missile, the method of support of the device in its transtainer must be considered. Certain structural elements while sufficiently strong for flight and launching pad loads, may be subjected to stresses during transportation that may lead to fatigue damage. In many cases, minor redesign of the missile structure can eliminate these potential trouble spots without weight increase. This should be done whenever possible since it tends to make the missile more foolproof. Where changes in the missile structure cannot be made without serious weight penalties, it is then necessary to incorporate in the transtainer additional support or shock absorbing devices to prevent damaging vibratory loads occurring in the missile structure.

Ground Exercise. Perhaps of most concern, from the standpoint of service life, are ground operations which may apply design loads a number of times prior to flight. Such operations are introduced, for example, by the fueling and defueling of the propellant tanks during captive tests, training exercises, operational check outs and ready alerts. The filling and pressurization of these tanks with the attendant thermal shock and pressure effects produce high stresses. Unless the design conditions allow for repeated operations of this sort, there would always be concern that the missile would fail at the next pressurization. As the worry about the availability of the missile as a weapon increased, there would arise a demand to take it out of service and replace it with a new missile, possibly before the useful life of the missile had, in fact, been reached. Thorough fatigue analyses and suitable fatigue development and demonstration test programs are required to give the desired assurance of service life.

IV. DEVELOPMENT AND TEST PROGRAMS FOR STRUCTURAL FATIGUE RESISTANCE

In order that a reasonable service life expectancy be achieved, development of fatigue resistant design and demonstrations of this accomplishment are required by suitable testing programs for materials, components, and entire structural assemblies. The captive firing programs for the R and D versions provide the basis for most of the fatigue resistant design confirmation demonstrations, for they subject the structure repeatedly to acoustic and mechanical vibrations from engine firing, and to repeated propellant tanking and detanking

operations which include cryogenic and pressure cycling. Certain additional development testing programs are required, however, in addition to the captive tests. Some of the more important ones will be discussed briefly.

Transportation Effects Tests. In order to determine the effect of the loads transmitted to the structure during transportation, hundreds of miles of actual instrumented transtainer trips may be required in order to obtain the possible load spectra caused by various types of road surface, velocity of motion and other variables involved in moving the device from one part of the country to another. Finally, complete missile stages are installed in the transtainer and given (typically) a 500-mile road test. Transportation by air involves similar problems as well as certain environmental conditions peculiar to air operations.

Structural Component Development Testing. In order that a reasonable number of pre-flight operational exercises can be assured a series of fatigue tests in materials, structural components, and complete assemblies must be made. The list below gives the usual sequence of testing although several of these tests may be run simultaneously:

1. Coupon tests on simple material samples. These are run at a range of stress levels which bracket the expected operational stress levels in the actual missile. The tests must be run under the proper environmental conditions since a material which has a high fatigue life at one temperature (for example) may be poor at a higher or lower temperature. This series of tests is used to choose the most promising materials from the standpoint of static and fatigue strength/weight ratios.

2. Coupon tests on simple structural elements. Small scale tests may be run on such structural details as joints and may involve such variables as spot-weld spacing and types of joints (lap or butt). Again these tests should be carried out for the complete range of expected stress and environmental conditions.

3. Small scale tests of major configuration assemblies. For example, scale tests of pressure vessel assemblies may be carried out to determine the effect of such variables as methods of construction, stress concentrations due to changes in cross section, effects of constructional tolerances, biaxial stress fields, etc.

4. Full scale tests on critical assemblies. Items which may require full scale testing could be the engine mounts, the main propellant tanks, the pressurization gas storage bottles, etc., any one of which might cause serious damage if failure occurred during operation.

5. Lastly, but not least important, is the fatigue testing of the complete missile. This should be carried out for the full range of loadings expected during prelaunch environmental conditions that are expected to be considered satisfactory for such operations.

Due to the statistical nature of fatigue failure, one must have many data points in order to employ statistical methods of analysis, and thus obtain a figure for fatigue life

with a high degree of probability. Such a technique is possible for the coupon tests of materials and simple structural elements but becomes economically unsound for full scale testing of major components and complete missile assemblies. Obviously, as more and more missiles of one type are ground and flight tested, a body of data will become available for use in predicting future missile life expectancies. Until that time, it will be necessary to carry out the full scale testing using many more cycles than would be expected in actual use in order to have a reasonable probability that the required number of cycles could be utilized. For example, if 30 operational cycles were required, a test of two or three missiles to 300 cycles would give a reasonable probability that the complete family could meet the operational specifications.

At the present state of missile development, complete test programs appear essential in order to establish a background of knowledge on the effects of cyclic stresses superimposed upon an already very high stress field. Once the critical areas have been established by both tests and operations, the test programs can be modified to cover those areas which have been changed by redesign or by new performance requirements. At no time, however, can the basic requirement of designing for fatigue as well as static strength be ignored since a failure is still properly designated as fatigue whenever it occurs after one cycle of operation.

V. SUMMARY

A few of the technical problems associated with missile service life as affected by structure, have been discussed. Most of the critical conditions have thus far been produced by the non-flight environment. The difficulties encountered that are somewhat different from conventional aircraft considerations are those associated with the specification of the environment, the generally high stress levels to which the structure is designed due to low safety factors, and the complexity and cost of test programs. In addition to these technical problems is a psychological one of lack of awareness that a device which is only to fly for two to five minutes can have fatigue problems and that suitable design development and test programs are necessary.

COMBATING STRUCTURAL FATIGUE IN AIRLINE OPERATIONS

By

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Colonel Taylor asked for a paper on the Air Transport Association. He said he'd like the general outlook of the airlines on fatigue and I can preface my remarks by saying that the airlines are definitely "agin it"! The primary purpose of this talk is to present a brief outline of the cooperative effort which exists between the airlines, the manufacturers and the government in combating structural fatigue in airline operations. Included is a brief resume of the accident record as it has been affected by structural fatigue. Progress in aircraft design has been very rapid during the last thirty years. The judicious use of better materials has increased the efficiency of the structures, but at the same time the structural engineers' problems have become more complicated. Each new type airplane presents different aspects of fatigue, requiring constant vigilance on the part of the manufacturers and the operators. The literature on this subject reveals structural fatigue problems in axles of railroad cars more than a half century ago. These were investigated and solved at that time, and, therefore, a railroad engineer must feel pretty sure of himself today when designing a new model, since the configuration of railroad car axles looks to a bystander to be pretty much the same now as then. In aircraft design, however, the forward pace has been terrific and the aircraft designer must use all his ingenuity to produce the efficient structures required by modern air transportation and, at the same time, reduce to a minimum the numbers of fatigue cracks which continually appear to plague him and the aircraft operator. Before we start being critical indirectly of the aircraft structural engineer (by the nature of this assignment), we should in all fairness recognize his tremendous accomplishments. Where, outside of aviation, do we find such demanding structural design requirements dictated by the everlasting battle against weight and space. Many DC-3 airplanes have accumulated over 50, 000 hours of flight and are still going strong. More modern aircraft have accumulated over 30, 000 hours of flight with the same excellent results. That these vehicles have accomplished their purpose is obvious--- they have been good enough to serve as the foundation for the present huge air transport industry which, by comparison, was non-existent thirty to thirty-five years ago. However, unlike the case of the railroad car axle, the problems get more complicated with each new design. A glance at the subjects listed on the agenda for this symposium will attest to this fact.

At the present state-of-the-art, it is obviously impractical to say the least, to reduce all allowable working stresses in all the elements of an aircraft structure to values which would absolutely guarantee no fatigue cracks for operating

lives up to and exceeding 50,000 hours. Since this cannot be done, a certain incidence of fatigue cracks is inevitable, and it is, therefore, essential that the operators of the aircraft employ maintenance methods and procedures designed to find and correct actual or incipient fatigue failures before they begin to have a significant effect on safety. The scheduled airline operators have had a lot of experience and are now expert in this field of endeavor.

Before going on as to how the industry works on this problem, it might be interesting to spend a few moments on the safety aspects of fatigue in scheduled airline operations. It is our business in airline maintenance to deal with the things that are wrong with airplanes and, therefore, we hear about, and come in contact with, a large number of structural fatigue problems in aircraft structures. Subconsciously, our minds are apt to contain a fairly dark picture of such fatigue as a contributor to airplane accidents. With this thought in mind then, let us as they say, "take a look at the record."

The modern, all-metal transport airplane, in this country, dates back about thirty years. It was in the early thirties when the Douglas DC-2 and the Boeing 247D all-metal airplanes were put into intensive airline service. Accurate records back that far are difficult to come by, so we confined our examination to the last eleven years --- 1947 through 1958. We believe this will give a representative picture, since it took a few years after the war for the larger, faster airplanes, such as the DC-4, DC-6's and Constellations to acquire substantial flying time.

The CAB defines an accident as:

> "An occurrence that results in serious or fatal injury to one or more persons, or in substantial damage to any aircraft between the time an engine or engines are started for the purpose of commencing flight until the aircraft comes to rest with all engines stopped for complete or partial deplaning or unloading."

During the period 1948 through 1957, the CAB reported and/or investigated 815 accidents in air carrier operations, excluding Alaskan Carriers and Helicopter Operators. Forty-seven of these 815 accidents were classified as airframe-failure accidents, with any fatigue failures of course included. Of these 47 accidents, 11 involved small aircraft or were caused by conditions other than actual structural failures according to the CAB classification. However, there are 12 more which might be excluded as structural failures for purposes of this discussion on fatigue. These are: 5 cases of doors coming off, 1 case of ailerons connected in reverse, 2 cases of tank buckling due to vent trouble, 1 case where taps on T.E. of the elevator loosened, 1 case of porpoising on the ground, 1 case of jammed nose gear against doors and 1 case of damage from separated tire tread. This adds up to a total of 23 which might be excluded as structural failures, leaving a total of 24 (or 3%) which may be properly classified as airframe-failure accidents. Included in "airframe" however, are landing gear

failures, and of the 24 accidents mentioned above, 11 were due to landing gear causes which occurred on the ground and are usually of less serious consequences. We now have 13 accidents left (out of 815) which are classified as caused by failure of the airframe in flight. Eight of these can be deleted from this analysis, since it is very nearly certain that they were not caused by fatigue. This leaves a total of five, three caused by fatigue (of the airframe in flight) and 2 in doubt, since their causes were never determined. The three fatigue cases were one on a twin-engine transport in 1948 which lost its wing in flight, and 2 on a four-engine transport in 1952, both of which landed safely.

This brief analysis of the accident record might lead one to believe that airframe structural fatigue in airline operations is really not much of a safety problem after all. Actually, however, we all know that it could be a serious safety problem and, therefore, it demands constant attention. Why is the safety record so good when we all know that fatigue is a good-sized airframe maintenance problem? One reason is that, by and large, present critical structures are "fail safe," although it is only fairly recently that this term has become popular to non-engineers. The wing structure of our oldest airline workhorse, the DC-3, is extremely fail safe. We have been repairing fatigue cracks in the DC-3 wing attach doublers for many years, and no one would call the DC-3 wing structure unsafe. To be sure, there have been a couple of exceptions of non-fail safe structures, but once these were recognized, and corrective measures taken, the structures became perfectly safe. Although this fail-safe feature has contributed to the excellent safety record, it is, undoubtedly, secondary to the most important reason, and that is the exacting and comprehensive inspection procedures used by all the airlines. As is well known, each airline utilizes an elaborate system of inspection checks for all critical structure for all type airplanes operated, and, as previously stated, from their years of experience airline maintenance people have become experts in this field.

Methods of conducting aircraft maintenance have improved considerably over the years. The original practice was a major overhaul in the neighborhood of 8,000 to 10,000 hours, together with repeating short time inspections to insure continued airworthiness. The modern practice, which is usually referred to as progressive maintenance, consists of periodic minor and major checks, involving sampling to some extent, and which amounts to continuous overhaul. This system is greatly superior, safety-wise, to the major overhaul idea, since it permits more thorough inspection of more structural areas immediately after a new type airplane has been put into service, and it continues according to the cycles established during the life of the airplane. When a new type airplane is put into service, these continuous maintenance programs are developed for the whole airplane with the cooperation of the manufacturer, and after finalization, they are approved by the FAA. With regard to structural items in particular, the cycle times, the areas to be inspected and the details of the inspection, for all operators, are based on the fatigue testing program of the manufacturer, augmented by the expert knowledge of the manufacturers' and airlines' engineers.

The airlines have adopted modern methods of inspection, requiring a minimum amount of disassembly and designed to probe into the hard-to-get-at areas. An airline standing committee, composed of experts from large and small airlines alike, continually trades experiences on techniques and methods for probing into these hard-to-get-at areas that are difficult to inspect. This committee, which works with the manufacturers of airplanes and inspection equipment, was originally concerned primarily with X-ray methods and techniques, but recently has expanded its activity to include all types of so-called non-destructuve testing. By working together, the airlines have become expert in X-ray inspection in a much shorter period of time than otherwise would be possible. An airline which, by experimental methods, hits upon a successful X-ray technique for a particular part of the structure, transmits this information, through procedures developed by the committee, to other operators of the same type of equipment. By sharing experiences in this fashion, it becomes possible quickly to establish successful inspections in the hidden areas of new model airplanes. On one of the new jets, radiography manuals are already being developed in cooperation with the manufacturers. In this effort the operators work together, each taking on a certain percentage of the aircraft and reporting his findings and techniques developed.

Another activity, which contributes to the airlines success in avoiding serious trouble due to structural fatigue, is an industry reporting system commonly known within the industry as the DMR's or the Daily Mechanical Reports. These reports are required by Part 40.508 of the Civil Air Regulations which reads:

> "(a) Whenever a failure, malfunctioning, or other defect is detected in flight or on the ground in an airplane or airplane component which may reasonably be expected by the air carrier to cause a serious hazard in the operation of any airplane, a report shall be made of such failure, malfunctioning, or other defect to the Administrator. This report shall cover a 24-hour period beginning and ending at midnight, shall be submitted by 12 o'clock midnight of the following working day, or sooner if the seriousness of the malfunction or difficulty so warrants, and shall include as much of the following information as is available on the first daily report following such incidents:"

The "following information" refers to the ground rules. This system of reporting was established in 1948 and of course, covers all types of defects, along with items of structural fatigue. There are two salient elements in this reporting system which should not go unnoticed, and these are: (1) the words "which may reasonably be expected by the air carrier to cause a serious hazard, etc.," and (2) its daily character. With regard to the first element, this permits a one agency decision each day as to what should be reported. Without this feature, there could be considerable disagreement each day between the FAA field agents and airline personnel as to what items should be reported, with the result that the reports would very likely be cluttered up with items of useless or very doubtful value to other operators of the same equipment. This could increase their volume to the point of delaying the reports and seriously jeopardizing their "news" value. This "news" value is the second element mentioned above. Obviously, if

the system is to have value, operators of like equipment must be advised rapidly of each other's serious problems so that suitable corrective measures may be put into effect as soon as possible. Another element of this reporting system (implicit in 1 above) is the absence of specific definitions as to what items should or should not be reported, and to what extent. At first it was attempted to develop such definitions, but the task force assigned to this job could not find the place to draw the line and the effort was abandoned. Further attempts at definition have been made from time to time, but it was not until recently that any semblance of the objective was accomplished. Unofficial guide lines for the use of the airlines and the FAA field agents were established about the first of 1959 by mutual agreement between the airlines and the FAA. These cover all components of the airplanes but the one pertaining to structure reads as follows:

> "8. All serious failures, permanent deformation or corrosion of critical structure, including the pressure vessel. (must be reported)
> Comment: (also included) Due to the complexity and redundancy of aircraft structure, it is intended that common sense be used in reporting in accordance with this item. The intent is to report serious cracks, deformations, and corrosion in critical structure, and not report the many unimportant cracks, deformations and corrosion which occur routinely as airplanes age or which are covered by manufacturers' instructions. With respect to the new high altitude jets it should be remembered that fail safe and safe life design concepts do not preclude cracks due to fatigue. Until adequate service experience is accumulated on the new jets it is recommended that all cracks in the primary structure including the pressure vessel be reported under this item."

Obviously, the reporting of all cracks as the airplanes gain service time would needlessly clutter up any reporting system, and further, there would be no useful purpose served in continually reporting serious items that are already included in the airlines' inspection programs and covered by the manufacturers' service bulletins. It is interesting to note what the system produces with respect to fatigue items. During the three years of 1954, 1955 and 1956 (in all probability a representative period) this system produced 3785 reports covering all components of the airplane. Of these, approximately 325 or 8% indicated that fatigue was the predominant factor involved. These items covered airframe and landing gear but excluded powerplant failures. Of the individual items, landing gears, including wheels, were the worst offenders, contributing 25% of the 325 items. Next came wing spars with 13% of the total. The lesser important stringers and frames combined contributed 19%. Attach fittings accounted for 7.5%. In summary, the wings accounted for 40%, fuselages 25%, landing gear 25%, empennage 57%, and miscellaneous 5%. These proportions appear to be what one would expect when it is recalled that, under the rules of the reporting system, only defects which might be serious hazards need to be reported. However, these same proportions would not necessarily apply to the

total maintenance workload resulting from the total number of defects which would include the non-serious as well as the serious.

Of course the extra assurance of safety provided by an industry-wise reporting system of this kind can only be estimated. However, it appears to have considerable value --- at least it has withstood the test of time and to date nobody in industry or government has been able to come forth with an alternate system of industry reporting which promises to do as well.

The incidence of fatigue cracks in airframe structures represents costly headaches to the airlines. Obviously, inspection procedures must be thorough, and these consume a large number of man hours. In addition, the repair bills can be tremendous, especially after the airplanes become older. Reskinning of lower wing structures can become million-dollar items for a fleet of airplanes. A good chunk of the cost of heavy maintenance checks is attributed to "repairing the metal." In addition, the more serious defects result in campaign repair programs which are extremely expensive because they may remove airplanes from service temporarily and can seriously disrupt the preplanned man hour and floor space programs in the maintenance shops. Involved in these programs are the mandatory directives issued by the FAA. Naturally, these cover the most serious items and are almost always accelerated programs, and many of them are concerned with fatigue type failures.

In keeping with progress in this field of activity, the Civil Air Regulations, as they pertain to fatigue, were modernized about three or four years ago. The regulations now require that the structure for a new design be evaluated on the basis of "fatigue strength" or "fail-safe strength." The "fatigue strength" concept states that the structure shall be shown to be capable of withstanding the repeated loads of variable magnitude expected in service. The "fail-safe strength" concept states that it shall be shown that catastrophic failure or excessive structural deformation, which could adversely affect the flight characteristics of the airplane, are not probable after fatigue failure, or obvious partial failure, of a single principal structural element. After such failure, the remaining structure shall be capable of withstanding certain static loads corresponding with certain flight loading conditions which are specified. In other words, a "fail-safe" structure is one that can continue to carry all or part of its design load even though a failure has occurred. Of course, all the new turbine-powered airplanes are designed in accordance with these concepts, but it should be emphasized that they are not at all new and have been used by engineers since fatigue problems began to be understood.

During the past years, the manufacturers have given more and more attention to the fatigue design problem, and there is every reason to believe that the new turbine-powered airplanes will be better than ever in this respect. Extensive fatigue testing has been done on many of the complete sections and joints of the primary structure. Also, elaborate and expensive test programs have been

conducted to assure the integrity of the airplane cabins in case of a sudden decompression caused by the failure of a particular section or part of the structure. As a result of this ever-increasing attention to fatigue, it is reasonable to expect, and it is fondly hoped by the industry, that the incidence of fatigue cracks will decrease proportionally on the new aircraft in the years to come.

AIRCRAFT INDUSTRY PHILOSOPHY ON STRUCTURAL FATIGUE

By

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For you to properly understand and appreciate the industry philosophy on structural fatigue, review is desirable of some of the background and history which have led to the development of this philosophy. Further, I will briefly touch on the recent industry activities to indicate the scope of the effort on the fatigue problem and attempt to assess the relative significance of several of the factors which contribute in a substantial way to the fatigue experience with an aircraft.

The fatigue problem is not new to the aircraft industry, but its emergence as a critical problem in aircraft design has only been within the past decade or so. Figure 1 gives a chronology of some of the fatigue incidents and accidents in the past decade. The problem has become increasingly critical in recent years for several reasons. A greater utilization of both commercial and military aircraft combined with the newer materials and configurations with increased static structural efficiency, plus changes in the applications of the aircraft, have in many cases resulted in the development of fatigue problems in service. The long useful calendar life of modern military airplanes often results in adaptation to new tactical requirements transcending their original design objectives. All segments of the industry have a part to play in resolving this problem. The manufacturing segment is well aware of the varied and complex factors affecting fatigue performance. However, there is no reason to believe that this problem will be any greater than many equally complex problems encountered and solved in the past. The technical tools to cope with this problem, testing facilities and trained personnel are generally available. Great strides have been made in recent years and are already reflected in the designs of today.

Testing is at present the designer's best tool in analyzing fatigue performance. This testing can be broadly divided into three categories: (1) basic materials tests, (2) component tests, and (3) full scale or "complete" airplane tests. Basic materials tests on smooth and notched specimens provide comparative data on the service potential of various types of material and alloys. Tests on small components representative of airplane joints or structural details can be done early in the design stages and thus can greatly influence the detail design. These small component tests are relatively inexpensive compared with full scale tests. Thus more specimens of a particular detail can be tested, including progressive improvements determined from earlier test results, thereby reducing statistical uncertainties and improving the design. Full scale tests are usually too late to influence detail design, although they can result in production modifications when critical fatigue areas are highlighted. Improvements of a less critical nature can also be programmed into later production airplanes of a long production series, but can seldom be economically justified on a retrofit basis fleet-wise. Probably the two most important technical benefits derived from full scale tests are the more exact stress distributions and the assurance that no fatigue critical areas have been

overlooked during the component development testing and analysis. Despite the limited number of specimens in full scale testing, for obvious economical reasons, much information useful in setting up inspection and maintenance schedules and methods of repair is obtained. In addition, this type of testing sometimes allows a better study of crack propagation and resulting load redistribution in statically indeterminate structures after the start of a fatigue crack.

The general complexity of the fatigue problem and the high dependence of fatigue experience on factors outside the direct control of the designer make it impractical to attempt to guarantee fatigue life of aircraft structures. The factors influencing fatigue performance might well be divided into two broad categories: those which determine the structural capability of the airplane and are substantially directly controlled by the manufacturer, and those which result from operational use over which the manufacturer has little control. These two broad categories are here subdivided to illustrate their relative importance and the potential variability as they affect fatigue performance.

By a very limited presentation of data on each element I will attempt to establish a very approximate variability factor for each significant element of the fatigue problem in order that they can be examined in relation to each other on a relatively quantitative basis. I am sure each structures engineer would establish a somewhat different number for each of these factors but the differences would probably not affect the major conclusions I wish to emphasize.

Considering first those elements of the fatigue problem affecting structural capability of the airplane, there are such things as overall vehicle configuration and detail design. Detail design encompasses the choosing of materials, the detail configuration, processes and manufacturing techniques. Herein lies the greatest variability of fatigue performance. Fortunately, it is the element over which the designer has greatest control and is most susceptible to early laboratory evaluation. Materials show variations due to grain structure, chemistry, heat treatment, etc. Tests show such subtle effects that significant variations exist between lots of a particular material and between specimens within the same lot, as well as the basic differences between materials.

Figure 2 is a bar chart showing the comparative results of one series of fatigue tests on typical structural joints of two high strength aluminum alloys. The ratio of the life of alloy X to that of alloy Y is shown for various stress levels. Excluding the lower stress levels, an endurance ratio of 2 could be assigned to materials on the basis of this data. Detail structural configuration is by far the most important detail design consideration. The same test series utilized for material comparison in Figure 2 included joints of six different configurations. Figure 3 shows the ratio of the lives of the highest and lowest life configurations for one of the alloys. Again, excluding the low stress level data, an endurance ratio of 18 might be considered representative for detail configuration.

Turning to processes and manufacturing techniques, the potential variations are almost infinite. For example, proper methods of working or heat treating materials can induce internal stresses that are beneficial to fatigue performance or prevent internal stresses that may be detrimental. Internal or residual stresses can also be introduced by the build-up of tolerances in complex structures. These stresses can be controlled to a significant extent by proper design and manufacturing techniques. Control of corrosion by finishes or corrosion inhibitors also falls in the category of processes and manufacturing techniques.

Selecting only one aspect of the materials problem for illustration, Figure 4 shows results of a test program designed to determine the effect of material condition at one stage in processing on its later resistance to crack propagation. Data are shown as crack growth versus cycles of alternating load for the same material subjected to two different processes. In one instance the material was prestretched 6% prior to introducing the starting crack. In the other instance there was no prestretching. It is seen that the prestretching decreased the number of cycles to what might be considered a critical crack length by about a factor of 4. This factor of 4 has been arbitrarily chosen as a conservative estimate of the potential variation in fatigue performance due to materials and processing.

The other factor mentioned as under the designer's control is general vehicle configuration. Load experience due to gusts and thus fatigue performance is affected by sweep angle, aeroelastic characteristics, wing loading, aspect ratio, as well as disbursement and management of fuel. Our studies have indicated that an approximate endurance factor of 4 could be presumed for these effects.

I have purposely omitted the general stress level as a separate element because reduction of general stress level is a poor approach to solution of fatigue problems. Fatigue is a local problem and is therefore most responsive to local attack. In addition, test data shows that a general reduction of stress level by 50 percent will usually result in a life improvement of only 5 or 10 to 1. Doubling the structural weight to achieve such modest gains is obviously impractical, and the gains from lesser general stress level changes are not large enough to provide a solution to the problem.

Turning now to those items over which the designer has little control; these can be further subdivided also. Operational use as to geographic location, the type of mission assignment, whether short or long flights, dominantly high altitude or low altitude flying, the severity of the maneuvering, etc., are all very significant to the ultimate fatigue performance of a design of any given structural capability. Load experience due to gusts varies considerably, as can be seen from Figure 5. However, since the gust experience is only a part of the load experience of the airplane, it is not possible to directly read an endurance factor from this figure. These curves were obtained from published VG and VGH data. Geographical assignment can also significantly affect the gust and corrosion environment experienced by the aircraft.

Figure 6 gives an indication of the effect of flight length on fatigue damage. Note that the fatigue damage accumulation seems to be more directly a function of flights rather than hours. The curves shown are representative of wing structure on a commercial airplane.

Even the sequence of loadings experienced by an airplane can affect fatigue performance. An example of load sequence effect is shown on Figure 7. This data is from some programmed fatigue testing recently reported from the Netherlands (2). Each spectrum contains the same total number of peak load experiences, but the relative test life varies greatly depending on the order of load application. This broad variation would be expected to decrease at increased stress levels.

The elements affecting fatigue performance selected for discussion here are summarized on Figure 8. The previously selected endurance ratios are shown in conjunction with the various elements. As mentioned previously, they are subject to considerable debate, but the point to be made here is that through design and processing control very significant improvements in fatigue performance are

potentially available. Furthermore, even given a particular structure with its inherent potential fatigue performance, a broad variation in the resulting performance is possible due to factors beyond the control of the manufacturer.

Turning now to some design concepts that have developed, it is unfortunate that a sharp line of distinction has emerged between two apparently contradictory philosophies. In reality the philosophies are not entirely contradictory and a successful structure generally incorporates a substantial element of each philosophy even though the designer may choose one or the other basis to demonstrate its acceptability. The two philosophies are the fail safe and the safe life design concepts. The safe life concept in its purest form visualizes a structure designed to such a high level of fatigue performance that it can be economically replaced as this life is reached, whether or not actual evidence of critical fatigue damage is apparent. Accordingly then, no specific tolerance for partial failure or cracking is required. The fail safe concept in its purest form implies design to tolerate a large degree of fatigue damage to assure a level of safety such that the problem is converted from a safety problem to one of economics. No specific life to first crack is required as the crack will be caught by inspection before reaching catastrophic proportions. On closer examination, each concept is really somewhat dependent on the other.

I believe it is apparent, from the earlier remarks on the tremendous potential variability in fatigue performance, that absolute dependence on the safe life philosophy requires a very conservative selection of structural replacement time if no tolerance to fatigue potential failure exists, if a satisfactory level of safety is to obtain. To achieve safety then, substantially premature replacement is necessary. Reliance on the fail safe concept, on the other hand, permits the use of inspection to control the fatigue damage within non-catastrophic bounds, and thereby utilizes more nearly the full fatigue potential of the structure. To be economically feasible, a substantial useful life must exist.

I have attempted to indicate, in Figure 9, my general feelings regarding the structural acceptability in terms of the degree of mixing of the two philosophies. The ordinate is intended to indicate the degree of safety and economy, and the abscissa the degree of mixing of the two philosophies. As long as there is a substantial element of both philosophies in the design, no matter which philosophy is dominant, a high degree of safety and economy of operation should be attainable. Either philosophy in its pure form can result in serious compromise of safety and economy.

Turning now to some of the activities in the industry to meet the challenge of fatigue, these are occurring on a broad front of both individual and collective action. Collectively, the Aerospace Industries Association serves as both a spokesman and industry clearing house of opinion for the whole range of industrial problems from management to technical phases. Figure 10 shows an abbreviated organizational chart designed to give an indication of the area in which fatigue problems are considered. The Association is ruled by a Board of Governors composed of executives from member companies. Of the various services performed by the AIA, the technical services, and specifically the Aircraft Technical Committee, are responsible for engineering phases. Of the specialized groups supporting the ATC, the three groups most directly interested in the fatigue problem are:

(1) ARC - The Airworthiness Requirements Committee. This committee considers problems associated with airworthiness and design specifications.

(2) ARTC - The Aircraft Research and Testing Committee. In general this committee concerns itself with industry problems in research, testing, and dynamics in aircraft and missiles. Such projects as standards for testing and advisory work on materials or design practices are part of this committee's work. Technical review of specifications for the ARC is often accomplished also.

(3) NASC - The National Aircraft Standards Committee. This committee has interests concerned with standardization and specification problems related to aircraft and guided missile parts, systems, installations, components, materials and processes.

Panels or projects are created to review certain aspects of fatigue as well as other problems. The W-76 Fatigue Panel, created to provide a summary of the status of fatigue knowledge, is an example. As illustrated in Figure 11, this panel studied the areas of environment, design and analysis procedures, material fatigue properties, and operational control of fatigue damage. Much of the data collected or developed by this panel is contained in the three volumes of the AIA fatigue handbook. Various other panels and projects are in existence or will be formed from the technical specialists in the industry when the need arises.

In addition to the AIA activities there is continuous informal liaison between the manufacturers, even internationally. Various other societies, such as the Institute of The Aerospace Sciences, American Society for Testing Materials, the Society of Automotive Engineers, to name a few, provide an interchange of ideas through meetings and published data. There is also continuous exchange of ideas with qualified members of the university, college and research institute technical staffs. There has been a trend by industry to utilize the full time services of university or college professional staffs for summer employment as well as in consulting capacities. While not of immediate gain, this provides a better understanding of industry problems to a research and educational area of technical society, hence long-range gains can be expected through knowledge of these problems being passed on to the future technical personnel.

Some mention of the scope of testing facilities within the industry appears appropriate. The following photographs give a brief pictorial review of some of the facilities and activities of the Boeing Airplane Company as a representative manufacturer. Figure 12 shows a portion of the Sonic Laboratory in which acoustical fatigue tests on structural panels are made. Figure 13 shows testing of a control system bracket installation in the Vibration Laboratory. A special area in our new Developmental Center has been set aside for fatigue testing. A portion of this area showing a 250,000 pound capacity fatigue machine with several smaller SF-10-U machines in the background is seen in Figure 14. Figure 15 shows a larger 500,000 pound capacity machine developed primarily for testing the dog-leg wing joints at the wing-body intersection of swept wing airplanes or other structures requiring loads in more than one plane. A full scale fatigue test of the B-52 airplane is now in progress. Figure 16 is a photograph taken during the early stages in this testing and shows some of the loading structure used. Figure 17 is a composite photo taken during the cycling of the wing to more closely illustrate the motions involved. A model KC-135 fuselage has been tested in the hydrostatic fatigue facility, as shown in Figure 18.

Summarizing the industry philosophy or attitude towards structural fatigue, it might be stated approximately this way:

(1) Although fatigue is a complex phenomenon, we believe we are capable of solving the problems involved.

(2) Because many of the factors of great significance in the final fatigue performance of a structure are beyond the manufacturer's control, industry cannot guarantee specified lives with absolutely no fatigue failures.

(3) Of the many factors affecting fatigue performance, detail design and processing display the greatest variability and fortunately are most available for control by the designer.

(4) Testing is an essential design tool in determining fatigue performance.

(5) There should be no sharp line of distinction between the safe-life and fail-safe concepts. To realize the greatest safety and economy, the design should incorporate consideration of both concepts.

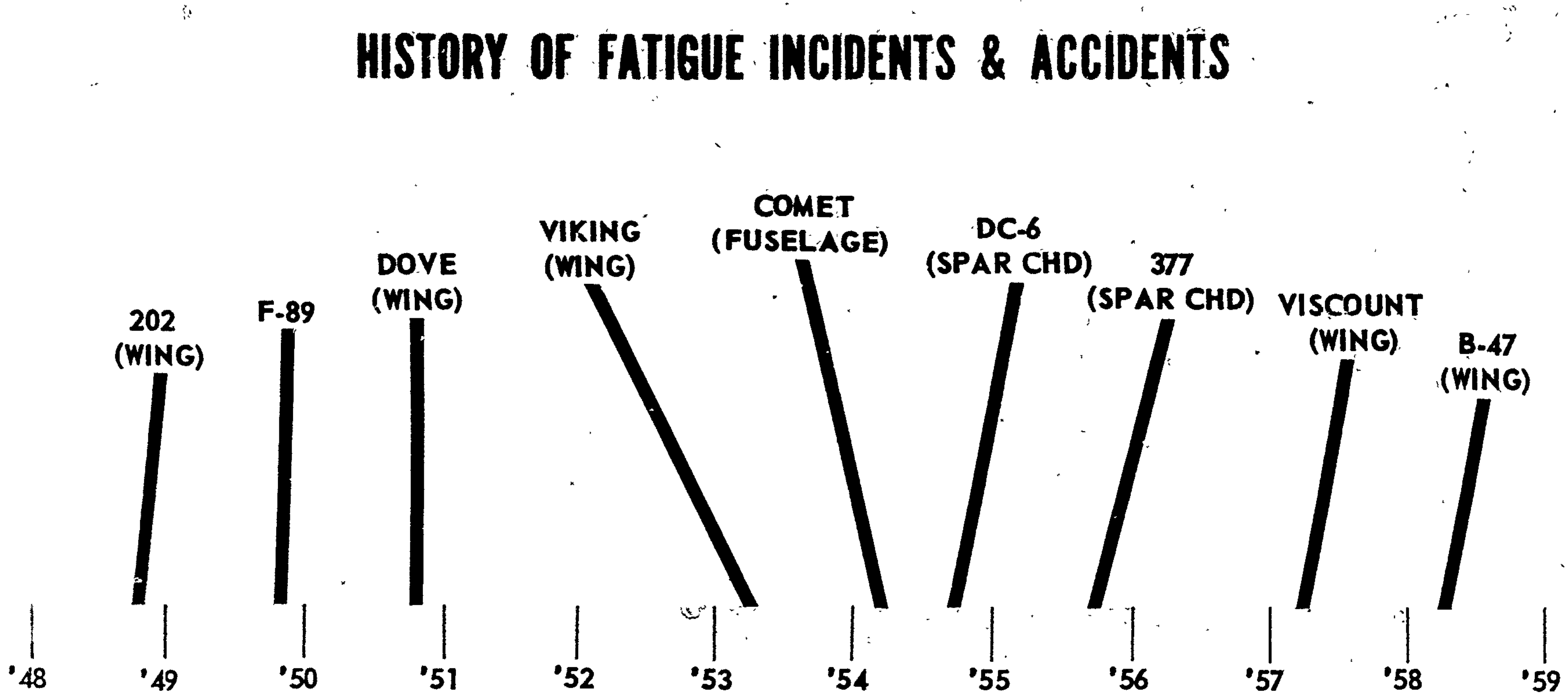
(6) Because of certain basic conflicts between high static structural efficiency and long fatigue life, costs of achieving a satisfactory level of each in combination will be higher than achieving either alone. Continued emphasis on basic understanding of the fatigue behavior of structures and materials may lead to a technological breakthrough and should be continued to advance our knowledge. However, until that time, the present state of the art permits attainment of the objective, albeit at some increased cost.

Some of my colleagues in reading this paper felt that it might convey an attitude of complacency. If such is the case, I have done the industry a disservice. I had hoped to convey cautious optimism for the future and the feeling that the solutions are within our grasp if we set realistic goals and pursue the attack.

REFERENCES

1. NACA TN3269 "Additional Static and Fatigue Tests of High-Strength Aluminum-Alloy Bolted Joints," by E. C. Hartmann, Marshall Holt and I. D. Eaton, July 1954.

2. National Luchtvaartlaboratorium Report MP.178 "The Endurance Under Program-Fatigue Testing," by J. Schijve, National Aeronautical Research Institute (N.L.L.), Amsterdam, May 1959.


HISTORY OF FATIGUE INCIDENTS & ACCIDENTS
202 (WING)
F-89
DOVE (WING)
VIKING (WING)
COMET (FUSELAGE)
DC-6 (SPAR CHD)
377 (SPAR CHD)
VISCOUNT (WING)
B-47 (WING)
'48
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'54
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Figure 1

MATERIAL FATIGUE VARIABILITY

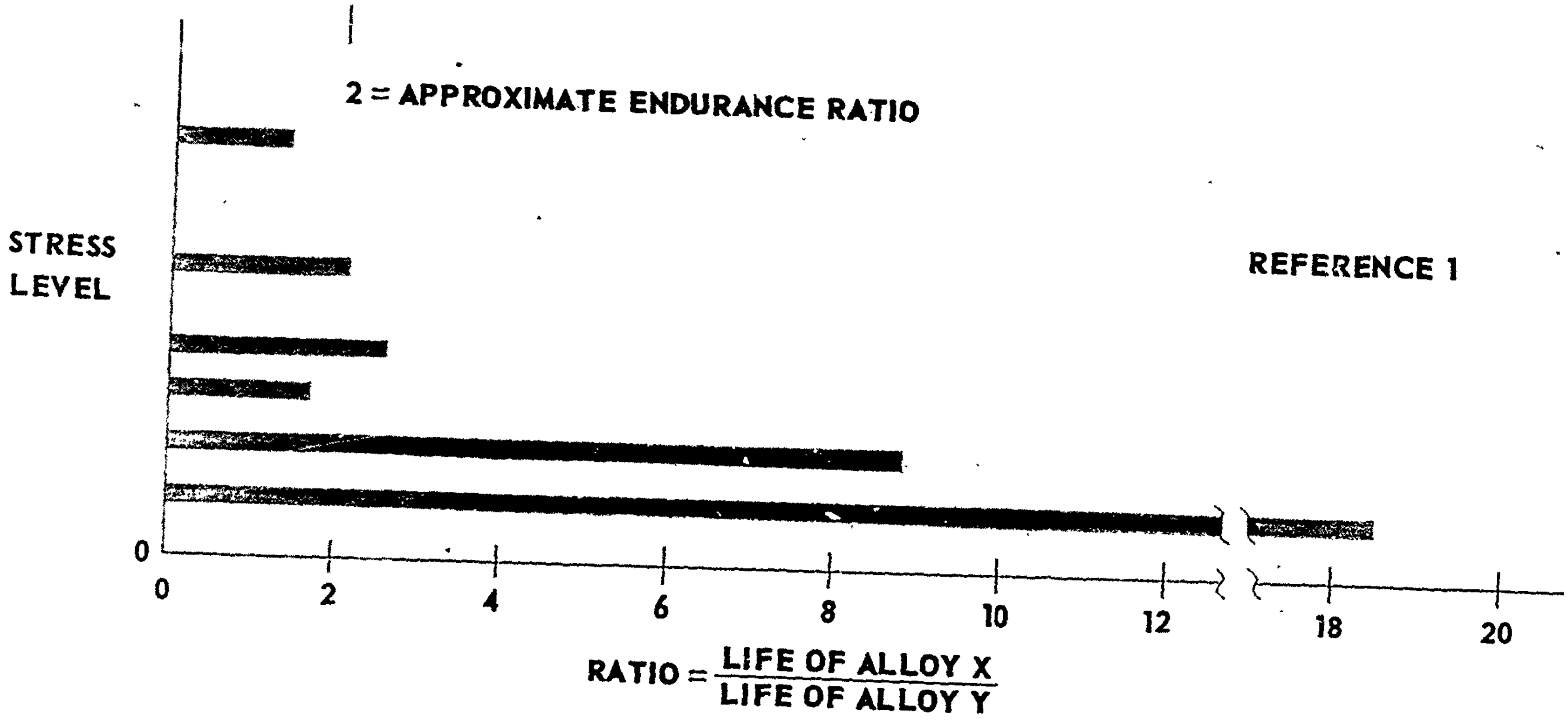


Figure 2

DETAIL DESIGN FATIGUE VARIABILITY

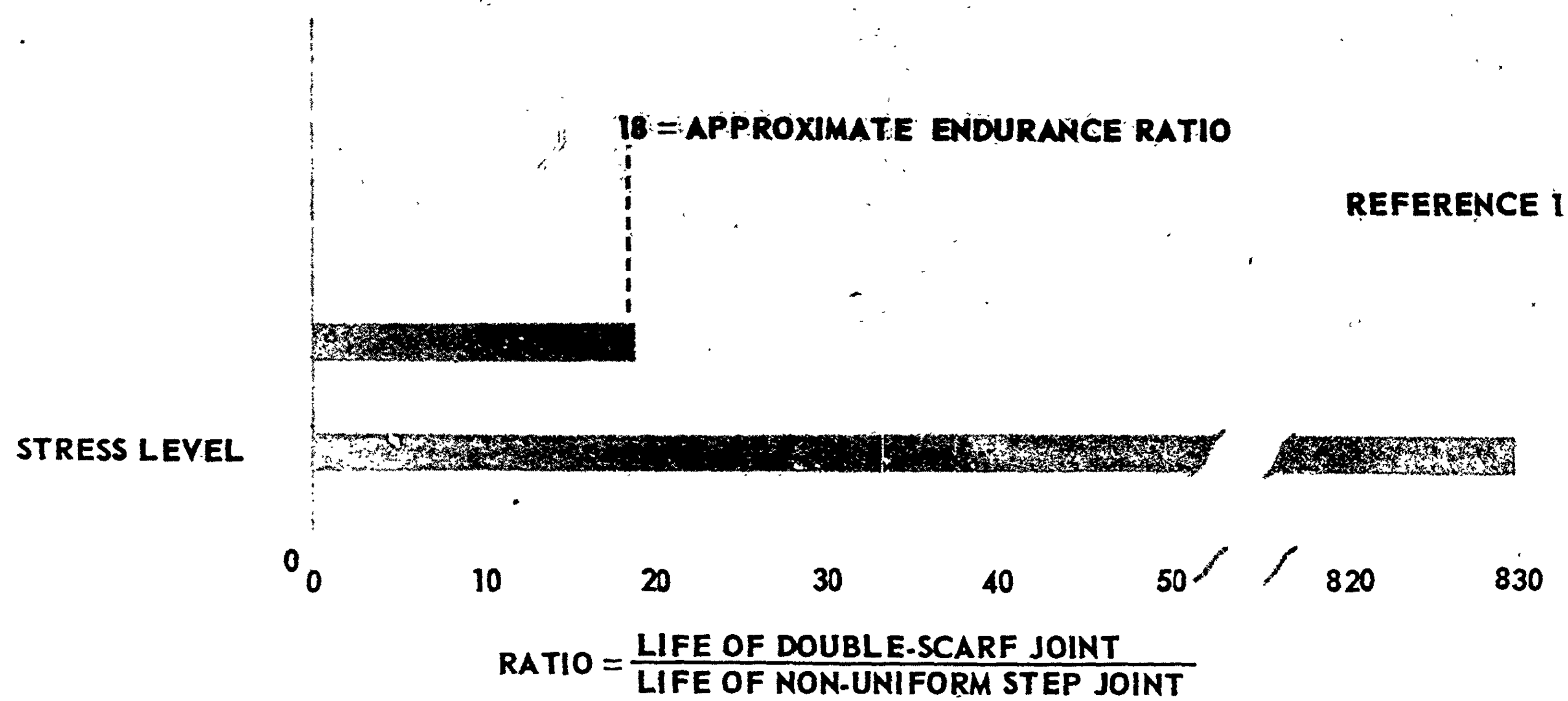


Figure 3

MATERIAL CONDITION EFFECT ON CRACK GROWTH

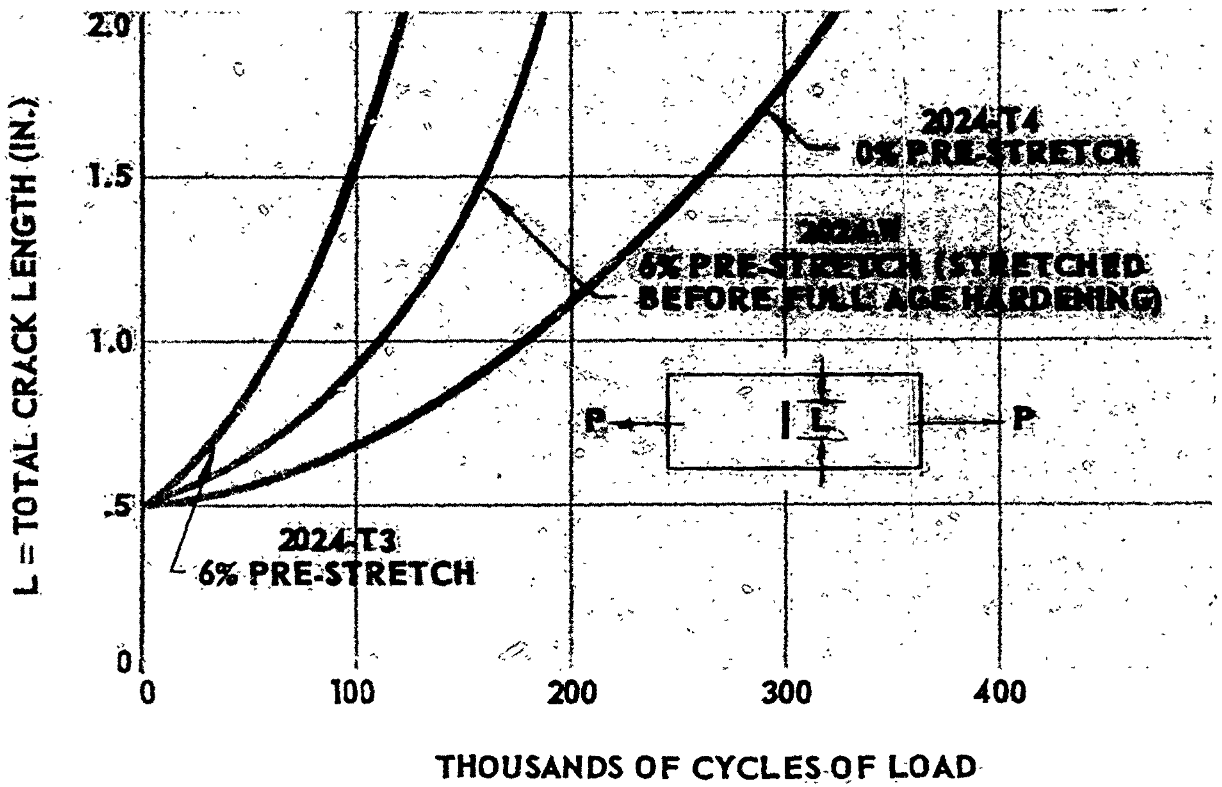


Figure 4

LOAD ENVIRONMENT VARIABILITY

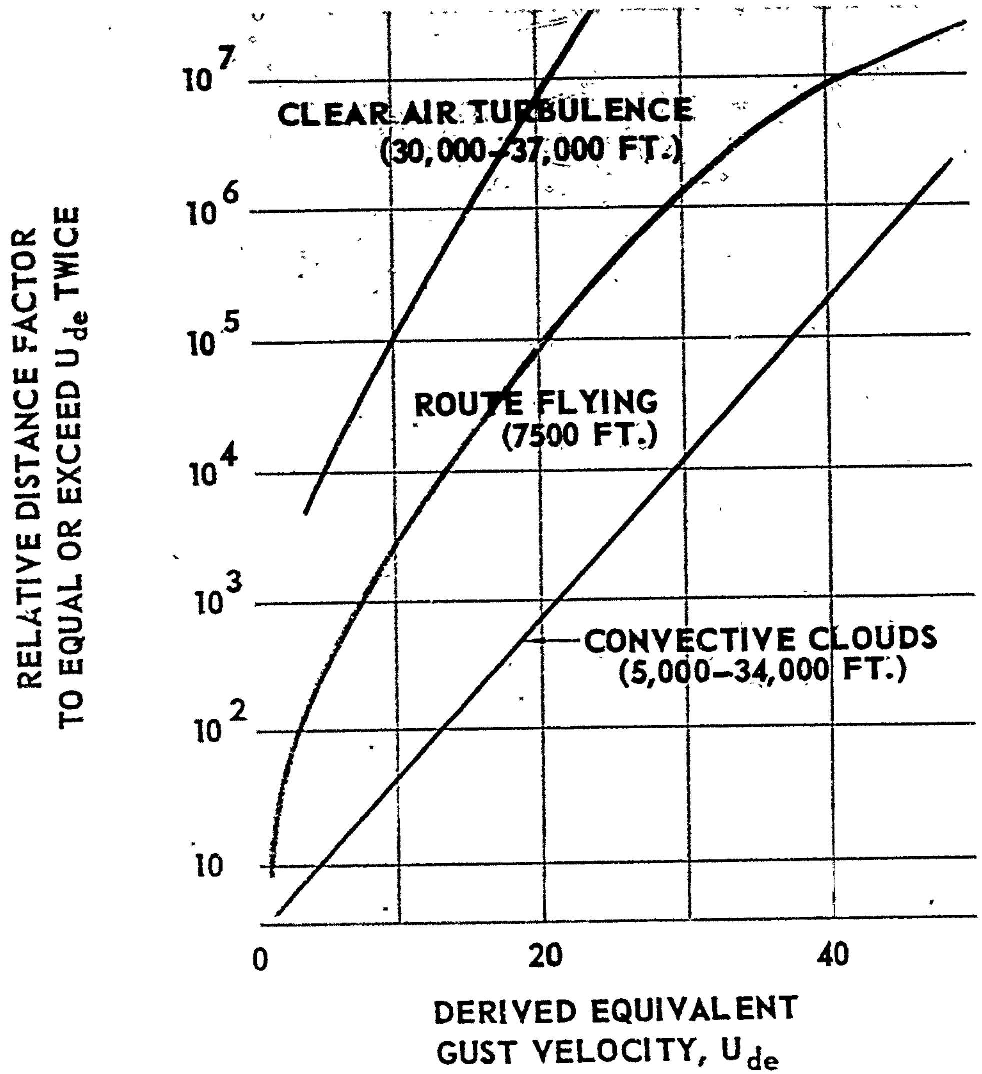


Figure 5

FATIGUE PERFORMANCE VS. FLIGHT LENGTH

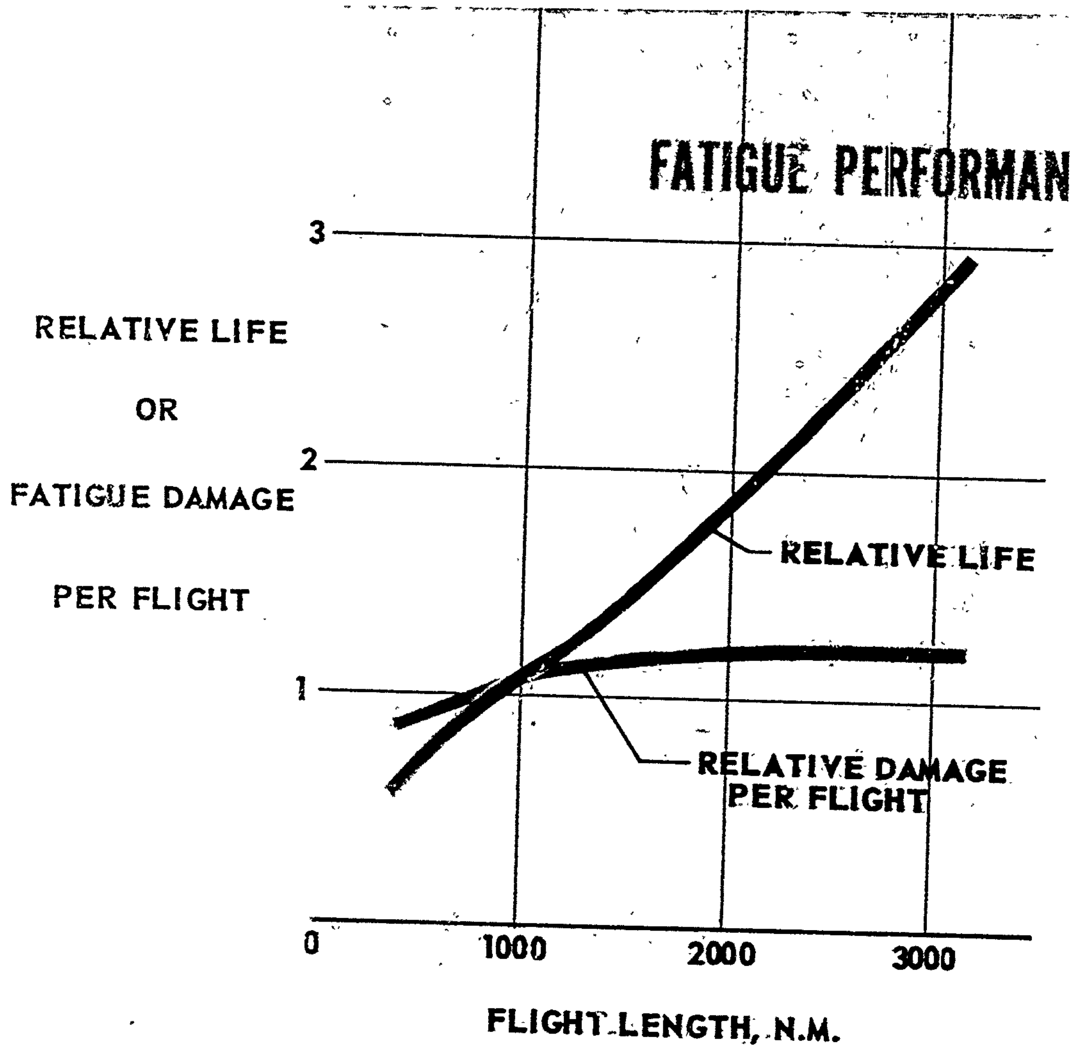
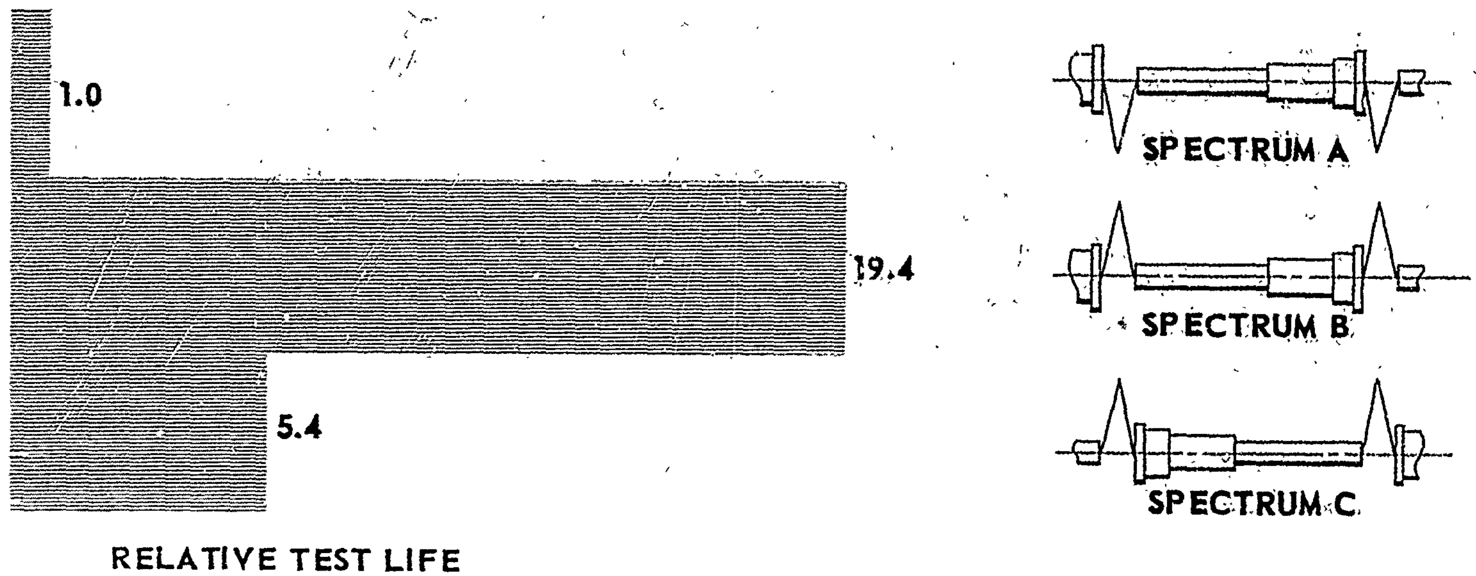


Figure 6

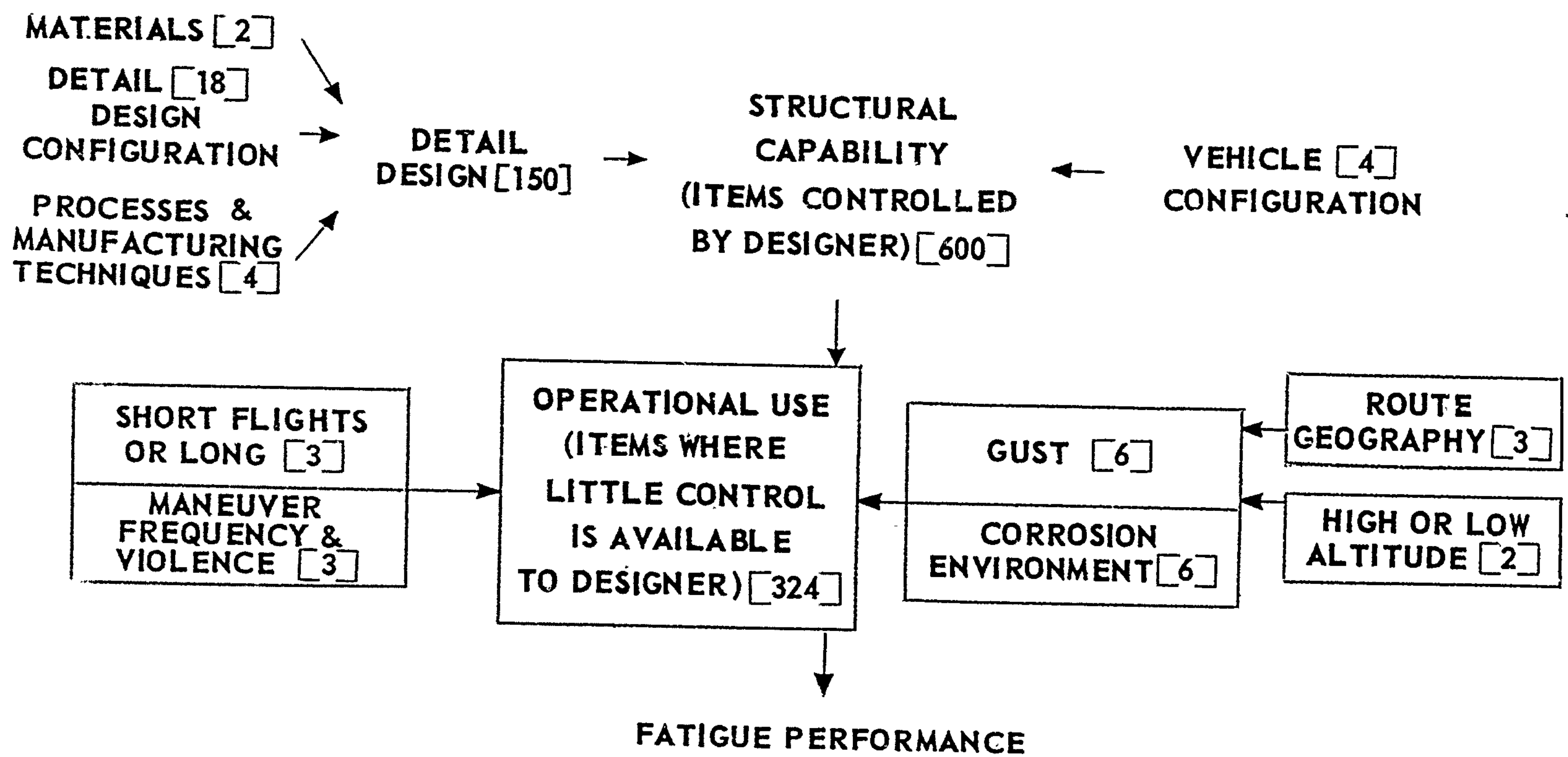
FATIGUE PERFORMANCE UNDER PROGRAMMED LOADS



REFERENCE 2

Figure 7

ITEMS AFFECTING FATIGUE PERFORMANCE



[ ] BRACKETED NUMBERS INDICATE ASSIGNED ENDURANCE RATIOS

Figure 8

SAFE-LIFE AND FAIL SAFE DESIGN

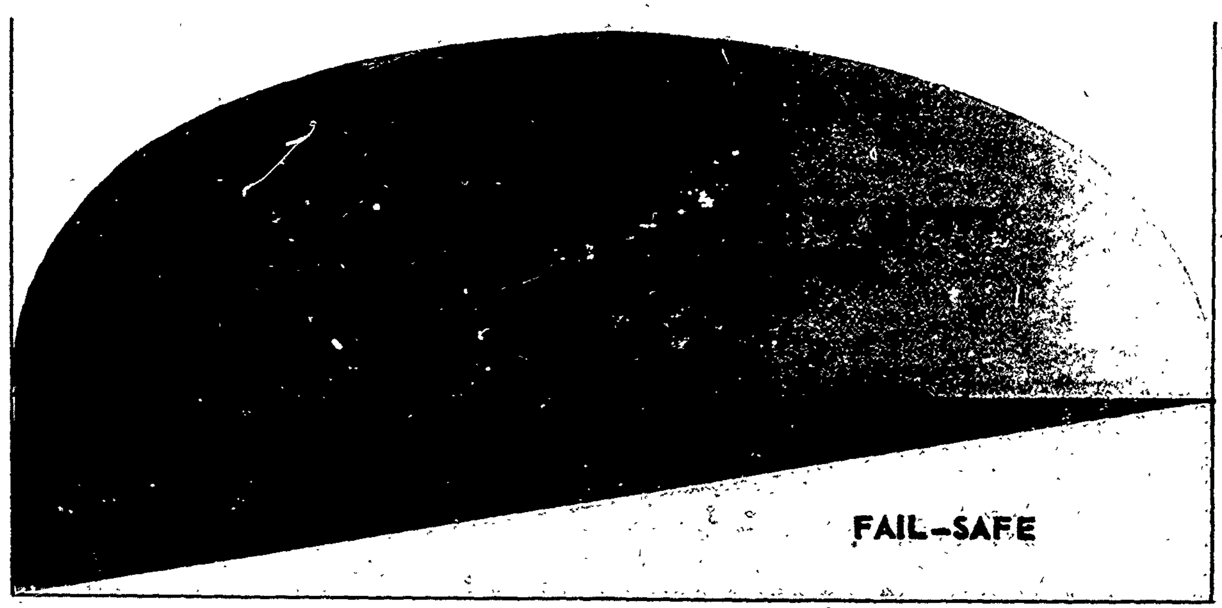


Figure 9

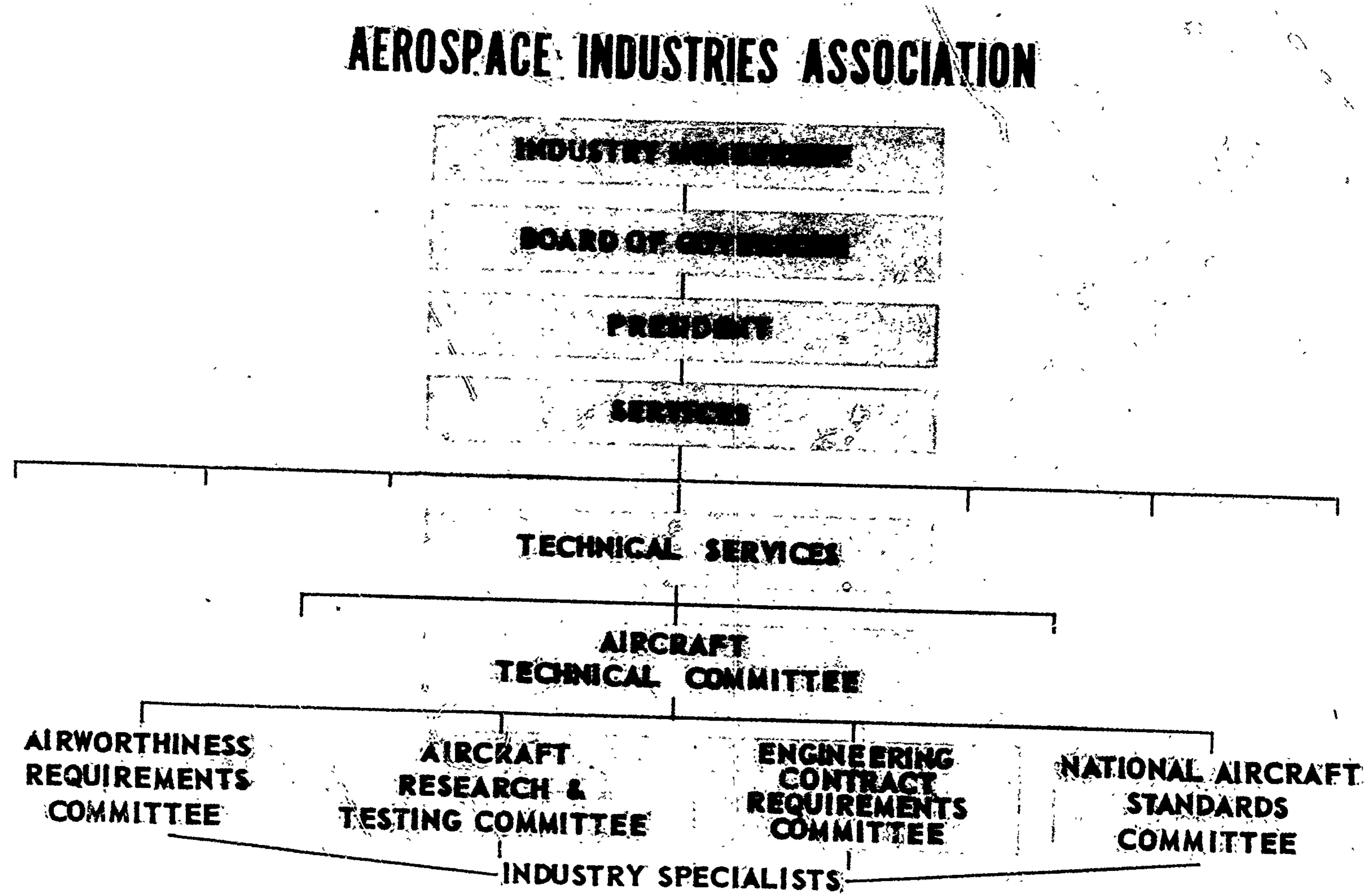

AEROSPACE INDUSTRIES ASSOCIATION
INDUSTRY
BOARD OF
TECHNICAL SERVICES
AIRCRAFT TECHNICAL COMMITTEE
AIRWORTHINESS REQUIREMENTS COMMITTEE
AIRCRAFT RESEARCH & TESTING COMMITTEE
ENGINEERING CONTRACT REQUIREMENTS COMMITTEE
NATIONAL AIRCRAFT STANDARDS COMMITTEE
INDUSTRY SPECIALISTS

Figure 10

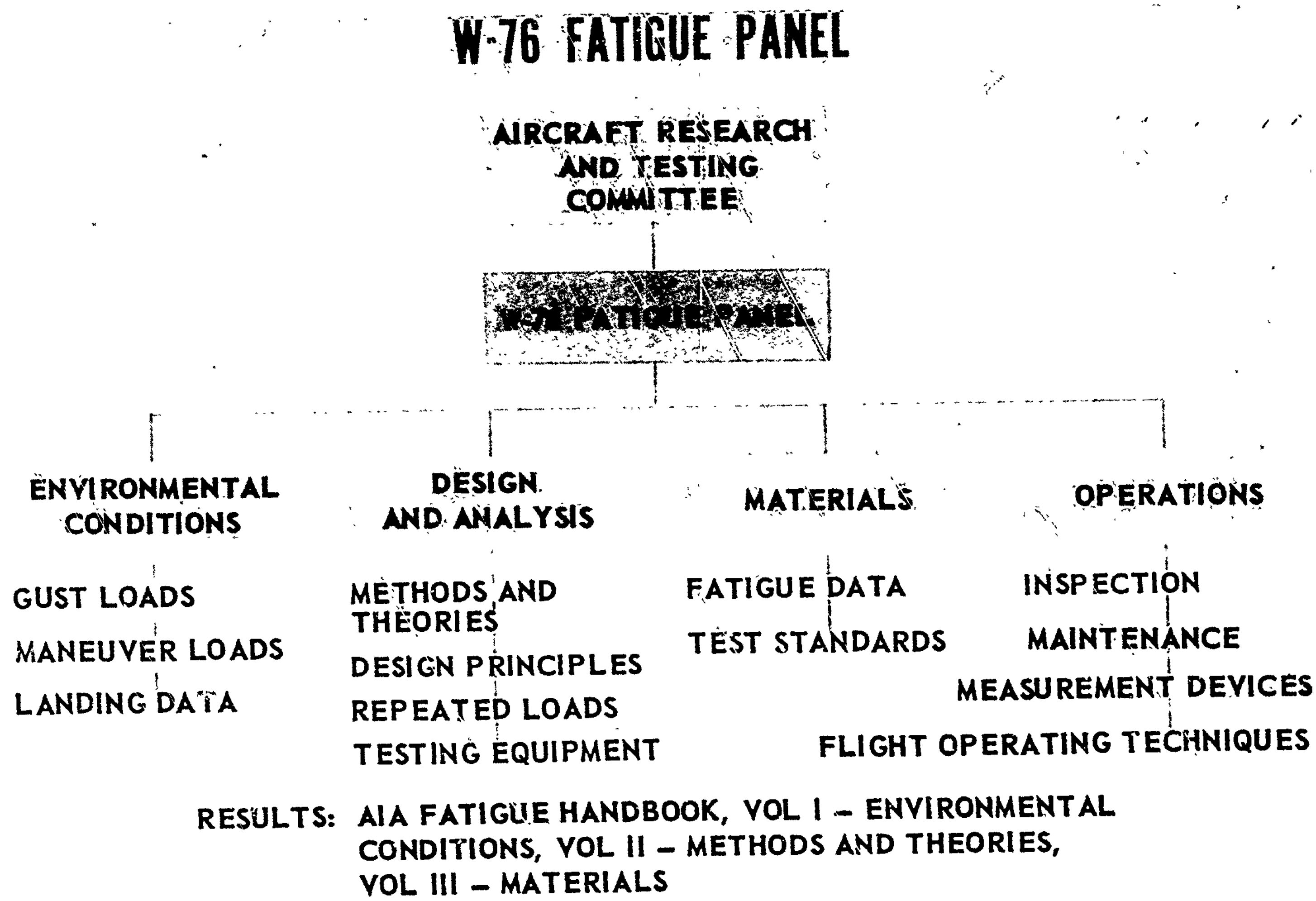

W-76 FATIGUE PANEL
AIRCRAFT RESEARCH AND TESTING COMMITTEE
W-76 FATIGUE PANEL
ENVIRONMENTAL CONDITIONS
GUST LOADS
MANEUVER LOADS
LANDING DATA
DESIGN AND ANALYSIS
METHODS AND THEORIES
DESIGN PRINCIPLES
REPEATED LOADS
TESTING EQUIPMENT
MATERIALS
FATIGUE DATA
TEST STANDARDS
OPERATIONS
INSPECTION
MAINTENANCE
MEASUREMENT DEVICES
FLIGHT OPERATING TECHNIQUES
RESULTS: AIA FATIGUE HANDBOOK, VOL I – ENVIRONMENTAL CONDITIONS, VOL II – METHODS AND THEORIES, VOL III – MATERIALS

Figure 11

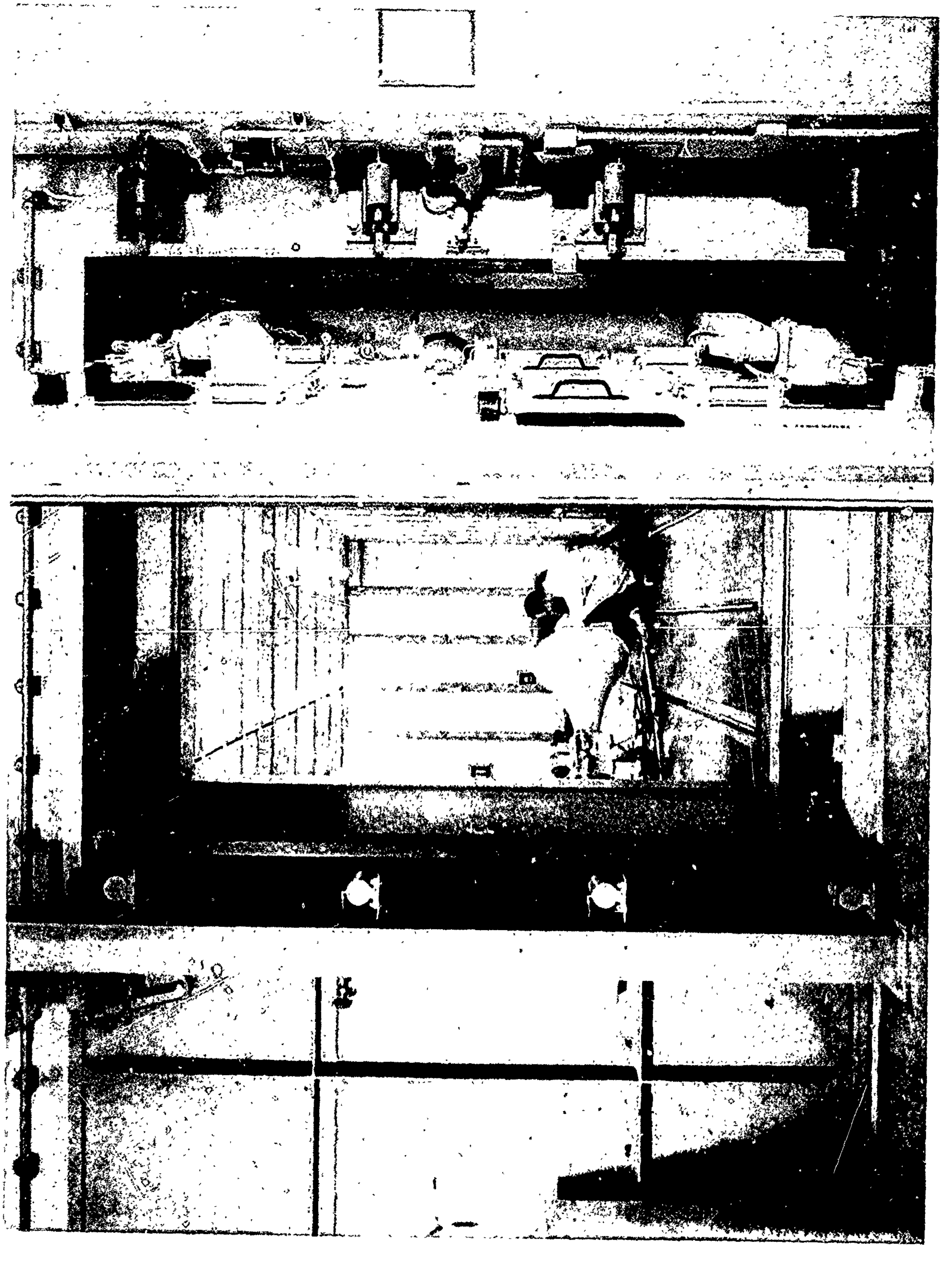


Figure 12 - Sonic Lab Showing Horn Test Room

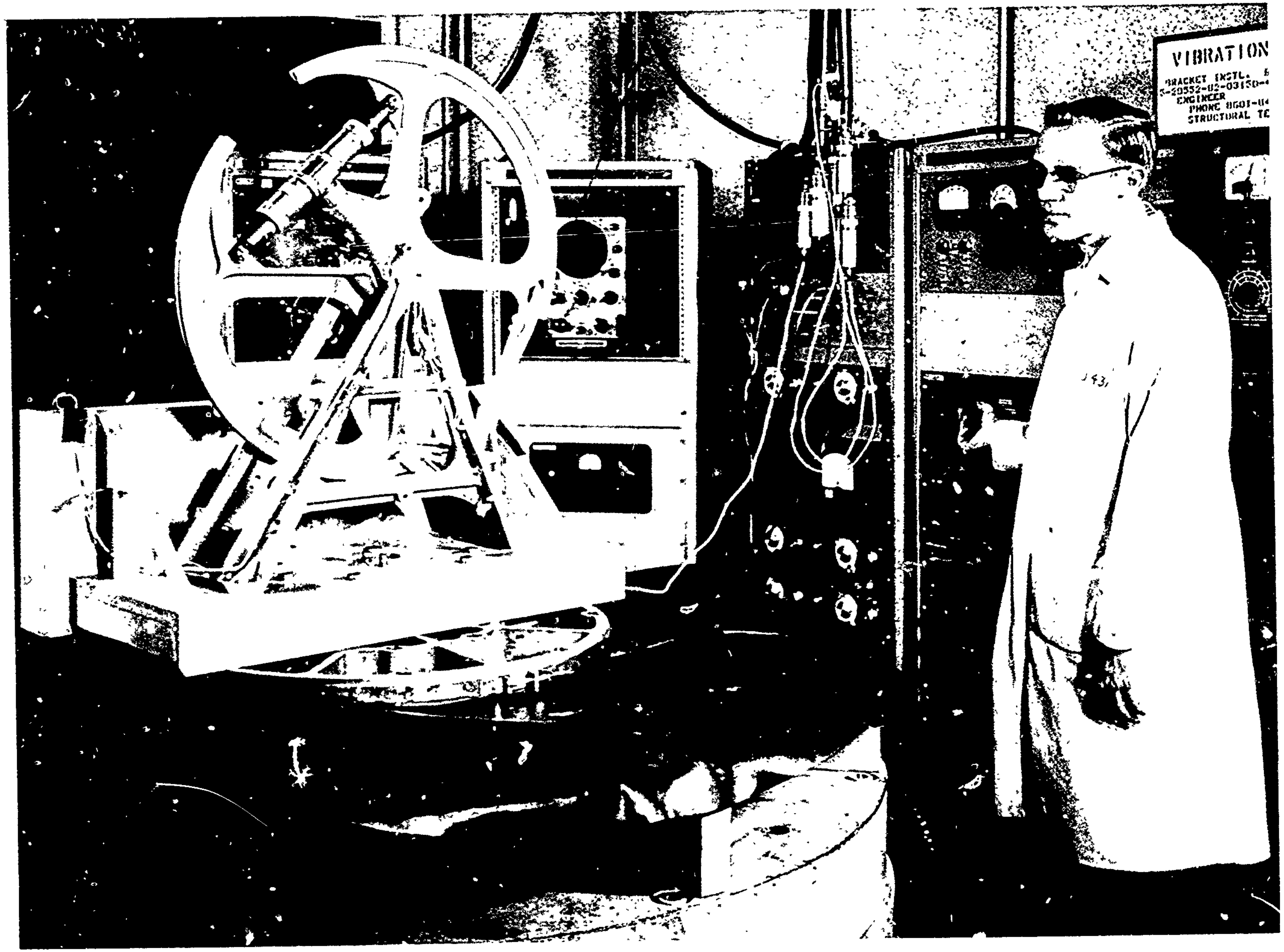


Figure 13 - Vibration Lab Showing Test Of Control Syst m Bracket Installation

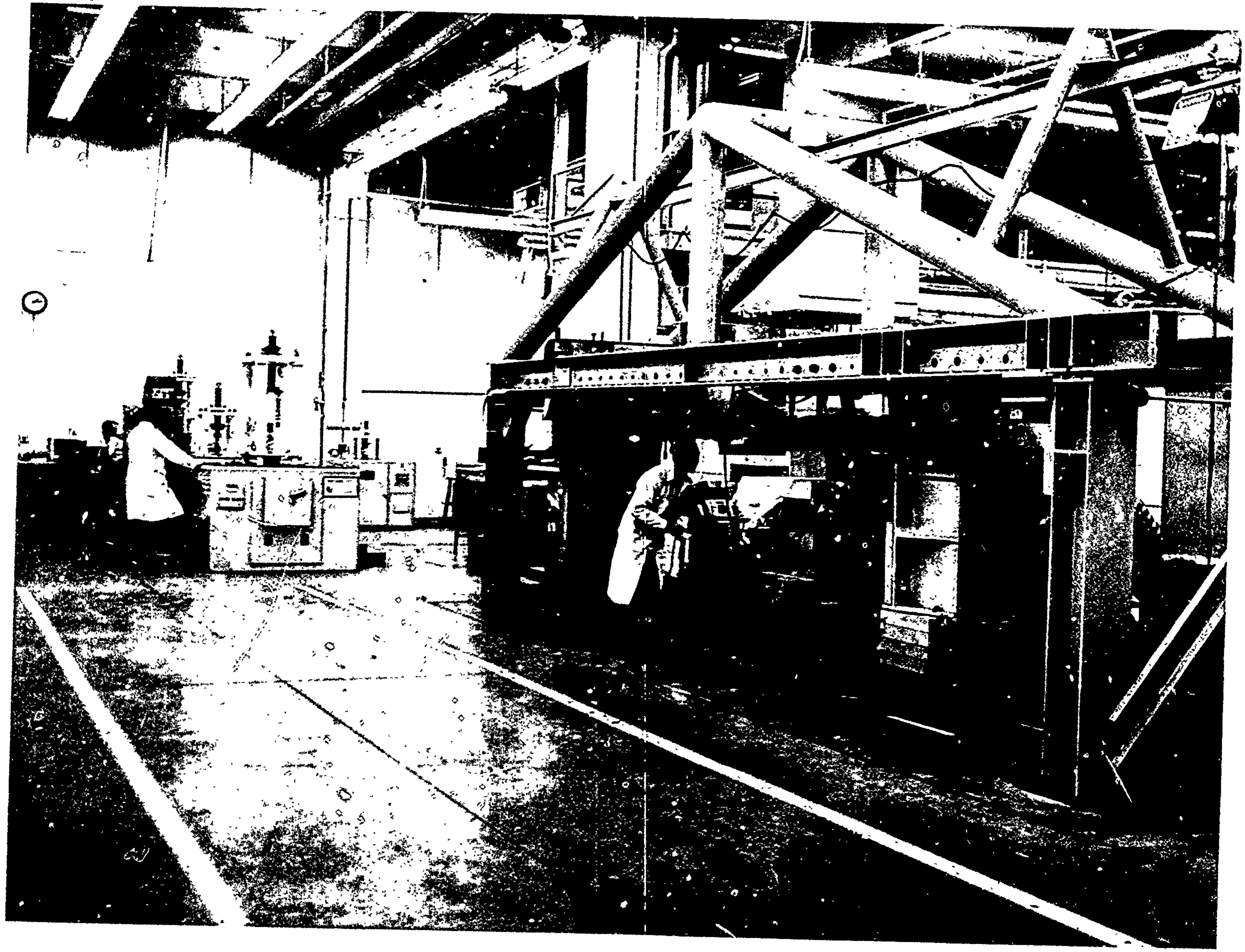


Figure 14 - 250,000 Pound Capacity Fatigue Test Machine

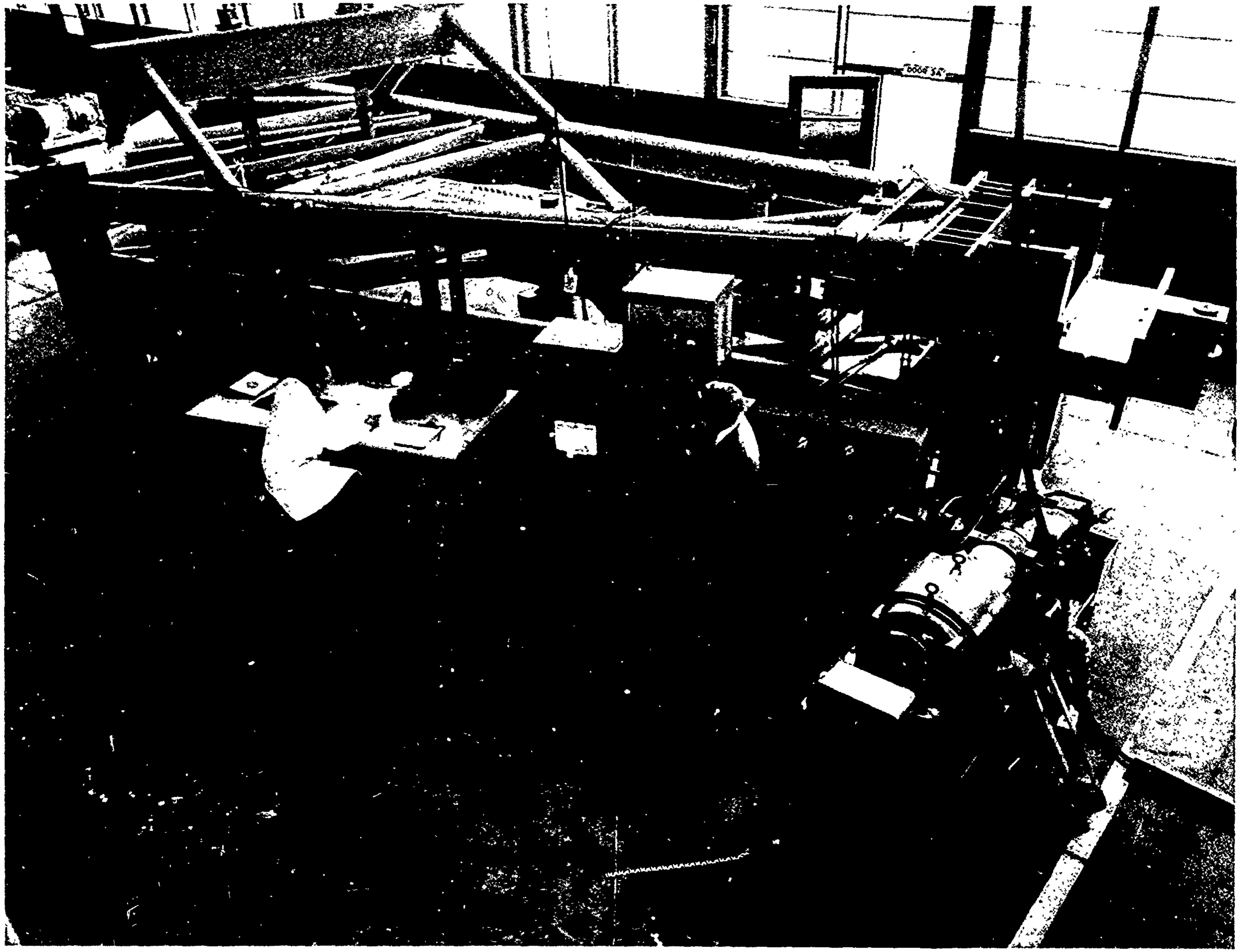


Figure 15 - 500,000 Pound Capacity Fatigue Test Machine Developed For Testing Dog-Leg Joints At Wing-Body Intersection

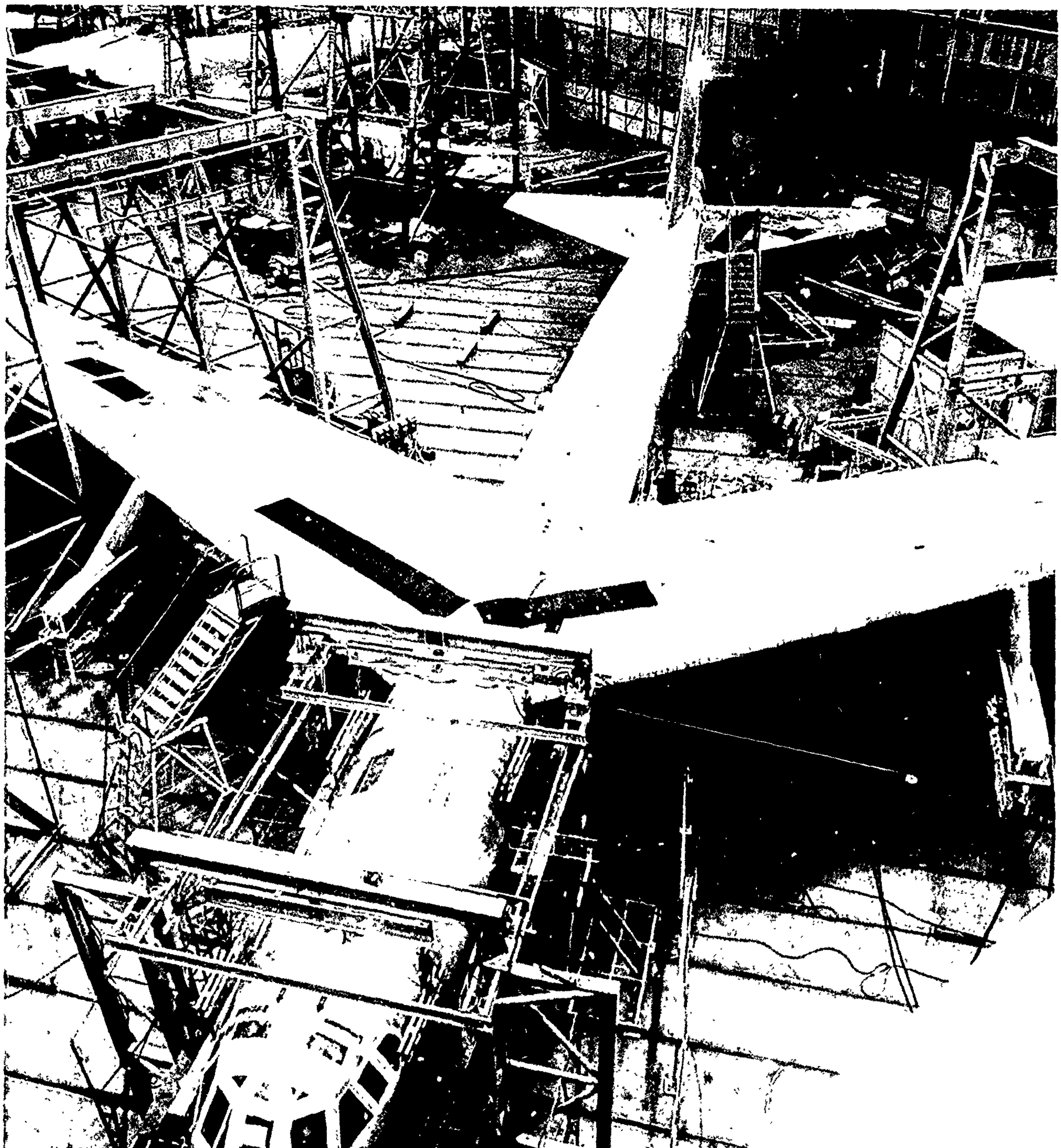


Figure 16 - B-52 Airplane Being Prepared For Full-Scale Fatigue Test

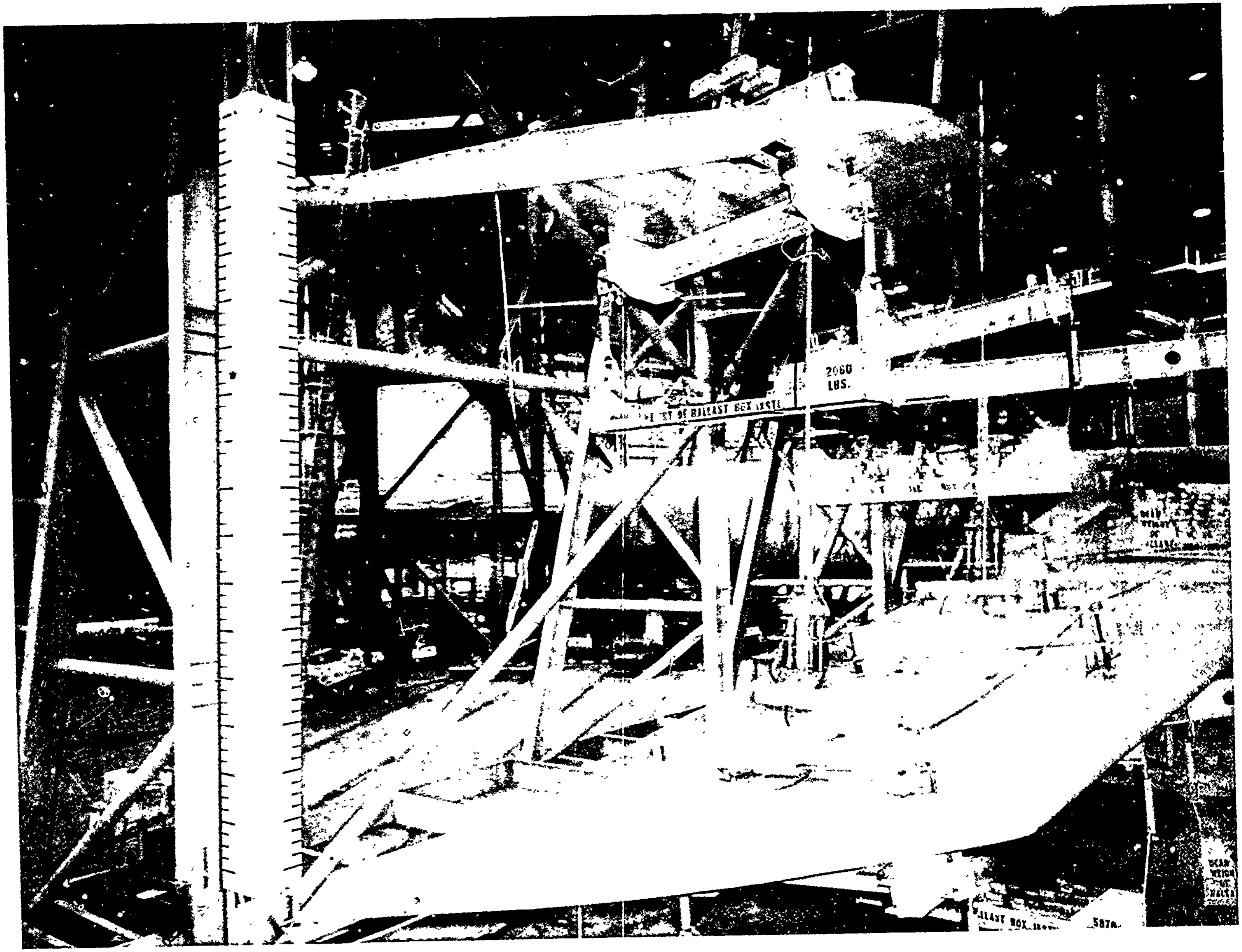


Figure 17 - B-52 ing Fatigue Test

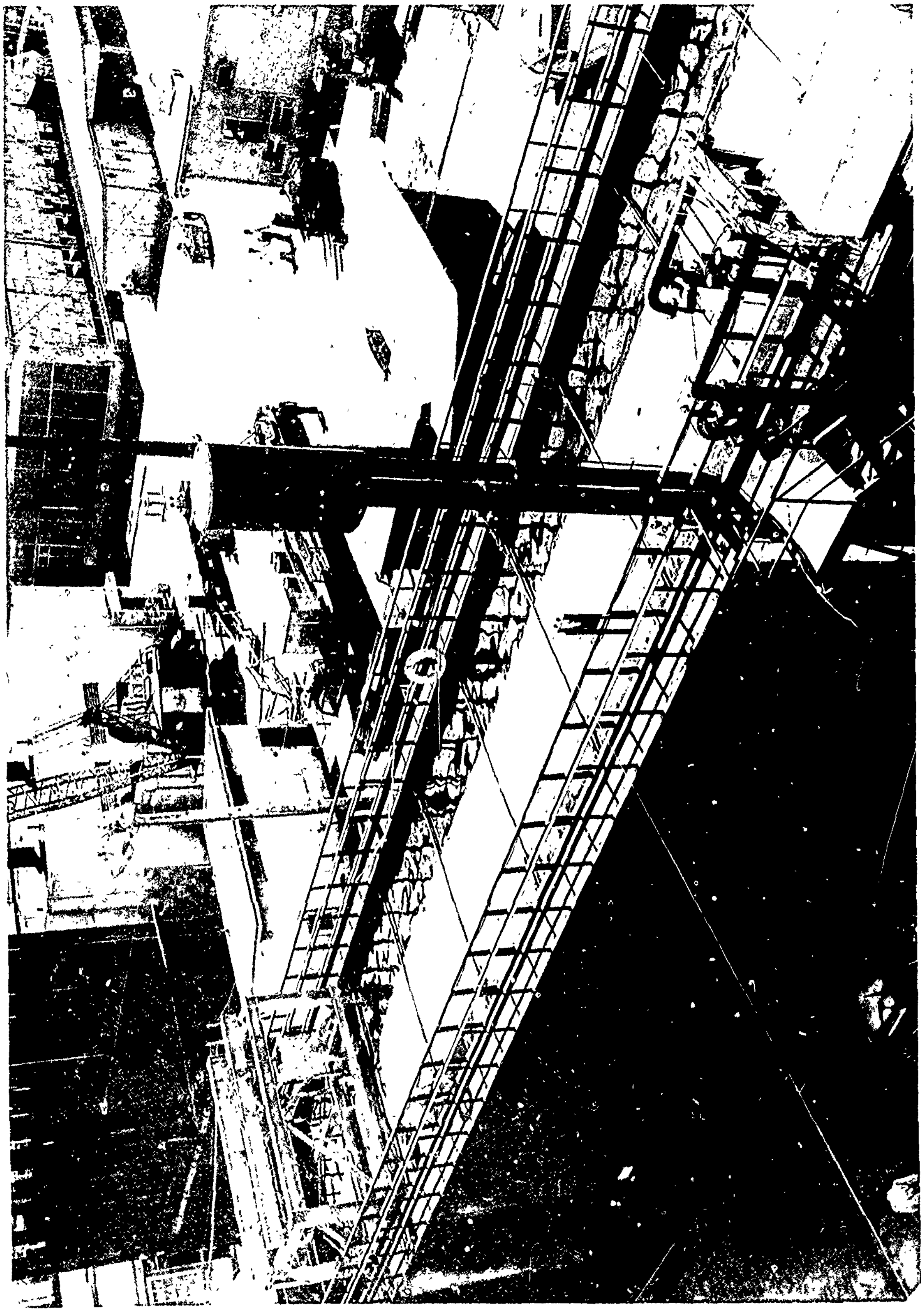


Figure 18 - KC-135 Hydrostatic Test - Test Tank

Editorial Note: Attention is directed to the editorial policies presented in the Preface which were followed in editing the impromptu questions and answers of the session.

DISCUSSION FOLLOWING MR. RHODE'S SPEECH:

MR. McGUIRE, EASTERN AIRLINES:

Was that a 1946 airplane you were talking about, Mr. Rhode?

MR. RHODE:

If you mean the one with the scatter, yes -- it was one of the older ones.

DISCUSSION FOLLOWING MR. KENNEDY'S SPEECH:

MR. SIERADZKI, ROHR AIRCRAFT:

You spoke of designs being conceived which introduce visco-elastic interface damping as a means of sound energy dissipation. Could you elaborate a little bit?

MR. KENNEDY:

Mr. Trapp, WADC, is here this morning and can answer questions in this area much better than I. Would you comment on this, Mr. Trapp?

MR. TRAPP:

Most of our work on damping and methods of utilizing it in structures to maximize structural resistance to acoustic energy is being done under contract with the University of Minnesota. They have contracted for some of this work with other organizations. There are several reports already out on some phases of these studies. We would be glad to include you on the distribution list for reports.

DISCUSSION FOLLOWING MR. VOLLMECKE'S SPEECH:

MR. DALLAS:

I noticed, Mr. Vollmecke, that you mentioned cooperation with the military. I wonder whether you have studied the philosophy used by the military in proving their aircraft in regard, let us say, to the specifications or regulations, and in regard to how that compares with the civil philosophy. In other words, is the military operation so different that the same kind of regulations can not apply?

MR. VOLLMECKE:

We have of course cooperated with the Armed Services in exchanging views on all important design criteria matters. This has been the case in the fatigue area. We have had contacts with the Air Force on this expanding program of theirs, and I believe at this point, it probably would be rather immature to consider Air Force criteria as being final. We are - to answer your question - actually and continuously cooperating with all government agencies to make use of the best know-how available in this particular field.

DISCUSSION FOLLOWING CAPTAIN KEEN'S SPEECH:

VOICE:

What do you mean by design life as opposed to a say, specification life of an airplane?

CAPTAIN KEEN:

It seems to us that it is unreasonable to expect anyone to sign a contract guaranteeing that each individual airplane will survive some specified life either in time or operating hours. I don't know how you could enforce such a contract. On the other hand, we feel that the Navy should specify, in terms of numbers, the hours of service operation that we expect to put on an airplane, and the loading spectra associated with such hours of operation. This would require that the contractor design for that life and loading spectra specified numerically in the contractual document. This further implies an obligation to prove by some satisfactory method that he has actually designed for this life as specified. But this can only be done at present by means of some sort of a laboratory repeat load program which we can only hope is representative. If an airplane goes into service and fails at some life less than the design life, a contractual obligation would exist only if the failure was the result, for example, of some deficiency in manufacturing or of failure to make the final article equivalent, life-wise, to the one which satisfactorily passed the repeat load test. Has this answered your question?

VOICE:

I'd like to make one remark. I think the procedure is a very sensible one - one of the most sensible I have heard so far, but I wonder on what basis you dare to specify fatigue life when you don't know what the life may be? However, I do think that the procedure is the most objective one that can be found.

CAPTAIN KEEN:

This is sort of like the following question: How can you use operational analyses to decide on the proper division of the budget for offensive and defensive capabilities? You can't, but you must do it anyway. You could draw it out of the air, use some random, yet incomplete process, or try to rationalize the problem as best you can on the basis of experience. We do have past history and we are continuing to collect history on the number of hours of operation in certain models of aircraft in various programs, many of which have been described here today. We are getting load histories for aircraft, and thus, we will have data to

improve our specification of load history for a desired design life. We do not pretend that this is ideal. Morever, we cannot question the operator's need to change the mission of an aircraft half-way through its service life. Under such circumstances, our only choice is to use what we know about its design life in determining its new life for the new mission it has been assigned. This leads us into another subject which I think is more properly a discussion for later papers during the Symposium.

DISCUSSION FOLLOWING MR. DALLAS' SPEECH:

VOICE:

Are airline maintenance inspection reports available to the Air Force?

MR. DALLAS:

My answer to that question is yes. The daily mechanical reports go to several divisions of the military services, as well as to several manufacturers. They are used by most of these agencies judging from what I hear.

MR. McGUIRE, EASTERN AIRLINES:

What contacts and activities do the airlines maintain at the manufacturing end in order to preclude fatigue problems and expedite inspection?

MR. DALLAS:

I know that the airlines do a lot in this area. Manufacturers, of course, have representatives at the airline shops to keep track of difficulties and problems so that they can anticipate and carry out fixes. The airlines also have representatives at the manufacturers plants. In the process of buying new airplanes, the airlines monitor the aircraft design to assure that inspection can be carried out in the simplest possible manner. I think that the airlines also encourage some fatigue testing on airplane parts.

DISCUSSION FOLLOWING MR. SMITH'S SPEECH:

MR. McGUIRE, EASTERN AIRLINES:

It appears as if I'm hogging this "mike," but I do have a question on an important aspect of design and operation of transport airplanes. I have noticed that the lighter the aircraft become, the more frequently they fail. Although you mentioned this and also some methods for control of fatigue, it seems that you failed to mention that one of the ways you can control fatigue is by making the parts a little bit heavier. I wonder why that item is missing on your list?

MR. SMITH:

I did not mean to imply that by good detail design, the part necessarily becomes lighter. The point I was trying to make is that I don't feel that an arbitrary reduction of stress level is necessary as a solution to the problem. I think that fatigue is a local problem -- not a general problem. All fatigue cracks originate from stress, pressures, etc., and later attack the structure in a local area. Does that answer your question?

MR. McGUIRE:

No. You answered my question, Howard, but it doesn't fix us up.

Chairman:
Colonel John P. Taylor, Wright Air Development Center

Panel Members:
Colonel Harvey P. Huglin, Wright Air Development Center
Mr. William B. Miller, Wright Air Development Center
Mr. Sidney Berman, Office of the Inspector General, Norton AFB
Col. Howard E. Watkins, Strategic Air Command
Mr. Richard V. Rhode, National Aeronautics & Space Adm.
Captain W. H. Keen, Bureau of Aeronautics, United States Navy
Dr. Millard V. Barton, Air Force Ballistic Missile Division
Mr. R. R. Kennedy, Wright Air Development Center
Mr. A. A. Vollmecke, Federal Aviation Agency
Mr. Allen W. Dallas, Air Transport Association
Mr. Howard Smith, Aerospace Industries Association

Editorial Note: Attention is directed to the editorial policies presented in the Preface which were followed in editing the discussions of the Forum.

OPENING REMARKS BY COLONEL TAYLOR:

Gentlemen, I would like to express General Schriever's regrets for not being able to remain with us all day; Colonel Huglin, Deputy Commander for Development, WADC will act for General Schriever in speaking for the Air Research and Development Command. General Wray is involved in a series of conferences; Mr. William B. Miller, Asst. Chief, Structures Branch, Aircraft Laboratory, WADC, will act in place of General Wray. General Caldara was called back to Washington and Mr. Sid Berman will take his place. We expected General Gerrity to be here with us and hadn't arranged for an alternate, so the questions that are directed to General Gerrity will be covered by the panel as a whole. We have microphones which will be passed to the speakers, and to those of you in the audience who have additional questions. We will take the questions you have asked, shuffle them, and take them in the order that they come up.

MR. O'BRIEN, WADC, TO MR. KENNEDY:

Can there be a structural brittle fracture as well as a fatigue ductile failure?

MR. KENNEDY:

I would say yes. I think there is quite a bit of fundamental evidence which indicates crack initiation can take place in a ductile manner, when you are speaking of an ordinary metalurgical examination of fatigue failure; however, the crack appears to initiate in a brittle manner and the crack becomes ductile as the crack progresses in final failure.

COL. TAYLOR:

Let me add that if anyone in the audience wishes to amplify any answer or ask additional questions, please feel free to do so.

MR. GLASSER, BUDD COMPANY TO MR. RHODE:

Would your recommended procedure require testing of every major structural area of the aircraft involved? If this procedure were applied to the transport aircraft mentioned with the failure scatter you have illustrated, would the stress levels arrived at be prohibitively low? How would you handle this?

MR. RHODE:

I'm not quite sure what you mean by "major structural area". If you're talking about large pieces of structure, that's one thing. If you're talking about critical area structure, that's something else again. Now I would say to the first part of your question - yes, if we define the term "major structural area" as critical area. I'm not at all sure that the stress levels arrived at would be prohibitively low, if we tested all major areas of structure. In fact, I'm inclined to feel that this would not be the case. This question reminds me of the situation with an airplane some years ago where they designed using high stress levels with consequent trouble. We went through a series of fixes or so-called fixes. First, joint design details were improved and verified by tests. Then we went into a second phase which involved swelling out the joint and reducing the stress level. These fixes proved to be inadequate. We found that other parts of the structure failed. Consequently we had to strengthen the entire spar, reduce the stresses, and this yielded a satisfactory fix. The airplane went back into service with no further trouble. This example appears to furnish fairly strong evidence that realistic stresses are not prohibitive.

VOICE FROM THE FLOOR TO MR. RHODE:

In the same area which was brought up by the last question, we heard General Gerrity mention this morning that the only way that he could get assurance of sound structure was by complete disassembly to the point of eliminating the airplane from service permanently. If you had your choice of pulling it apart and inspecting it as against subjecting the airplane to some test to see, after it had accumulated this service, whether any, all, or none of the structure was good for additional service, what would be your choice? That is, would you disassemble to the point where you have no airplane or use up the airplane in a complete load test with a suitable spectrum? Which do you feel might be the more profitable way to expend the airplane?

MR. RHODE:

That's a real humdinger, Al. The fact of the matter is that I haven't thought through that particular possibility. I'm afraid I don't have a ready answer to it. I'd want to ponder that for a long time before trying to give a reply.

COL. TAYLOR:

I don't think that General Gerrity really meant that he wanted to tear down all his aircraft. One chart I used during my presentation showing the distribution

of cracks throughout a large sample of the B-47, was an indication that fatigue or incipient fatigue can occur in many different places; for example, in practically all the bolt holes of the critical sections. What I believe General Gerrity said is that in these critical areas - to be certain that you are turning something back to the operational fleet with full confidence that it can carry out its mission - you almost have to look at each bolt - bolt by bolt. This is not a complete answer, but in talking with General Gerrity at other times, I believe this is his feeling.

MR. VALENTINE, ARDC, FROM THE FLOOR:

Question was lost -- not recorded.

MR. MILLER:

I think what General Gerrity was talking about primarily is the problem of stress corrosion, the inter-granular corrosion that we experienced in some of the B-47 forgings. Really there isn't any way to find inter-granular corrosion except to dig in and look for it. On this other problem that you mentioned, Val, certainly if you look into an airplane and you don't find a crack, the crack may start within a short period of time or just a few cycles after you put the airplane back together. But we hope that after a certain amount of testing has gone on that we will know where cracks are likely to start and at least where the cracks that do start are likely to progress so rapidly that you will have a dangerous condition before the next inspection.

VOICE FROM THE FLOOR:

Permit me to rebut just slightly here. The program that was laid out today was very comprehensive and I think it shows an excellent desire to get on top of various aspects of the problem, but there was this gap of really knowing how much aircraft you had left at any particular time. One of the ways that I think you could get the assurance you need is to take and impound several airplanes coming out of service using those that either have unusual histories or high time in service and find out what you have left. In this manner, you will have integrated all the factors that you are going to be exploring in that airplane's history.

COL. TAYLOR:

This is correct in part. We expect to take the high time aircraft in the B-52 fleet and the KC-135 fleet, for example, and conduct thorough examination of at least one complete vehicle. We will be able to use the VGH recorder we discussed earlier and place some of them in selected aircraft as they leave the factory. We will have a complete life history on these aircraft, and know what loads they have been subjected to. We don't have the instruments available in quantity today, but we hope to have them available in the very near future. The suggestion is a good one and we expect to accomplish the general type of inspection mentioned in our critical first line weapons systems.

MR. BROWN, HUGHES AIRCRAFT TO DR. BARTON:

What do you think of marrying static and fatigue test programs by putting on cyclic loads whether there is one cycle or nine cycles to determine the cumulative effects?

DR. BARTON:

Well, if I understand the question, it involves making a static test and a fatigue test simultaneously. Normally the static tests are to limit and then to ultimate loads. In this case you would probably damage your structure and indications are that it might not then be satisfactory for use in fatigue tests. You are also probably looking for different things on these cyclic tests, so my offhand opinion would be that it might be difficult to arrange such a combined test.

MR. SMITH, CONVAIR, FROM THE FLOOR:

If you remember reading some of the Australian reports in regard to testing Mustang wings, they found a substantial improvement in the fatigue life for those wings that were loaded to about 85% of allowable load prior to fatigue testing.

MR. RHODE:

I think it depends on what you are looking for. If you can actually define the load conditions specifically that you are trying to make a demonstration of, and this involves a pre-loading of say 5%, why then, perhaps, this is the thing to do. But I think you should try to duplicate the actual conditions as closely as possible.

MR. BROWN, HUGHES, FROM THE FLOOR:

We talk about fatigue and cyclic loading causing failures and we talk about static loads causing failures. We separate these two things. I think this is somewhat dangerous. You are faced with a situation where you may have ten limit loads, as well as a thousand or ten thousand lighter loads - where do you break one out and where do you start considering the other? I think that in a test program, these two things ought to be meshed together.

DR. BARTON:

Well, again, I think it's desirable to do such a thing when you can. However, it isn't always apparent that you can use the same specimen for doing both. It is true in the case of missiles that we're generally concerned about the load cycle - endurance, if I may call it that. And if it's possible for you to devise a test that will satisfy both the static requirements and the dynamic requirements, simultaneously, do so. I was trying to recall again, if in our program, this has actually been the concept of any of the tests. Frankly, I can't recall that it has. However, I don't see any reason why it couldn't be done in a specific instance.

CAPTAIN KEEN:

The Bureau of Aeronautics actually had a proposal from a contractor who was developing one of the current models, to do exactly that - to combine the static and fatigue testing programs into one program. The proposal was very intelligently thought out. We gave it quite a lot of serious consideration and eventually decided against it because we didn't really know enough about how to interpret the repeat load program to be sure of getting the information that we still need out of the static program. We still need to know what parts of the airplane will fail, and if you don't find this during a static test program, why then

you don't get that information. Ultimately, some day, when we know enough about the expected loading history and the expected life that we would have in an airplane, to interpret them in terms of a peak load test, then we might eliminate the static test for failure entirely. Have I represented this correctly, Mr. Creel?

MR. CREEL, BU AER:

Yes, I believe so. There is one point that has been mentioned here this morning and from our point of view it is important. An article can be made available earlier for static testing than for fatigue testing because almost invariably there are fundamental changes in an airplane in the production line such that it takes a while to settle down into the actual production article. We can't wait until then to perform the static test. We need to build the static test article early in the game to prove that the airplane is safe to operate. I find that the pilots are interested in a factor of safety of one and a half for flight conditions. Even though the fatigue designer may not be interested in it, I find that the pilots are still enthusiastic about the one and a half factor. How do you find this out? You do it by analysis only. But let's not forget about the history of flagrant fallacies in analytical procedures that have been brought out. So we need the static test. We must build the static test article and wait for a more complete article for the fatigue test.

MR. RHODE:

I want to make one comment on this question. It seems to me that there is a fundamental point which has to be considered. While we hear a great deal about fatigue life, we hear little about the life that is associated with the probability of static failure. It is statistically true that there is from the static viewpoint, a certain life time and probability of failure - a number of failures per hundred thousand hours of flying. Thus, we are involved with two life times, fatigue life being the other. If the areas of failure are about the same or are precisely the same, I would say that the marriage of static and fatigue tests might make sense. But I think we are stuck with a situation where the fatigue life is less than the life due to the probability of static failure and under these conditions, I don't think it can be done in principle.

MR. WARD, NASA, FROM THE FLOOR:

By conducting the fatigue test and then the static test on the same article, and presuming you detected no cracks, you would have an indication of any effects that fatigue had upon the static strength of the aircraft. These results might turn out to be quite different than had you run a static test at the beginning of the program.

VOICE FROM THE FLOOR:

The question was asked by a missile man. Both of the answers have been given by airplane men. There is a fundamental difference in attitude on missiles and airplanes and I think it needs to be considered.

COL. TAYLOR:

Do any missile men present wish to make a contribution? There being none, we will try a new question.

MR. CARLSON, HILLER AIRCRAFT, TO GENERAL CALDARA:

Since military superiority is a relative situation, it seems pertinent to ask whether the U. S. Air Force has knowledge concerning the statistics of metal fatigue mishaps in the Air Forces of the Soviet Union? If so, how do they compare with those presented in your paper?

MR. BERMAN:

I would like to say that I personally don't know of any statistics that the Soviets have concerning metal fatigue. But I do hope and believe that our espionage system in Russia is as good as they have over here and I hope we have the right information in the proper place in Washington.

MR. CREEL, BU AER, TO GENERAL GERRITY:

Have ARDC studies isolated the fundamental reasons for USAF stress corrosion problems? If so, will you please brief them for us?

MR. KENNEDY:

Stress corrosion is something that we run into occasionally. It is a combination of stress and corrosion occuring to a point where a crack occurs. It may be due to a variety of causes. If the stress is very high, it doesn't take very much corrosion to start a crack and vice versa. Extensive work has been done in the laboratory on this problem. Unless you have such an environment that you get no corrosion, which would be an ideal condition, you can expect stress corrosion to occur in airplanes.

COL. TAYLOR:

Is there anyone here from the metals and materials end of the business who would like to add a few thoughts along this vein? It is a very important one. As General Gerrity mentioned this morning, the one thing that we really can't forecast for the B-47 fleet is the effect of intergranular corrosion. We just do not know enough about it. Since there are no comments, we will proceed to the next question.

COL. STONE, ARDC, TO COL. TAYLOR:

To what extent would placing aerodynamic weapons systems inside a protective shelter accelerate sonic fatigue effects?

DR. BARTON:

The indications are that if one were to fire a missile from a silo, the acoustic noise level would be vastly increased unless special precautions were taken to reduce it. The various tests that have been made in this country and England indicate that it is possible, however, to put on an acoustic awning which will bring the level down to about that of free shielding.

MR. MILLER:

In the case of aircraft, I don't believe it is practical to insulate an airplane from its own power plants. However, there is a probability that we could reduce

some of the effects caused by ground run-up, because, certainly, in some of our newer weapons sytems with the amount of sound that is being generated by the power plants, a mechanic can do a considerable amount of fatigue damage in running up the engines on the ground while he is working on the engines. It could be that some type of shield to keep the sound levels from becoming quite so high in the case of ground run-up would be useful. However, I don't believe that there is any way I can think of right now to build a shield like this around an airplane that is taxiing out and making a take-off run or flying in the air.

MR. BERDAHL, SAC, TO COL. TAYLOR:

Don't we need an expanded VGH recorder program on all medium and heavy military aircraft?

COL. TAYLOR:

Let me state, gentlemen, that our original intent was to place recorders in all of our first line aircraft as a safety of flight item. However, in looking at the problem objectively, we felt that we could learn a great deal - certainly initially - through a statistical analysis of a reasonable statistical sample and this is what we are attempting to do. Later on, depending upon the results, we will take further steps either to reduce the statistical sample or to increase it. We first must find out what this mass of information will mean to us. What we are attempting to do is to learn to reduce all data automatically, having the information flow from the operational squadrons to the AMA's, study the results, and pass the vital data on to the engineers. When we learn how to best utilize this vital data, we will then pass the essence of it on to you through the medium of Tech Orders, Technical Reports, memoranda, etc. We will try to get this information to you just as rapidly as possible. It would be useful in certain ways to instrument all first line aircraft but it would also be an extremely expensive program so we must try to find a reasonable compromise.

MR. Mc GUIRE, EASTERN AIRLINES, FROM THE FLOOR:

Just a few days ago I had a discussion with a flight recorder company on just that particular aspect and we were talking about using various information - flight recorder information. We found out it would have cost us somewhere around a hundred thousand dollars for the piece of equipment which would actually edit this information picked up by the flight recorder. It is a very expensive proposition.

COL. TAYLOR:

Yes, it is an expensive proposition when you look at it from the viewpoint of putting the recorder on every aircraft. What we are trying to do is to temper engineering conservatism with statistical theory so as to come up with the degree of accuracy we feel we must have. The results of this work will be made available to the industry. We hope it will be useful to you as an airline operator.

MR. WACHS, SIKORSKY, TO MESSRS. RHODE AND KENNEDY:

Will you please elaborate on the extrapolation of small population fatigue data on structure to yield satisfactory life estimates?

MR. RHODE:

I'd like to preface my answer to that question with the statement that I'm no statistician. I can't speak as a technical expert in this area. But, from where I sit, it certainly appears to me that the comments I made in my discussion this morning are strictly correct. Walter Tye gave a paper in England a couple of years ago in which this question was dealt with rather extensively. He required a sample consisting of six components to apply his method of analysis. We assume that he used a very large statistical sample to arrive at the appropriate probability distribution - a frequency distribution - for the particular picture he displayed. With six specimens as a basis for an estimate, it was found possible to determine the sample mean value, and also through statistical procedures, to estimate that the safe life (population mean) was located at a level .32 times the mean value of the test for most of the specimens. Mr. Tye went one step further and examined the reliability of this estimate of the mean on a Silly Curve (translation of this transcription is left as an exercise for the reader - Ed. Note). To do this, he determined the set of limits or interval which would contain the safe life with a high degree of probability or confidence, say 90%. He was able to establish that if an interval .32 times the mean value of the test was assumed to contain the safe life, a 90% confidence level could not be reached. Since for a given sample size, as the degree of confidence increases, the length of the confidence interval must increase. He found that to provide an interval which would contain the safe life with 90% confidence - one in ten chance of failure - the mean value of the test would have to be multiplied by 1.4 instead of .32. Mr. Press is in the audience and he may have a comment on this question.

MR. PRESS:

There seems to be a habit of throwing hot potatoes around here! I think the question is this -- we are trying to arrive at a series of airplanes that will have an average life of so much, and beyond that, have only one out of a hundred, for example, that will fail within a certain minimum lifetime. No one is really concerned about the average life; what we are really concerned about is the life of the first airplane that fails, and failure is something we want to avoid. We are talking about estimating from some part of a probability curve that is way out in the neighborhood of one-hundredth. To be 90% confident of this lifetime in which only one failure will occur in a hundred airplanes, something like a thousand samples would be required. This, of course, gets beyond the question, but it is the kind of thing that people are talking about. I think that a sample of six components would not give a satisfactory statistical solution using Mr. Tye's method of analysis. A sample size of one thousand would be required and this, of course, is typically unsatisfactory from an economics point of view. So statistics do not appear to offer a solution. Perhaps the solution lies in designing materials that don't have the scatter.

COL. TAYLOR:

Mr. Kennedy, do you have any comments on the question?

MR. KENNEDY:

Personally, I can't say too much on it. I received a great deal of help from Mr. Trapp and from Mr. Forney in preparing my paper, so I'm going to ask Mr. Trapp if he cares to comment on the question.

MR. TRAPP:

The work Mr. Kennedy was referring to this morning, was done by Dr. Weibull, Sweden. It was based mainly on the regression theory. To explain very simply, it concerns the development of an equation which fits the SN curve or any other curve of this kind. This gives us the possibility of using all data points, which are distributed over the whole SN curve, to obtain a statistical evaluation at any stress level which we select.

MR. JAMES TO MR. SMITH:

Would you please comment on the status of the fatigue problem regarding composite sandwich structure both as individual test panels and relative to edge connection of panels?

MR. SMITH:

I'll have to confine my remarks to sandwich panels as typically used in trailing edge structure of subsonic transports and airplanes in that general class. I'm not qualified to speak on missile use or high supersonic airplanes. If I understand the question correctly, any sandwich panel presents the problem of attaching or joining it to another panel or the substructure. This is one of the most critical problems in the sandwich panel design area. It is usually not too difficult, for the noise environments that we find in the trailing edges of the KC-135 or the B-52, to design a sandwich panel which does not fail except at its attachments. The critical area is local and requires a great deal of test work and a great deal of attention to detailed designs if the attachments are to attain anywhere near the full benefit that is inherent in the sandwich design. I don't know whether I caught the point of the question or not.

COL. TAYLOR:

If not, will Mr. James please clarify?

MR. JAMES:

My question was directed toward the problem of sandwich structures in general. We seem to be dealing more with bolted attachments. This seems to be a fundamental question, that's going to disappear. I might have addressed the question to any one of you here. I just happened to pick Mr. Smith. Perhaps you, Colonel Taylor, might have a comment.

COL. TAYLOR:

In the more modern airplanes of today that we're contemplating, it becomes a real problem. Let me say that we have had some experience on sandwich construction and more recently on double wall sandwich construction in which we've tried to simulate the static conditions which one will encounter later on. We've also added aerodynamic heating effects. We are beginning to learn something about the effect of crumpling loads on this kind of structure. Mr. Miller has been working in this area and may have a comment.

MR. MILLER:

Of course, when we have a piece of structure with a relatively soft core and hard faces on it, we do have a problem of the edge attachment. It is something that is the subject of a considerable amount of thought and research right at the present time, so I can't solve the problem now.

MR. SCHULER, LOCKHEED, TO COL. TAYLOR:

When and in what form will the results of the B-66 low altitude data be made available?

COL. TAYLOR:

When we started the B-66 program on an expedited basis about a year ago, we didn't realize how difficult it would be to separate the background noise problem from the data we were trying to get nor did we realize the amount of data that we would have for reduction purposes. At the present time we are getting about two and a half million data points for each five minute run. We wanted to try to shoot for a hundred hours actual flight time in every part of the country and seek out different climatological and meteorological conditions. The contractor has been carrying on and now expects to make the following releases. First progress report is due in about six weeks and will cover gust values between 0.6 and 2-1/2 cycles per second. Follow-up reports will be issued every two months. The final report is due in October 1961 and will cover a complete spectrum from 0.1 to 12 cycles per second. When we started to get this information, we were not completely familiar with the cyclic range which would be involved. This too has required an extension of the state of the art. We have been discussing this problem with the contractor recently and have been assured that their program will be adhered to and reports will be issued in the time periods mentioned. We are most anxious to make it available to all of you.

MR. BROWN, HUGHES, TO COL. TAYLOR:

Were you gathering strength data from the test on the B-66 or environmental data? Your chart shows strength data.

COL. TAYLOR:

Mr. Miller, would you like to answer that question?

MR. MILLER:

We were looking for two things -- the nature of turbulence below a thousand feet above the terrain, and the kind of response we get from the relatively stiff type airplane that we're using. If you're referring to the statement on the chart which referred to strength - we are talking about the gust strength rather than some other type of strength. The airplane we have is one which was used in an operation in the Pacific to measure loads resulting from a nuclear explosion. We had the airplane thoroughly instrumented to determine loads on the airplane from certain inputs. New equipment was built into the front end of the airplane, which enters a gust before the airplane responds. The instrumented boom will tell us what type of turbulence the airplane is entering. We will also obtain the response of that vehicle to this type of turbulence as well as the pilot's reaction at the time. At the same time, we are taking terrain photographs to

establish the type of terrain we are over while recording the turbulence data.

MR. BYLER, OCAMA, TO MR. VOLLMECKE:

What procedure does the FAA have for recording flight data on commercial jet aircraft? Will the data from the many different airlines be collected and evaluated?

MR. VOLLMECKE:

The FAA has no data collection program. The airlines are engaging in a voluntary data collection effort in cooperation with NASA, and VGH recorders are being carried on various fleets of jet airliners. NASA, I am sure, will publish the results when sufficient information has been collected. A second program is also under way. It involves both airplanes which have been imported -- the Viscount, and the Fairchild manufactured F-27. Both types of airplanes are equipped with fatigue meters. Relative to the Viscount, Capitol Airlines and Continental Airlines report they they send recorder strips direct to Vickers in England for their use as well as the ARB in possibly resolving fatigue life figures. The F-27 program is just getting under way and the results will also be sent back to the aircraft manufacturer - in this case Fairchild. I am sure that Fairchild could make the results available to anyone who is interested.

COL. TAYLOR:

Would Mr. Dallas care to make any comment on that?

MR. DALLAS:

That is substantially correct.

MR. GLASSER, BUDD COMPANY, TO DR. BARTON:

Should ground handling loads be allowed to influence the live missile configuration? It would seem that enough shock absorbers could be placed in the ground handling equipment to relieve this condition.

DR. BARTON:

I don't believe that the configuration should be compromised with ground handling loads in the large sense. However, it is possible to make minor redesigns of local areas without weight penalty, which benefit the whole system.

COL. TAYLOR:

Thank you very much, Dr. Barton. Just before I came in here, Mr. Schleicher of North American brought up an interesting point. Perhaps it would be well to cover this point he was making with the entire group. Would you give us your comments, Mr. Schleicher?

MR. SCHLEICHER:

Mr. Chairman, gentlemen. As one who has been interested in fatigue for a good many years, I couldn't help but make an observation at the meeting today

which I would like to pass on to you. It is simply this. If we go back approximately ten or twelve years, and my memory doesn't tell me exactly when this happened, but there were about fifteen of us who met in New York City at that time under the banner of the AIA to discuss fatigue problems. I think it was the forerunner of the W76 Committee mentioned by Howard Smith in his presentation. We discussed the various aspects of fatigue and went through the old routine. We referred back to practically every heading on this subject; i.e., we divided the fatigue problem into basic elements -- solid state physics, materials research, crack propagation, design, accurate stress analysis, and lastly load information, load spectra, and the combining of these two in statistical analyses of the data. Finally, at long last, we got to the crack or failure -- what caused it and so forth.

We know that each one of these aspects of fatigue is highly related to probability analysis and all the assumptions that go with it. But the one fundamental thing that we did discuss at that time, which we thought was most important, was the careful analysis of case histories. In this area, we drew a lesson from the medical profession. When an unknown disease makes its appearance, the doctors don't start out with a microscope and all the fundamentals, trying to synthesize the cause of the disease, and immediately come forth with an answer. They study each case history as it comes up, extremely carefully, going into a lot of detail study of each factor, looking for its recurrence in hope of coming up eventually with some basic answers as to what caused the disease and what will cure it. In our discussion, we applied this principle to the fatigue problem. Among the many recommendations that we made at that time was one in which we went on record stating that we were in favor of better load data acquisition, better load information, a more refined method of analysis, more accurate stress analysis, and development of instrumentation. But we felt this wasn't enough. We needed something over and above this, namely, a careful review and analysis of each live case history as it came up and studying it under the full knowledge of the state of the art to see what recurring pattern we could find - this we thought might lead to the solution, at least in part, of this big question of fatigue.

I would like to leave with you the thought that out of that meeting in New York came the idea that we needed to put together all of the information that we could accumulate on live case histories. This does not mean that you have to go out into an open corn field and analyze a crashed airplane. You could take a case history of a major fatigue failure which you might get in the laboratory, and break it out into separate studies, the results of which might build up and add a little credence to one method of analysis of fatigue over another.

We have misused the term, I think, of "designing for fatigue". There is nothing more contradictory on the face of the earth. We are all designing against fatigue as Al Dower said, and let's not forget that.

I believe that what we need to do, as a result of studying each case history that comes before us, is to devote more time to making an analysis of that situation with the idea of seeing how much more accurately we can now predict certain phenomena than we could ten years ago. I believe that this job is over and beyond the problem of licking any particular fatigue problem that arises. I know how these problems are handled - something happens, the telephone rings, you get on a hot program, you work night and day, you get a fix which you think will work, you make a recommendation and then you go on a vacation. But, there is a step beyond this point. One needs to put together more of the fundamental information, apply it to the case at hand, make up a full case history, and then

try to learn from it. Otherwise, I'm afraid we will meet here ten years from now without having synthesized enough basic data to enable a full prediction of airplane life. So I highly recommend this once more to you.

COL. TAYLOR:

Thank you very much. I suggest this thought as an example of what might be put into our questionnaire.

We only have time to read one more question. The question is: "The simplest analysis that could be made of the results obtained from such individual fatigue analyses would be a percentage of how many times was the trouble due to insufficient knowledge and how often was it due to inadequate stress analysis. I should like very much to get an answer from Mr. Schleicher on what his figures show on his analysis of his case histories?"

This is an interesting question, but our adjournment time is here. I hope it will be discussed during the social hour coming up! I am sure that some of us are suffering a little bit from sonic fatigue by this time and other forms of physical fatigue. If you had one of the many excellent questions not yet answered, our speakers will be available generally at the cocktail hour. You might negotiate proper answers at that time. Thank you all very much. Today's meeting is adjourned!

INTRODUCTION TO SESSION I

COLONEL HARVEY P. HUGLIN

WRIGHT AIR DEVELOPMENT CENTER

In this morning's session on the "Analytical Approach to Fatigue Design", your chairman will be Professor A. M. Freudenthal of Columbia University. Professor Freudenthal has been a member of the Civil Engineering Department of Columbia University since 1949, where his efforts in the field of structural fatigue have gained him national recognition. He was awarded the Norman Medal in 1948 and 1957 by the American Society of Civil Engineers and in 1956 was honored by the Swedish Aeronautical Society. He has authored numerous papers and several books including "Inelastic Behavior of Engineering Materials" and "Fatigue in Aircraft Structures". His 30 years of professional experience include engineering consultant and teaching duties with the Port of Tel Aviv, Palestine; the British Army of Trans Jordan, the Hebrew Institute of Technology and the University of Illinois. He is a native of Poland and completed his formal education leading to a Doctorate of Science in Civil Engineering in Prague, Czechoslovakia. He is a member of several engineering and honorary societies such as the American Society of Civil Engineers, the Institute of Aeronautical Sciences and Sigma Chi.

2

DISCUSSION OF METHODS OF FATIGUE ANALYSIS

By

F. R. Shanley

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A mathematical theory of fatigue is presented from which stress analysis and design methods can be developed. The basic physical principles are, (1) that rate of crack growth (or damage) is controlled primarily by the amplitude of cyclic plastic strain, (2) the rate is also affected, in a secondary manner, by the magnitude of the normal stress acting on the slip plane. The theory includes the effects of endurance limit, mean stress, combined stresses, stress concentrations, variable amplitude strain or stress, and thermal fatigue. A comparison is made between the prediction of fatigue life and the prevention of fatigue failures. It is shown that the latter procedure can be placed on a satisfactory engineering basis.

1. INTRODUCTION

Historically, the first engineering approach to the fatigue problem was from the viewpoint of design for prevention of failure, rather than prediction of fatigue life. (See Timoshenko's History of Strength of Materials, (Ref. 1), pp. 162-173.) The object was to find a "safe working stress" for repeated loading. Wöhler, who introduced the S-N diagram, used the endurance limit as a working stress, with a factor of safety of 2.

As we well know, this simple method cannot be used for aircraft and missile structures. Even if there were a definite endurance limit it would be out of the question to provide sufficient structural material to insure that the applied stresses remained below this limit at all times. Another

complicating factor is that the applied loads have variable amplitude. In our efforts to solve this problem we have developed methods that are expressed in terms of cycles, or "life," rather than stress. Fatigue tests also give results in terms of life. For these reasons the emphasis has shifted from determination of allowable stresses to the prediction of life. This is unfortunate, because fatigue life cannot be predicted accurately.

The object of this paper is to develop a system of stress-analysis and design for repeated loading of variable amplitude, using basic physical principles. Because of the wide range of subjects covered, the treatment of each cannot be taken up in great detail, nor can an exhaustive list of references be given. The paper is intended to serve as a basis for coordinating the various aspects of fatigue that will be covered in greater detail in other papers to be presented.

2. MATHEMATICAL THEORY OF FATIGUE FOR CONSTANT CYCLIC STRAIN AMPLITUDE

We begin with the case of cyclic plastic strain of constant amplitude. The physical postulates (conceptual model) to be used are as follows.

(a) Cyclic plastic strain is controlled so that the amplitude is constant.

(b) A fatigue crack starts at the beginning of loading, i.e. the "nucleation" period is considered to be negligibly short.

(c) Rate of growth of a crack is a function of cyclic plastic strain amplitude.

(d) Rate of growth of a crack is affected in a secondary manner by the normal stress on the slip plane in which the crack is growing.

(e) Fatigue "failure" is arbitrarily defined as the attainment of a certain critical crack area (not necessarily that at which actual failure occurs).

No specific assumptions are made regarding the detailed mechanism of crack growth. Various hypotheses have been advanced, nearly all of which are covered in a general way by postulates (b), (c) and (d).

Many investigators (Refs. 2, 3, 4, 5, 6, 7) have found that when cyclic strains of constant amplitude are applied in a test the ϵ-N diagram plots as a straight line on log-log paper, over a large range of strains. Even more remarkable is the fact that the curves for widely different

materials and testing temperatures fall within a fairly narrow band and have approximately the same slope. This is shown on Fig. 1, from Ref. 2. In general, there seems to be a unifying principle that virtually eliminates "material properties" as such. A single line has been drawn on Fig. 1 to indicate that the average slope of the ϵ-N lines (in terms of logs) is about $-\frac{1}{2}$.

As stated by Coffin (Ref. 3), the use of plastic strain as a basis for fatigue theory is strongly supported by recent investigations of the physical nature of fatigue. It is therefore important to develop a mathematical theory for controlled strain of constant amplitude.

In accordance with postulate (c) we assume that rate of crack growth is a function of the cyclic plastic strain. For the moment, we ignore postulate (d).

Since the cyclic plastic strain is assumed to remain constant, we assume that the rate of crack growth is also constant. This gives

$$\frac{dh}{dn} = C_1 \Delta\epsilon_p \tag{1}$$

where

h = crack depth
n = number of cycles
$\Delta\epsilon_p$ = plastic strain increment (half width of hysteresis loop)
C_1 = constant

In Refs. 8 and 9 the author defined "failure" as the attainment of a critical crack depth. If this criterion is used with Eq. (1) the failure equation becomes

$$N \Delta\epsilon_p = C_2 \tag{2}$$

where N = number of cycles to failure, under constant strain increment $\pm \Delta\epsilon_p$.

The slope of the corresponding $\Delta\epsilon$-N curve is -1, which does not agree with the experimental value of approximately $-\frac{1}{2}$, from Fig. 1. A possible explanation was suggested by W. R. Micks, of The RAND Corporation, who proposed that the criterion for failure be changed from crack depth to crack area.

Fatigue cracks grow in depth and width at the same time. Furthermore, during most of the fatigue life of a properly designed part the cracks remain small and have a circular or elliptical shape. We therefore express crack area as

$$a = C_3 h^2 \qquad (3)$$

where

a = area of crack
h = crack depth

The rate of growth of crack area, per cycle, is given by

$$\frac{da}{dn} = C_4 (\Delta\epsilon_p)^2 \qquad (4)$$

The constant C_4 can be thought of as representing the degree of unbonding in terms of crack area. This constant is actually a function of normal stress, spacing of slip planes, environmental conditions, etc.

For constant rate of growth

$$a = C_4 (\Delta\epsilon_p)^2 n \qquad (5)$$

We now define fatigue "failure" as the attainment of a critical crack depth a_{cr}, and fatigue "life" as the number of cycles, N, at which this occurs. For the present we assume that a_{cr} is a constant. Then

$$a_{cr} = C_4 (\Delta\epsilon_p)^2 N \qquad (6)$$

Combining constants, we obtain the fatigue failure equation

$$N = \frac{K}{(\Delta\epsilon_p)^2} \qquad (7)$$

This gives a slope of $-\frac{1}{2}$ when log $(\Delta\epsilon_p)$ is plotted against log N, as on Fig. 1. A line having this slope has been drawn through the point $\Delta\epsilon_p = 0.006$ and $N = 10^4$. The corresponding value of K in Eq. (7) is 0.36.

Equation (7) represents a theory of fatigue that is independent of material properties. It can be used directly for situations in which plastic strains of constant amplitude are cyclically applied. Coffin (Ref. 3) has used such an equation for the analysis of thermal fatigue.

3. THEORY OF FATIGUE FOR CONSTANT CYCLIC STRESS AMPLITUDE

In the conventional fatigue test the applied cyclic force (or moment) remains at constant amplitude. For a given loading condition the stress can be calculated by the usual engineering methods. The question is: how is cyclic plastic strain related to cyclic stress?

Fatigue cracks usually start within single crystals and in many cases remain within the crystal boundaries for a substantial portion of the fatigue life. We therefore ask: how is the cyclic plastic strain in a single crystal affected by the state of stress? For polycrystalline materials a single crystal may be regarded as a tiny "test specimen" embedded in an aggregate of other crystals. This aggregate may be thought of as a "testing machine" which imposes a controlled cyclic strain on the individual crystal. In other words, the cyclic strain in the single crystal must conform with the cyclic strains of the aggregate in the vicinity of the crystal. These strains are controlled by the integrated behavior of the aggregate, not by the stresses in the crystal alone. This concept permits the cyclic strain theory of fatigue to be used in conjunction with conventional stress formulas, theories of plasticity, etc.

As a first step in introducing stress into the theory we can determine how the stress varies in a test conducted at constant strain amplitude. Fig. 2, from Johansson's tests (Ref. 4) indicates how the applied bending moment varied with cycles, for a certain constant cyclic strain. For one material the stress increases with increasing cycles, during the first part of the test. This is a "strain-hardening" effect. The other material shows a "strain-softening" effect. We see that the relationship given by the conventional "static" stress-strain diagram does not provide an accurate basis for converting the ϵ-N diagram into the S-N diagram.

For a strain-hardening material under cyclic stress of constant amplitude the cyclic plastic strain decreases as the number of cycles increases. For a strain-softening material the width of the hysteresis loop would become larger with increasing cycles.

To convert Eq. (7) to a stress basis we need to know how the cyclic plastic strain varies with stress and with cycles. If the strain-hardening or strain-softening is completed during a small fraction of the fatigue life the stabilized plastic strain can be used. It is assumed to be a function of the cyclic stress.

$$\Delta\epsilon_p = f\ (\Delta\sigma) \qquad (8)$$

where $\Delta\sigma$ = half amplitude of cyclic stress.

In Refs. 8 and 9 the author used a power function for this relationship.

Substituting such a function in Eq. (7) gives

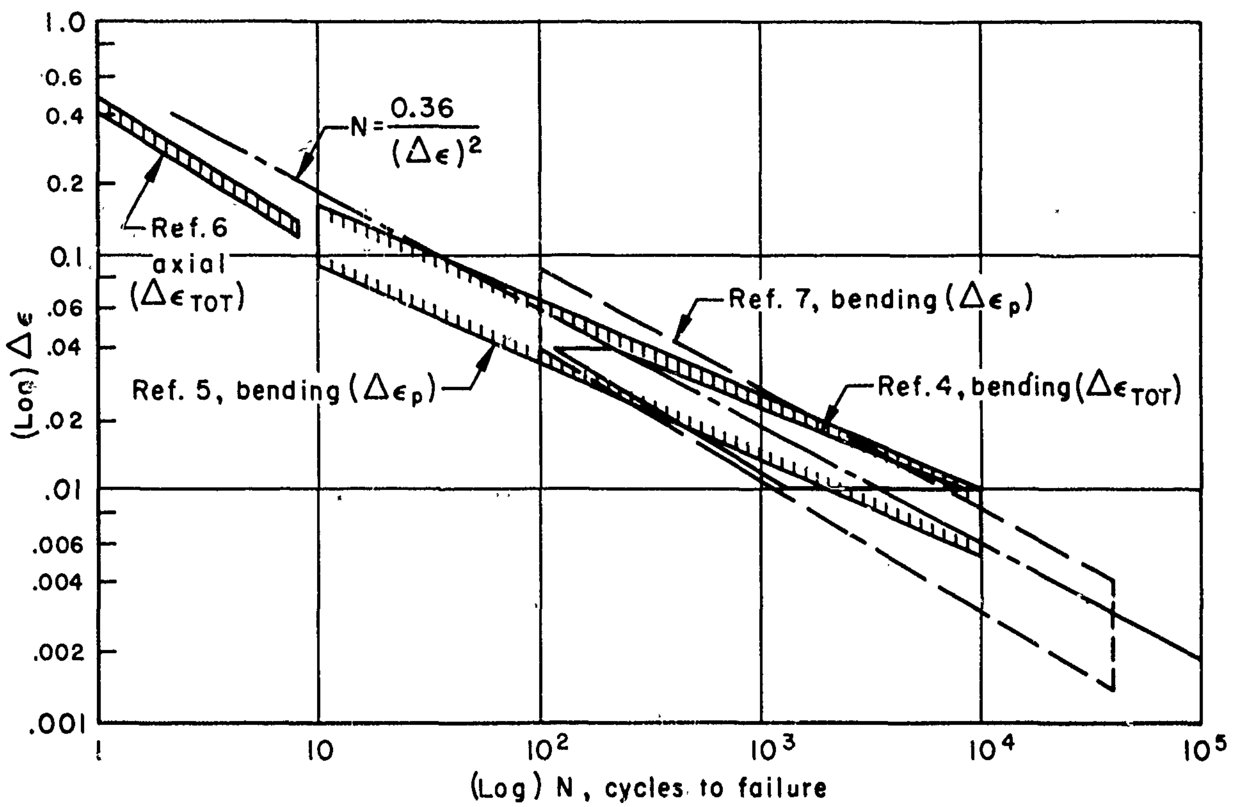

Fig. 1—Fatigue data from controlled strain tests (from Ref. 2)

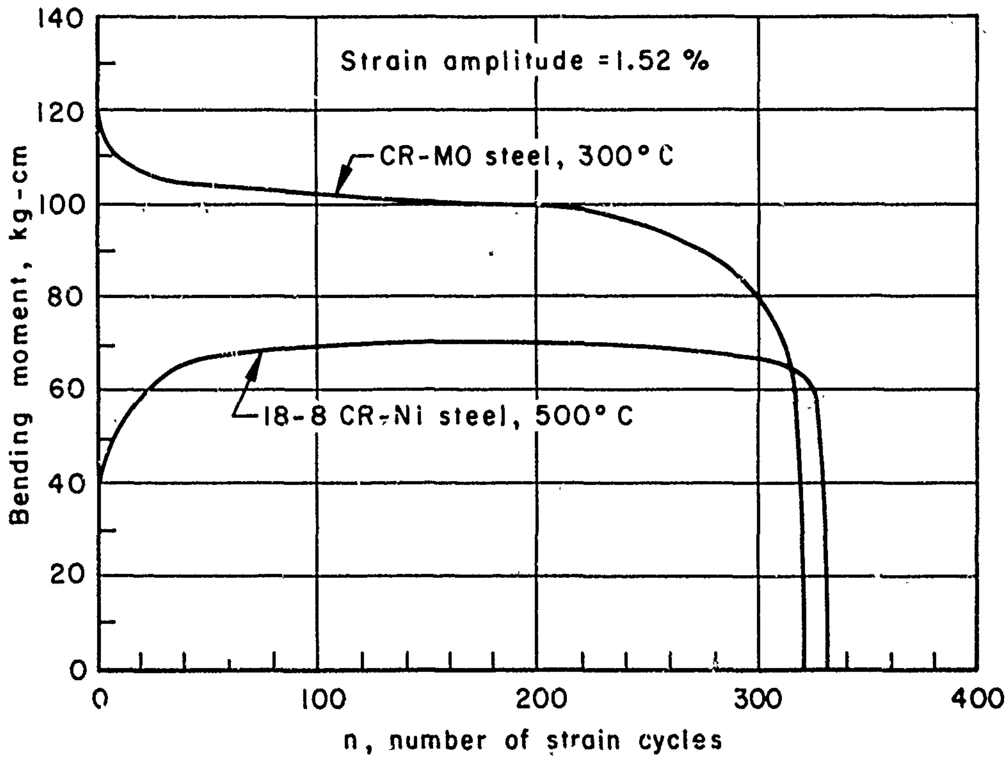

Fig. 2—Bending moment required to produce constant cycle strain

$$N = \frac{K}{(\Delta\sigma)^x} \tag{9}$$

This failure equation plots as a straight line on log-log paper. If an exponential function is used the resulting equation plots as a straight line on semi-log paper. Both equations underestimate the plastic strain at high stresses. This can be empirically corrected in various ways.

Although Eq. (9) was derived by assuming a linear rate of crack growth (in terms of depth) it can be shown that the same equation will be obtained regardless of the shape of the crack growth curve, provided that the curves are affine. (In Refs. 8 and 9 the author derived the equation by using an exponential function of cycles.) This point will be treated more fully in connection with variable-amplitude loading.

A modified (non-affine) theory developed in Ref. 8 can be used to provide a good fit up to the ultimate tensile stress. The failure equation is

$$N = \frac{K - B \log(\Delta\sigma)}{(\Delta\sigma)^x} \tag{10}$$

4. THE ENDURANCE LIMIT EFFECT

If a value of cyclic stress exists below which no cyclic plastic strain occurs, the theory equation can be modified in either of two ways. Fig. 3 shows a case in which the cyclic plastic strain is assumed to be a function of $(\Delta\sigma - \Delta\sigma_o)$, where $\Delta\sigma_o$ is the endurance limit. Eq. (9) then becomes

$$N = \frac{K}{(\Delta\sigma - \Delta\sigma_o)^x} \tag{11}$$

Fig. 4 shows the behavior of a material which has a sharp yield point, such as found in low-carbon steels. In this case Eq. 9 is used for stress increments greater than $\Delta\sigma_o$. The S-N diagram is cut off at $\Delta\sigma_o$.

5. EFFECTS OF MEAN STRESS

It is known that a compressive mean stress tends to increase fatigue life at a given cyclic stress, while a tensile mean stress tends to lower it. This suggests that the primary phyrical action is connected with the unbonding mechanism. With this interpretation, mean stress may be thought of as changing the value of C_4 in the rate-of-crack-growth equation (Eq.(4).

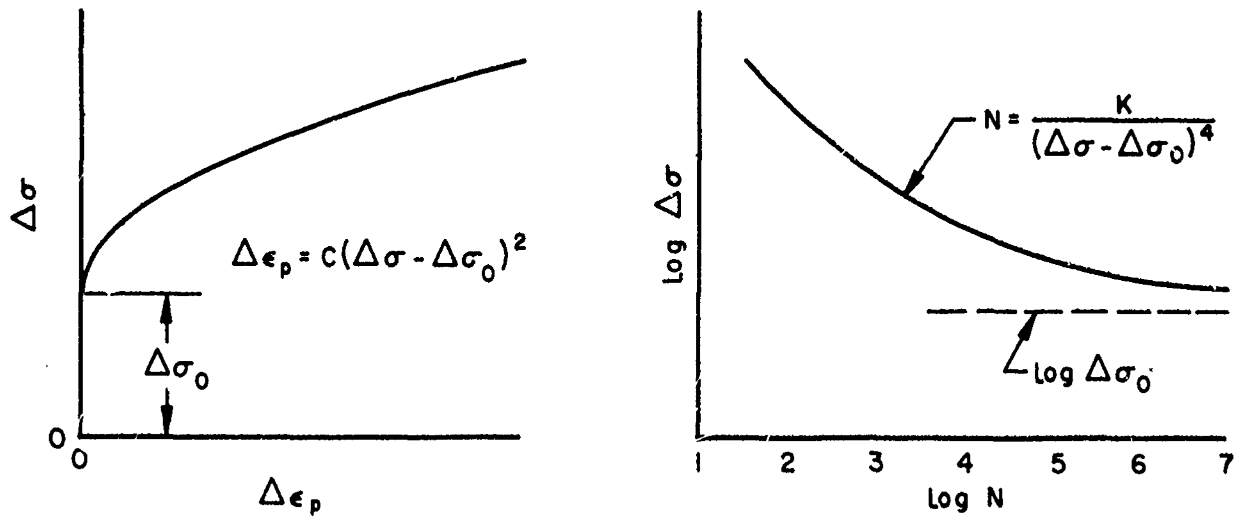

Fig. 3 — Asymptotic type of endurance limit

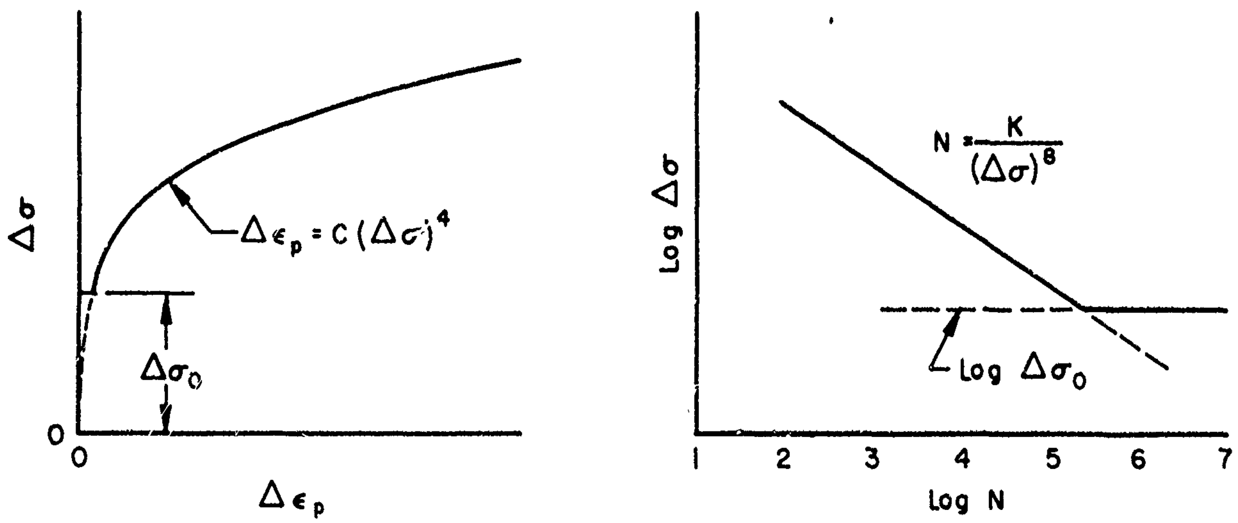

Fig. 4 — "Cut-off" type of endurance limit

Let k_m represent a multiplying factor for C_4 such that $k_m = 1.0$ when $\sigma_m = 0$. Then the crack growth equation becomes

$$\frac{da}{dn} = k_m \, C_4 \, \Delta\epsilon_p \tag{12}$$

where $k_m = f\ (\sigma_m)$.

Then Eq. (7) becomes

$$N = \frac{k_m \, K_1}{(\Delta\epsilon_p)^2} \tag{13}$$

Equation (9) becomes

$$N = \frac{k_m \, K_2}{(\Delta\sigma)^x} \tag{14}$$

If k_m is linear with mean stress, the effect on the S-N diagram will be to displace it parallel with itself, on the log-log graph. The amount of displacement will be proportional to σ_m. This is approximately confirmed by experiment as shown in Fig. 5, except at very high values of mean stress. At such high values the maximum stress in the cycle exceeds the value for which the power law is valid; consequently the equations cannot be expected to apply.

Findley (Ref. 10) has derived formulas by assuming a linear function for k_m and solving for the orientation of the slip plane on which the combined effects of shear stress and mean stress are a maximum. For design work, the effects of mean stress are usually found from tests.

6. EFFECTS OF COMBINED STRESSES

For combined stresses we apply the previously stated concept that the cyclic plastic strain in a crystal is controlled by the state of cyclic plastic strain in its general vicinity. This permits the use of theories of plasticity in which the effects of shear stresses are integrated over the volume of material. Virtually all such theories are based on the premise that plastic strain is caused by shear stresses.

Equation 8 can now be written, for combined stresses, as

$$\Delta\epsilon_p = f\ (\Delta\tau_{eff}) \tag{15}$$

where $\Delta\tau_{eff}$ = an effective cyclic shear stress representing the combined effects of all cyclic shear stresses in the aggregate.

If the octahedral shear stress is used as the effective stress, the effective cyclic shear stress can be expressed as an equivalent uniaxial stress, in the form

$$\Delta\bar{\sigma} = \frac{1}{\sqrt{2}} \sqrt{(\Delta\sigma_1 - \Delta\sigma_2)^2 + (\Delta\sigma_2 - \Delta\sigma_3)^2 + (\Delta\sigma_3 - \Delta\sigma_1)^2} \tag{16}$$

where $\Delta\sigma_1$ = principal cyclic stress, etc.

The effects of normal stress can be included by using the octahedral mean normal stress as a basis. This is the average of the three principal mean stresses:

$$\sigma_{OCT} = \frac{\sigma_1 + \sigma_2 + \sigma_3}{3} \tag{17}$$

An "equivalent mean stress" can be introduced such that it will have the correct value for simple uniaxial loading:

$$\bar{\sigma}_m = \sigma_{m_1} + \sigma_{m_2} + \sigma_{m_3} \tag{18}$$

These values of $\Delta\bar{\sigma}$ and $\bar{\sigma}_m$ can be used in place of $\Delta\sigma$ and σ_m in the foregoing equations, to provide for combined stresses.

Sines (11) has made a thorough study of experimental data for many types of combined loading. He proposed a failure formula that can be expressed in terms of the above parameters as follows

$$\Delta\bar{\sigma} = \Delta\sigma_a - \alpha\,\bar{\sigma}_m \tag{19}$$

where $\Delta\bar{\sigma}$ = equivalent failure stress in combined loading (Eq. 16)
$\Delta\sigma_a$ = failure stress for completely reversed uniaxial loading, at a given value of N
α = influence factor for mean stress
$\bar{\sigma}_m$ = equivalent mean stress (Eq. 18)

In this equation the effects (of combined stresses) on the cyclic plastic strain are contained in the term $\Delta\bar{\sigma}$ and the effects of normal stresses are given by the term $\alpha\bar{\sigma}_m$.

The work of Findley, Sines, and others indicates that practical methods of sufficient accuracy can be developed along these lines. The differences in the various methods are of a minor nature.

7. STRESS CONCENTRATION EFFECTS

Notch effects in fatigue should be interpreted in terms of strain instead of stress. Using the foregoing methods, we can investigate the cyclic plastic strain at the root of a notch, as affected by the nominal stress level, notch geometry, etc. Pian and D'Amato (Ref. 2) have done some interesting experimental work along these lines. Some of their conclusions are quoted below.

> A series of notched specimens were fatigue-tested by cycling between zero and a maximum load. Three specimen types were considered with elastic stress concentration factors of 2.0, 2.5, and 4.0. From the results of these tests the following observations and conclusions are given:
>
> (1) A linear relationship exists between the cyclic range of strain measured at the edge of the notch and the number of cycles to failure when these two variables are plotted using log-log scales.
>
> (2) When compared on the basis of range of strain at the edge of the notch, the specimens with the notch factor of 2.0 had no significant difference in fatigue behavior from the specimens with a 2.5 notch factor.
>
> (3) The strain range and the maximum strain at the edge of the notch did not change during cyclic loading until a relatively few cycles before failure when a marked increase in the strain range was noted in most specimens.

Item (1) appears to justify the use of the foregoing basic theory of controlled cyclic strain in cases having stress-concentration effects.

Item (2) indicates that the elastic stress concentration factor is <u>not</u> the proper parameter to use in dealing with fatigue. We need, instead, a "strain concentration factor," which includes the effects of combined loading at the root of the notch. The <u>depth</u> of notch probably plays a more important part than the root radius, in fatigue. "Size effect" (as measured by Peterson's notch sensitivity factor) is an indication that we are not using the proper parameters in evaluating the notch factor. The strain <u>gradient</u> at the root of the notch is important in size effects.

Fortunately, it is often possible to make fatigue tests of parts containing particular notch effects. This is the best way to obtain reliable engineering information, provided that the models are full-scale.

Another factor in the fatigue of notched specimens is the possibility of inducing <u>residual stresses</u>. The main concern of the designer should be to make sure that unfavorable (tensile) residual stresses are not present. The improvement of fatigue strength by the deliberate production of compressive residual stresses is utilized in various surface treatments (e.g. shot-peening). The designer must be sure that such effects are not

"washed out" during service, if he is to utilize them to obtain higher allowable stresses.

8. EFFECTS OF VARIABLE AMPLITUDE LOADING

The method often referred to as Miner's method was first proposed by Palmgren (Ref. 12) in 1924. In 1937 Langer (Ref. 13) derived the basic cumulative damage equation. In 1945 Miner (Ref. 14) conducted tests which gave a reasonably good verification of the method. Although several other methods have been proposed, the cumulative damage equation remains our most useful tool in dealing with loading of variable amplitude. It is therefore of interest to supply a theoretical basis for the method and to clear up some of the misconceptions that appear to have developed. It will be shown that the method is more general and more accurate than commonly supposed.

Following Langer, we first assume that the crack growth curves for different stress levels of constant amplitude are affine, i.e. they all have the same shape. This can be expressed by the dimensionless "damage" equation

$$\frac{a}{a_{cr}} = f\left(\frac{n_i}{N_i}\right) \tag{20}$$

where

a = crack area (or "damage")
a_{cr} = critical crack area, defining failure
n_i = number of cycles at $\Delta\sigma_i$ (constant amplitude)
N_i = cycles required to produce a_{cr} at $\Delta\sigma_i$.

Figure 6 shows assumed crack growth curves for three different stress amplitudes.

Figure 7 shows the same curves reduced to a dimensionless ("one-one") basis by means of Eq. 20. This approach was used by Newmark (Ref. 15), who plotted "damage" against n_i/N_i. He also included non-affine curves and discussed overstressing and understressing effects. His use of an arbitrary linear curve as a basis might have added to the popular misconception that the cumulative damage equation requires a linear rate of damage; however, he did not intend to imply this, as his paper clearly shows.

For variable amplitude loading the stress levels $\Delta\sigma_i$ are changed during the test. Equation 20 must then be written as follows:

$$\frac{a}{a_{cr}} = f\left(\sum \frac{n_i}{N_i}\right) \tag{21}$$

at failure $a/a_{cr} = 1$. Then, on Fig. 7,

$$\sum \frac{n_i}{N_i} = 1.0 \tag{22}$$

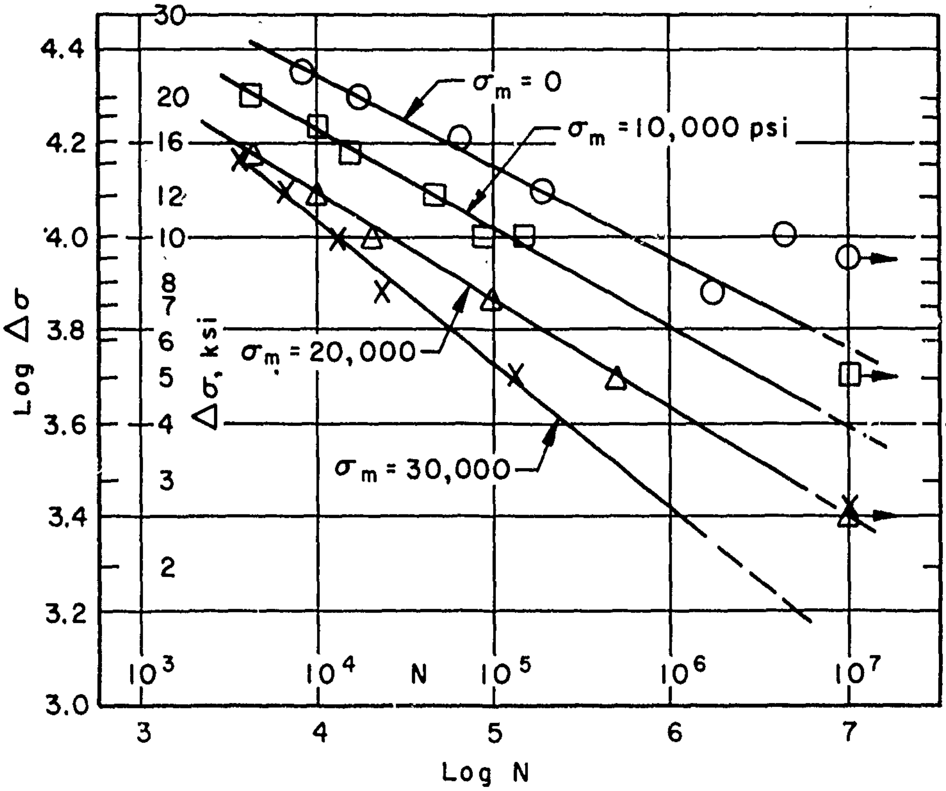

Fig. 5—Effects of mean stress (NACA TN 2389)
(notched 7075-T6 alum alloy, $K_t = 4.0$)

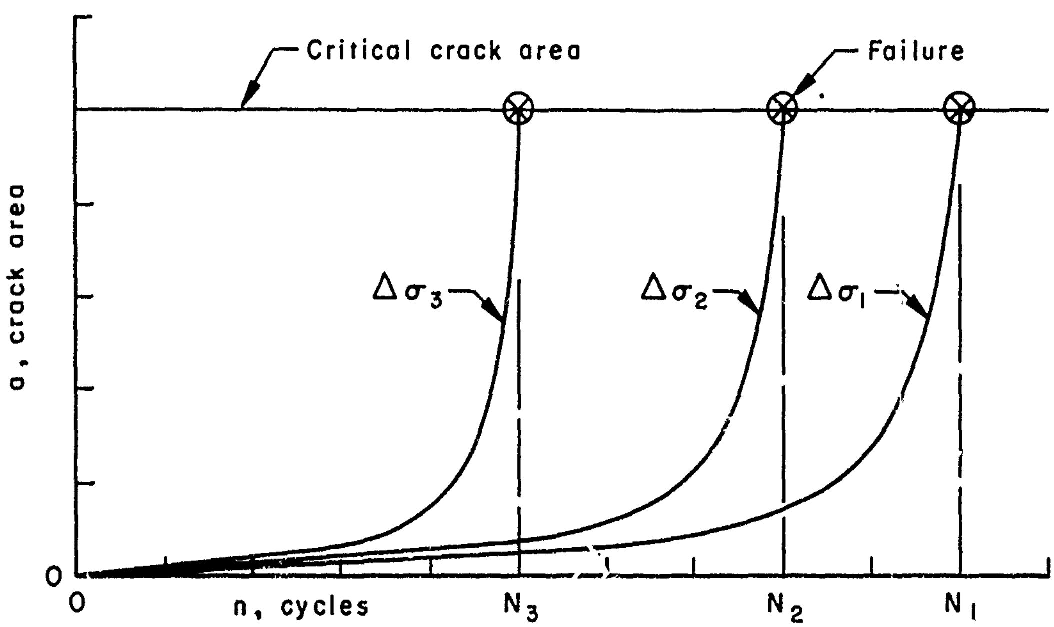

Fig. 6—Affine crack growth curves (schematic)

This is the familiar cumulative damage equation. It can be used either for controlled stress or controlled strain. In the latter case the values of N_i are determined from the ϵ-N diagram, or from Eqs. (6) or (7). (See discussion of Coffin's paper (Ref. 3) by Murphy.)

The cumulative damage equation (22) is not restricted to linear crack growth (or damage) curves. This was shown by Langer in his derivation of the equation. However, Miner used a different derivation which incorrectly implied that the damage curve must be linear. Many writers incorrectly refer to this feature as a limitation of the method. Although the "cycle ratios" are added linearly, the accumulation of damage need not be linear.

Equation (22) also does not place any restrictions on the order in which the various stress (or strain) levels are applied. It is well known, however, that order does in some cases have an appreciable effect on life. These effects can be explained by strain-hardening or strain-softening phenomena, and probably involve the pinning or generation of dislocations.

Two important and related questions arise in applying the cumulative damage theory:

(a) Can the S-N diagram (for stresses of constant amplitude) be used directly in determining values of N_i?

(b) What endurance limit effects can be expected under loading of variable amplitude?

Langer suggested that the values of N_i be obtained from a "damage curve," instead of from the conventional S-N diagram. Freudenthal (Ref. 16) discusses this matter in detail and concludes that the conventional S-N diagram "cannot provide the basis for the prediction of fatigue life under randomly-applied amplitudes, no matter what type of damage accumulation law is used." He suggests that a fictitious S-N diagram be established for use in connection with random loading. This diagram would have lower values of $\Delta\sigma$ than those found in conventional tests.

The lowering of the S-N diagram, for random loading, can be explained by dislocation theory. Under constant stress amplitudes the density of "free" (reversible) dislocations tends to decrease (i.e. some dislocations become "pinned"). This reduces the cyclic plastic strain and thereby increases fatigue life. This is reflected in the conventional S-N diagram. When intermittent high stresses are applied, some of the pinned dislocations may be released and even more free dislocations may be generated, thereby increasing the cyclic plastic strain and reducing life.

These arguments apply with greatest force to the high cycle and of the S-N diagram. In this region we can expect to obtain the greatest benefit from strain-hardening effects, under constant amplitude loading. The effects of random loading will therefore be to lower this end of the diagram, involving the endurance limit.

In using cumulative damage theory, the endurance limit should be defined as the limiting value of cyclic stress below which no crack growth (or damage) is produced in random loading. Its value will usually be lower than the endurance limit found in conventional tests. For analysis and design purposes the above factors can be taken into account by modifying the conventional S-N diagram along the lines suggested by Freudenthal (Ref. 16), as shown in Fig. 8. The reduction factor is arbitrarily increased as the stress decreases. The endurance limit is lowered. A sharp "cut-off" at the endurance limit should be used.

In using Eq. (22) the value of N_i for stress increments less than the endurance limit $(\Delta\sigma_o)$ is taken as infinity, i.e. the cycle ratio n_i/N_i becomes zero for these stresses.

The effects of mean stress, combined stresses, etc. can be provided for by applying the principles and methods discussed in earlier sections. If the mean stress changes during a flight mission (for example, when bombs are dropped), the effect on rate of crack growth is revealed by the displacement of the S-N diagram, as previously shown (Fig. 5). This changes the values of N_i. Using the modified value of N_i has the same effect as changing the crack growth rate by the factor k_m in Eq. (14).

It has been found that notched specimens sometimes have a higher life under random loading than that predicted by the cumulative damage theory. This is undoubtedly caused by the building up of favorable residual stresses in the critical regions, thereby reducing the cyclic plastic strain. For design purposes, such effects should generally be ignored, because of the possibility that a particular structure might experience a less favorable loading history.

Fig. 9 summarizes some recent tests of notched specimens under random loading, by Fralich (Ref. 17). The above-mentioned influences can be observed. The cumulative damage equation was used in the theory which was developed by Miles (See Ref. 17).

9. PREDICTION OF FATIGUE LIFE

In using the cumulative damage theory to predict fatigue life it is necessary to have a typical loading spectrum, such as shown in Fig. 10. This may represent the entire expected life or it may be a small "block" of cycles. In any case, we can determine the corresponding value of $\sum n_i$. The predicted life can be found from the equation

$$N_p = \frac{\sum n_i}{\sum (n_i/N_i)} \tag{23}$$

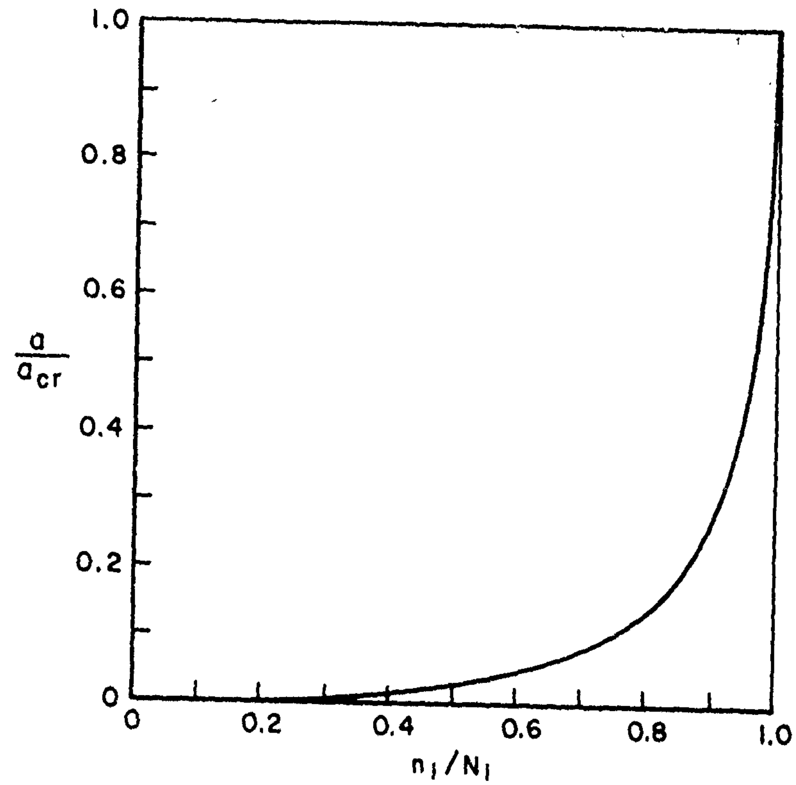

Fig. 7—Dimensionless ("one-one") crack-growth curve

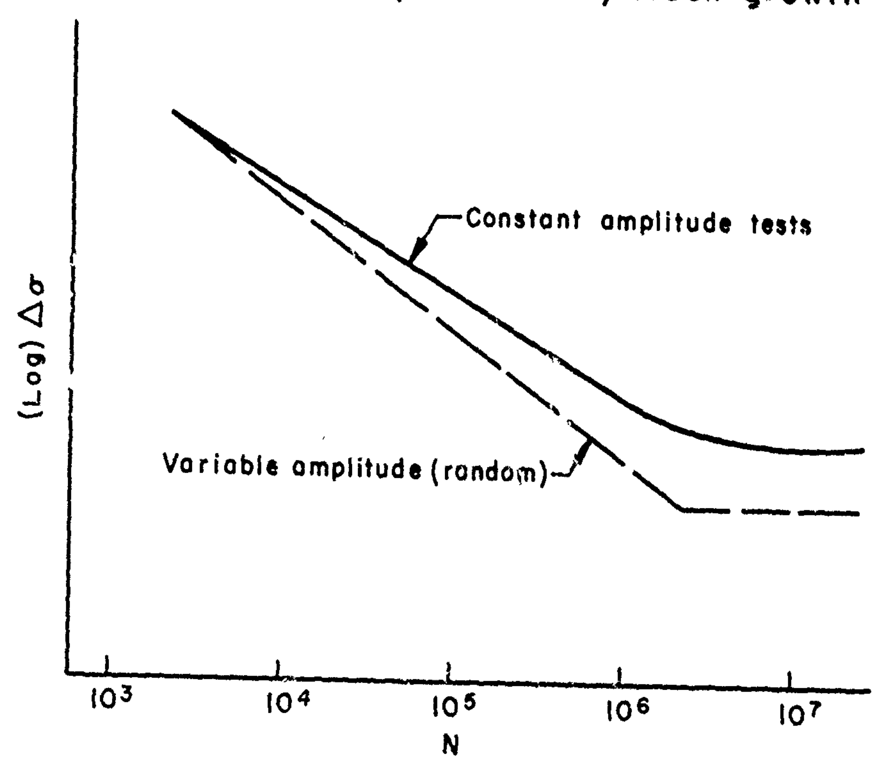

Fig. 8—Reduction of S-N diagram
for random loading (schematic)

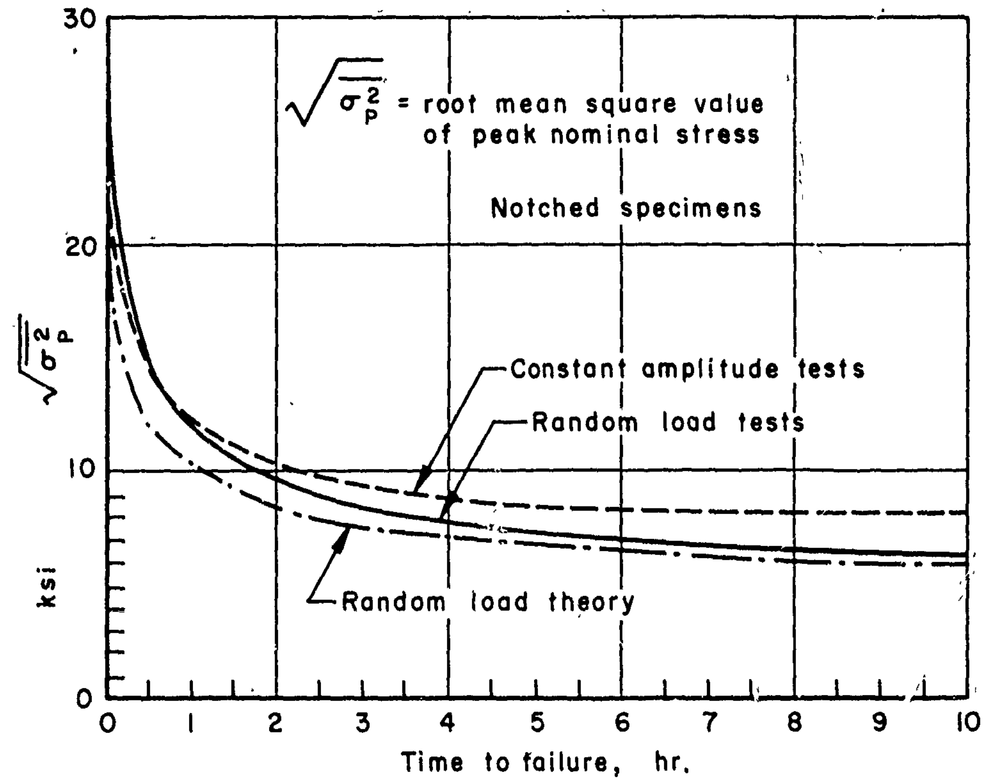

Fig. 9 — Test results for random loading and constant amplitude (NASA memo 4-12-59 L)

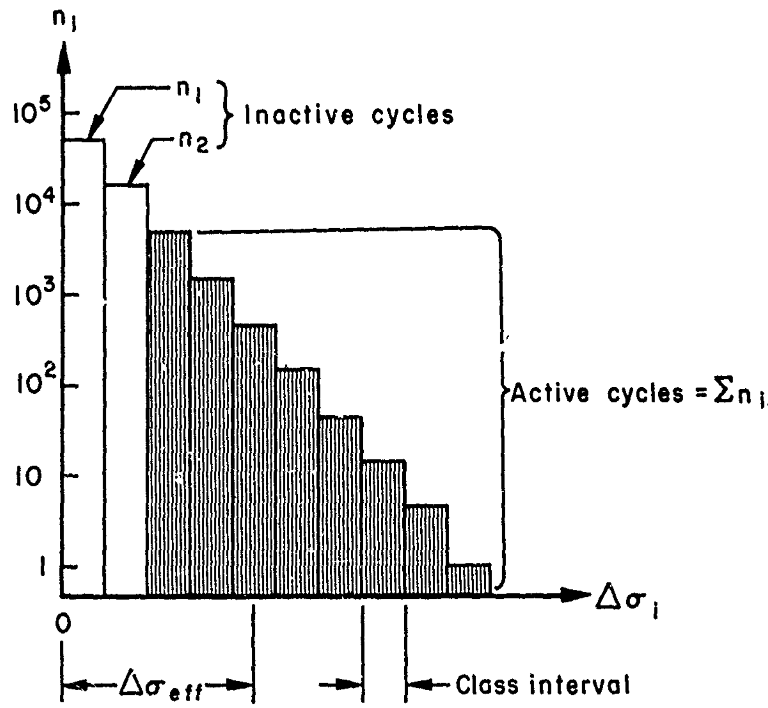

Fig. 10 — Stress spectrum (schematic)

Fig. 10 shows two "inactive" class intervals of $\Delta\sigma_i$, representing stresses below the endurance limit. The crack growth curve for these stresses ($a/a_{cr} = 0$) does not belong to the affine family and it is therefore theoretically incorrect (but mathematically satisfactory) to include the ineffective values of n_i in Eq. (23). The correct procedure is to calculate the predicted life in terms of "active" cycles, then increase this life in the proper ratio to determine the total value of N_p. (If a time basis is used there is no problem.

In comparing theory with test results, the total number of cycles must be used. In calculating the theoretical life it is necessary to select some value for the endurance limit and to eliminate the values of n_i for all stress increments below this limit. Since the smaller stress increments usually appear in very large numbers, in the spectrum, the choice of the endurance limit can have a strong effect on the comparison between theory and experiment. All such comparisons should be carefully examined to see if proper evaluation of endurance limit effects has been made.

To obtain information on endurance limit effects it is highly desirable to make spectrum tests in which one or more of the lowest class materials of $\Delta\sigma_i$ are omitted, along with control specimens subjected to the full spectrum loading.

10. ALLOWABLE STRESS FOR VARIABLE AMPLITUDE LOADING

In the issue of the Journal of Applied Mechanics in which Langer's theory (Ref. 13) appeared there appeared also a paper by Clinedinst (Ref. 18) which contains an expression for the effective stress for variable amplitude loading. This was based on the power formula for the S-N diagram, given by Eq. (10). A similar equation was derived by the author from the theory of Ref. 8.

The effective stress ($\Delta\sigma_e$) is defined as that value of stress increment which, if applied at constant amplitude, will give the same life (N_p) as the variable-amplitude loading.

For affine crack-growth curves, the effective stress may be obtained by predicting the life (N_p) from Eq. (23) and entering the S-N diagram at this value. (The modified diagram for variable-amplitude loading should be used.)

As an illustration, the formula for effective stress will be derived by this method, using Eq. (10) as a basis for the S-N diagram.

By definition, we may write Eq. (10) in terms of the effective stress increment ($\Delta\sigma_e$) and the predicted life (N_p).

$$N_p = \frac{K}{(\Delta\sigma_e)^x}$$

From Eq. (23)

$$N_p = \frac{\sum n_i}{\sum (n_i/N_i)}$$

But,

$$N_i = \frac{K}{(\Delta\sigma_i)^x}$$

Equating these two expressions for N_p and solving for $\Delta\sigma_e$ gives

$$\Delta\sigma_e = \left[\frac{\sum n_i \ (\Delta\sigma_i)^x}{\sum n_i}\right]^{\frac{1}{x}} \tag{24}$$

This agress with Clinedinst's method and with the author's derivation of Ref. 8. The advantage of determining the effective stress from the predicted life (Eq. 23) is that the S-N curve can be used directly, regardless of its shape. Furthermore, it is not necessary to know anything about the shape of the crack-growth curves, except that they must be affine.

Since all stress increments below the endurance limit ($\Delta\sigma_o$) are considered to be inactive, the method of computing the effective stress by entering the S-N diagram at N_p requires that N_p correspond to the active stress increments only. Otherwise there is a violation of the assumption that the values of N_i are reached through affine crack-growth curves.

It can easily be shown that the use of the total predicted life, N_p, is erroneous in calculating the effective stress, if there are any inactive stress increments in the spectrum. For example, an elementary two-load spectrum can be assumed, in which one of the stress increments is above the endurance limit, the other below it. Using the active cycles only, the effective stress will be found to be the higher of the two stress increments, which is correct. Using total cycles, the predicted life may fall in the endurance limit region. In this case the effective stress is erroneously found to be the endurance limit.

By working with effective stress, instead of life, it is possible to specify a factor of safety on stress. The method is illustrated in Fig. 11. The "allowable stress" is found by entering the S-N diagram at the required value of life. (This must contain "active" cycles only.) The "applied stress" is the effective stress found by entering the S-N diagram at the predicted life, N_p. The "factor of safety" is, as usual, the ratio of allowable to applied stress. In this method (proposed in Ref. 19) the factor of safety for fatigue is based on the stress increment, $\Delta\sigma$, rather than on maximum stress. This appears logical in view of the relatively small influence of mean stress on the allowable stress increment.

One important advantage of working with effective stress based on active stress increments is that the effective stress and the allowable stress both occur at stress levels considerably above the endurance limit, where scatter is much less than in the endurance limit region.

11. STATISTICAL EFFECTS AND DESIGN CRITERIA

The well-known scatter in fatigue life, as obtained from tests, has been a discouraging factor in attempts to predict fatigue life. For structures that must have very long life, it is virtually impossible to predict the fatigue life with reasonable accuracy. For example, it is entirely meaningless to speak of life prediction for a rotating shaft that operates under constant amplitude loading and which has been designed by using the endurance limit as the allowable stress, together with a factor of safety. Yet it is easy to determine the allowable cyclic stress for such a shaft. The very thing that makes life prediction meaningless (flat S-N diagram) also makes the stress analysis simple.

Although this reasoning cannot be applied directly to an aircraft structure the fact is that for "long-life" structures we are much closer to the rotating-shaft example than is generally realized. However, instead of designing for "infinite" life it is necessary to design for a "very long" finite life. Just how long this life should be is a matter for detailed study, involving the service or mission of the aircraft, availability of data on applied loadings, etc. It will only be noted here that a high degree of conservatism can be used, with little weight penalty, in establishing life requirements.

Assuming that the required life has been established and reduced to "active" cycles, the only remaining question is the selection of a factor of safety, or alternatively, the reduction of the allowable fatigue stress to provide for scatter in material properties, dimensional tolerances, etc. Here the problem becomes similar to that for "static" design. There is scatter in yield stress, ultimate stress, column stresses, etc. In fact the scatter in the buckling stress of a thin-walled shell is far greater than that found in fatigue stress (at a given life). For this reason the author believes that the rational establishment of factors of safety, based on probability theory and statistical data, is a matter that should be studied for all loading conditions, including both static and "spectrum" loads.

Fig. 11 indicates schematically how the factor of safety is related to scatter. Lines of constant probability (P-lines) are shown. If these are well established for the material and shape of a part, by spectrum tests, the factor of safety can be adjusted accordingly, using probability theory. However, the factor of safety also covers other types of scatter such as dimensional tolerances, surface conditions, etc. (See Freudenthal's remarks in Ref. 20.)

In Ref. 21, Lundberg proposed methods of fatigue analysis based on the concept of an arbitrary (very small) probability of fatigue failures. In

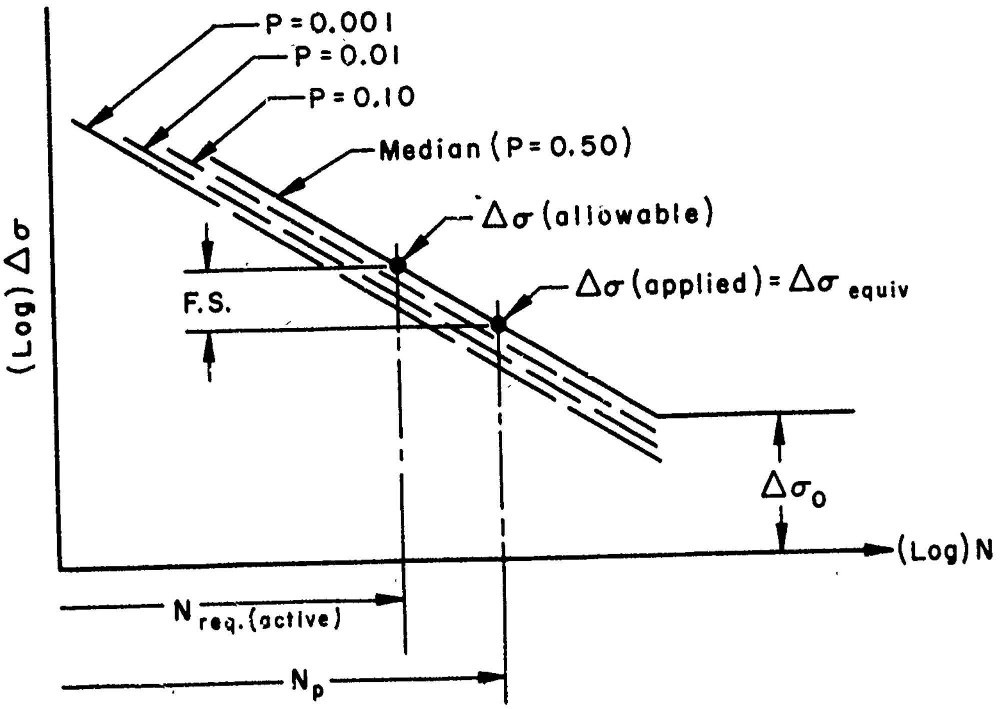

Fig. 11 — Applied and allowable fatigue stresses

Ref. 22 he made specific suggestions of a practical nature on how such fatigue analyses could be carried out. In general the author agrees with the objectives and methods proposed by Lundberg, with the exception that it may not be necessary or practical to work with probability theory directly in designing for fatigue. (Some of the differences in approach are summarized in Ref. 23.)

12. CONCLUSIONS AND RECOMMENDATIONS

In evaluating methods of analysis it is most important to decide whether the immediate objective is to prevent fatigue failures or to predict fatigue life. Methods that are entirely satisfactory in the first case are likely to be quite inadequate in the second case. Much of the pessimism that one encounters in discussions of fatigue analysis is the result of looking at the picture from the wrong direction.

For example, nearly all the tests made to evaluate the cumulative damage theory have been reported in terms of life only. Comparing predicted life with test life often gives discouraging results. But if the same results are compared on the basis of effective stress, the picture is much more encouraging. This was demonstrated in Ref. 8 (Supplement), using some of the earliest results from variable amplitude tests (Ref. 24). In comparing test results with the "linear" cumulative damage theory it was found that the life ratios ranged from 0.38 to 9.27. But the corresponding range in stress ratios was from 0.77 to 1.12 (for sinusoidal variation of amplitude). Applying the techniques of the present paper to more recent tests made by Rey (Ref. 25) gives similar results. Variations between the predicted equivalent stress and that found in tests were less than ± 3 percent in almost all cases.

It would be difficult to find any other mode of structural failure in which the agreement between theory and tests is much better than this. We can conclude, therefore, that the cumulative damage theory ("Miner method") is satisfactory for design purposes. The only refinement needed appears to lie in the modification of the S-N curve to be used as a basis, particularly in the endurance limit region (See Sec. 8). Even this is likely to be relatively small in terms of stress.

The major problem in setting up criteria seems to lie in the philosophy of safety, or airworthiness. Various interpretations have been proposed by Freudenthal (Ref. 20), Lundberg (Ref. 21 and 22), Tye (Ref. 26), Hoff (Ref. 27), Van der Neut (Ref. 28), to mention only a few. Studies of this type should include static loading criteria as well as fatigue. For example, the overall efficiency of a wing structure might be improved by transferring some of the structural material from the compression to the tension side, i.e., by providing less material for static buckling strength and more for fatigue strength, at no overall increase in weight.

The theory presented in this paper suggests that safety in fatigue should be obtained by designing in such a way that fatigue cracks will

remain in the slow growth stage during the entire service life of the structure. It follows that all static strength requirements should be fulfilled at the end of the service life, as well as at the beginning, since very small fatigue cracks have no appreciable effect on static strength.

Nothing very encouraging can be said about the prediction of fatigue life. Although the "average" life might be roughly predicted for an entire class of vehicles, this is not satisfactory for a particular vehicle. The ultimate objective should be to design structures so that there is no need to predict their life. When such a need arises it will be necessary to take into account the large scatter in life that is a characteristic of fatigue failures. The uncertainty can be reduced somewhat by the use of monitoring devices, warning devices, etc., the discussion of which is beyond the scope of this paper.

The following detailed recommendations are made:

(a) Tests should be made in which specimens are subjected to spectrum loadings that are identical except for the omission of one or more of the smallest stress levels. This will establish the endurance limit values for variable amplitude loading.

(b) Test data for variable amplitude loading should be analyzed in terms of effective stress, as well as life. Existing data should be reviewed on this basis.

(c) Systematic studies should be made to determine the relative amounts of additional structural weight corresponding to different criteria, different lives, different materials, etc. (See Ref. 19 for example.) Such analyses should be extended to include the effects on overall performance, economics, payoff, etc., for various classes of vehicles.

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CUMULATIVE DAMAGE THEORIES

by

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Aircraft structural components are subjected to repeated loads of varying amplitudes and frequencies. Accordingly, fatigue-life estimates must be based on some criterion for cumulative damage.

Theories for cumulative fatigue damage have been sought for more than 30 years. An early suggestion, the summation-of-cycle ratio, has been widely used in design. Since numerous experiments have shown this summation inadequate for many situations, more elaborate methods have been suggested. Several of these will be discussed briefly.

Most relations so far advanced have one or more of the following limitations: (1) no physical mechanism is clearly defined so the relation contains factors identifiable with concepts useful in design; (2) too many experimental data are required for engineering application; (3) mathematical calculations are cumbersome.

Presently available methods are, for many applications, as accurate as warranted---in view of other uncertainties in the fatigue problem. As these other uncertainties are resolved, a better theory of cumulative damage will be needed. It is suggested that such a better theory should be based upon a realistic account of the physical mechanism or mechanisms involved.

INTRODUCTION

An important objective in design evaluation of an airframe is to insure no failure within a scheduled service lifetime. Allowance for uncertainties in several areas of information make this objective a high challenge.

The designer of an airframe component usually has available some information about the missions for which the vehicle is scheduled and corresponding estimates of load spectra to be anticipated for the component. There may be considerable uncertainties in these estimates. He also has, from theory or experiment, estimated values of critical stresses. Since fatigue is responsive to localized stresses, he requires stress-concentration factors or fatigue test data on the component. For either, the best values that can be obtained usually have further uncertainties. Thus in the stress spectrum available for analysis of fatigue lifetime, there may be a very significant total margin of uncertainty.

Then, there is needed information about the fatigue properties to be expected in the material of which the part is to be made. Fatigue data usually show considerable scatter, so the available estimates contain further uncertainties in design information. Moreover, the usual fatigue data are from laboratory tests at various constant-stress amplitudes. For estimation of lifetime under varying-stress amplitudes, some theory of cumulative damage is necessary to relate these to the design stress spectra.

The need for design allowance for cumulative fatigue damage was recognized at least 35 years ago. Since then, a great deal of effort has been expended and progress has been made toward understanding the behavior of materials in this respect. A major objective of this paper is to review damage theories which have been proposed and to indicate the present understanding of cumulative damage in structural materials.

However, application of theories of fatigue-damage behavior of materials should be viewed in the light of all uncertainties in the whole problem of evaluating the expected lifetime of a structural component. Accordingly some factors in design application of available theories will also be discussed with the objective of providing an over-all appraisal of the status of knowledge of cumulative damage in fatigue.

THE SUMMATION-OF-CYCLE RATIOS

Palmgren (Ref 1) suggested a simple approach which Miner (Ref 2) discovered independently and applied to aircraft problems. This method, and some of the nomenclature useful in subsequent discussion, may be described with reference to Figure 1. Suppose a specimen (a material test piece or a structural part) has been subjected to n_i cycles of stress (or load) amplitude at which it could last N_i cycles. Then the Palmgren-Miner hypothesis is that the damage incurred is

$$D_i = n_i/N_i \quad . \tag{1}$$

The quantity n_i/N_i is called the "cycle ratio at stress S_i". If the repeated load is continued at constant-stress amplitude, until $n_i = N_i$,

$$D_i = 1$$

and failure occurs. Now, it is supposed that when the stressing involves many levels of stress amplitude, the total damage is

$$D_T = \sum_i D_i = \sum_i n_i/N_i \quad , \tag{2}$$

and that failure will still occur when

$$D_T = 1 \quad . \tag{3}$$

In this case, the total number of cycles of stress (of all levels) is

$$N_T = \sum_i n_i \quad . \tag{4}$$

This theory of the summation-of-cycle ratios has not been justified by any detailed mechanism of the fatigue process. It is rather a heuristic speculation that a single, simple parameter (the cycle ratio) characterizes accumulating fatigue damage. However, this simplicity is a major virtue. No information about material beyond the S-N curve (with proper statistical considerations) is needed for design by this method. Moreover, in many situations, the method affords conclusions as accurate as warranted in view of uncertainties in the S-N curve, in load spectra anticipated, and in detailed stress analysis.

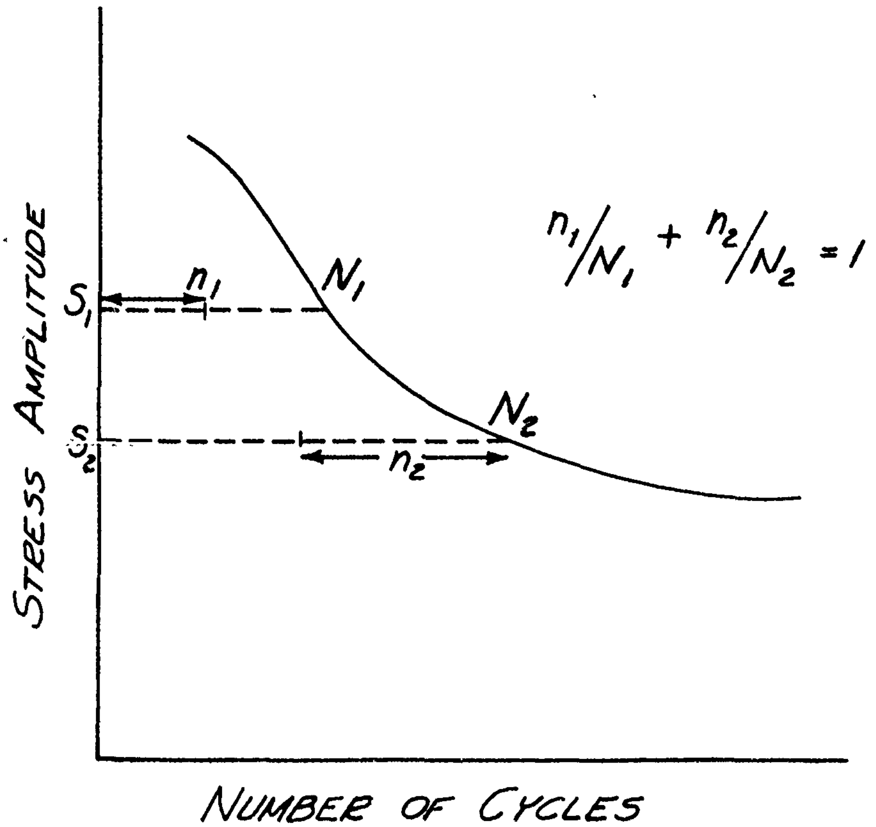

FIGURE 1. SUMMATION-OF-CYCLE RATIOS

EXPERIMENTS AND EMPIRICAL DEVELOPMENTS

A number of investigations (Refs 2 through 17) have shown results of specific tests incompatible with the summation-of-cycle ratios. Figure 2 showns, as an example, results of some two-level rotating-bending tests on specimens of SAE 4130 steel. Damage may be defined as the fractional decrease of life at the second test level on account of previous running at the first level. Hence, on this graph, Equation 2 predicts observed points should fall on the indicated straight line. Not only do the points deviate from this line, but there appears to be an effect of the order of stressing. If the high stress is applied first, points tend to fall above the line; if the low stress is first, points tend to be below the line. Other tests at two or more levels have shown similar disagreement with the summation-of-cycle-ratio hypothesis, and load-spectrum tests have also indicated the inadequacy of Equation 2 to predict total lifetimes.

Richart and Newmark (Refs 5 and 6) outlined empirical formulations based upon the damage rate (in terms of cycle ratio) being considered stress dependent. Figure 3 shows hypothetical curves of this nature and Figure 4 indicates "predictions" from these curves in comparison with data presented previously in Figure 2. The possibility of such a procedure providing better fit to data, including effects of the order of load application, is clear. A major difficulty with this procedure is the need for a very large amount of experimental data for each structural material (and, possibly, for several types of stressing and for several types of stress concentration).

Marco and Starkey (Ref 7) suggested for damage at stress amplitude, S_i,

$$D_i = (n_i/N_i)^{\alpha_i} \quad . \qquad (5)$$

Moreover, they noted appearance of fatigue fractures and suggested "a correlation between stress level and the number of fatigue nuclei which result in independent cracks" as a possible explanation of the variation of α_i with S_i. They further emphasized that an individual specimen used in sequential loading for damage studies has its own S-N curve, which may differ from the average curve from which the N_i are evaluated; thus, in design, a theory of damage must take into account the statistical nature of fatigue data. These ideas have been used in most recent fatigue-damage theories.

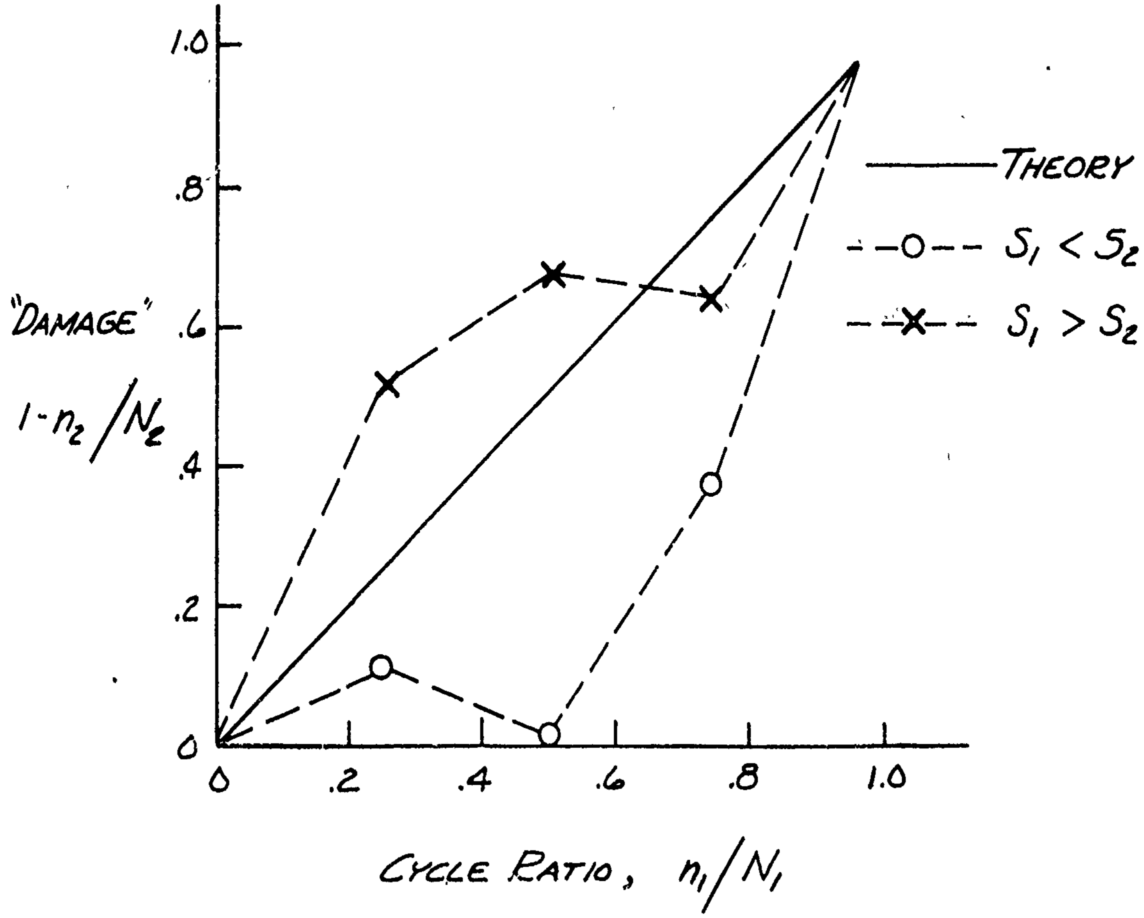

FIGURE 2. RESULTS OF TWO-LEVEL DAMAGE TESTS (REF 17) IN COMPARISON WITH PALMGREN-MINER THEORY

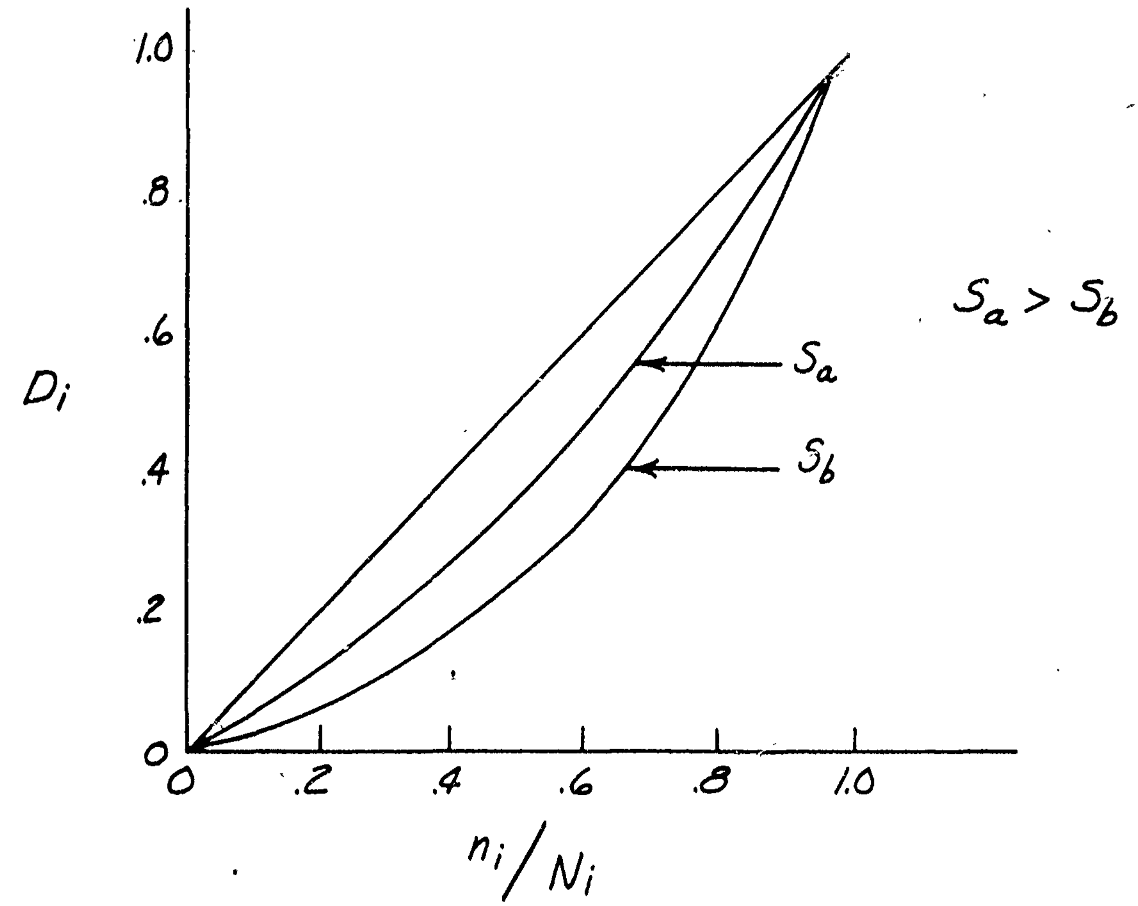

FIGURE 3. HYPOTHETICAL NON LINEAR DAMAGE CURVES VARYING WITH STRESS LEVEL

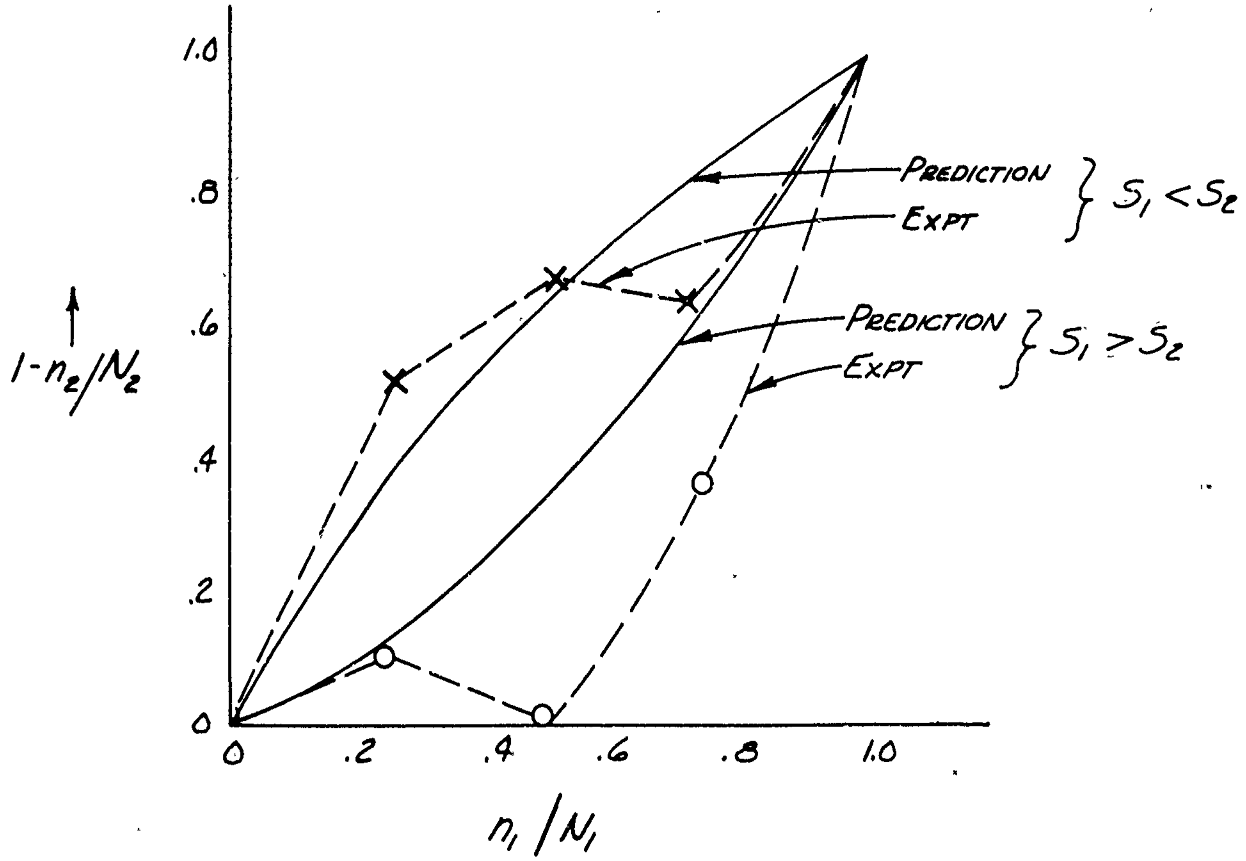

FIGURE 4. PREDICTIONS FROM ASSUMED DAMAGE CURVES (FIG 3) AND EXPERIMENTAL DATA (FIG 2.)

SOME RECENT THEORIES OF CUMULATIVE DAMAGE IN MATERIALS

Within the past 5 years, several "theories" for cumulative damage in structural materials have been suggested. These all include damage rate (in terms of cycle ratio) being stress dependent and generally being higher at higher stress levels. Strengthening by understressing, often ascribed to strain hardening, is considered an additional effect to be added, if pertinent.

Since the approaches and usually the nomenclature of different investigators differ widely, the following brief reviews can only indicate a few features of each suggested theory; references to available papers will permit those interested to obtain more complete information.

Corten and Dolan (Ref 11) have suggested, for damage at a fixed-stress amplitude,

$$D_i = m_i r_i n_i^{\alpha} , \qquad (6)$$

where m_i = number of damage nuclei at the stress level S_i

r_i = coefficient of damage propagation at stress level S_i

n_i = number of cycles at S_i .

To obtain a useful expression for evaluation of damage, some additional assumptions were made: that damage nuclei, once introduced at any stress level, remain and subsequently propagate at any other stress level, and that the rate of propagation is determined solely by the stress level. These assumptions lead to an expression for the total number of cycles in a test at varied stress amplitudes, namely

$$N_T = \frac{N_1}{\sum_i p_i (r_i/r_1)^c} . \qquad (7)$$

The quantity N_1 is the number of cycles a specimen would run continuously at stress level S_1 (the highest in the spectrum). The quantity p_i is

$$p_i = n_i/N_T . \qquad (8)$$

Experiments on the fatigue of wires (of 2024-T4 and of 7075-T6 aluminum alloys, and of hard-drawn steel) implied that, for any

one of these materials,

$$(r_i/r_1)^c = (S_i/S_1)^d \quad . \tag{9}$$

Then, Equation 7 can be written:

$$N_T = \frac{N_1}{\sum_i p_i(S_i/S_1)^d} \quad . \tag{10}$$

This can be used for design if the exponent d is determined experimentally; it "may be obtained most simply from a two-stress-level repeated-block-fatigue experiment of the component or full-size structure".

Henry (Ref 9) suggested for steels having fatigue limits the following relation for damage at stress amplitude level S_i.

$$D_i = \frac{n_i/N_i}{1 + \frac{1 - n_i/N_i}{\gamma_i}} \quad , \tag{11}$$

$$\text{where} \quad \gamma_i \equiv (S_i - E)/E \tag{12}$$

and E is the fatigue limit. Figure 5 shows, for the same data as in Figure 2, a comparison of observed results with calculations from Equations 11 and 12. For some data on a steel at elevated temperatures, the agreement is less good (Ref 17) and, in its present form, this theory is inapplicable to materials not having fatigue limits. A virtue of Henry's approach is that no information beyond the S-N curve and the stress spectrum is required.

Freudenthal (Refs 15 and 20) suggests, on the basis of observed development of slip striations in fatigue, the need for greater allowance for damage at high stress levels. He suggests a formulation which can be expressed in the following notation.

In the terms of Equation 8, the summation-of-cycle ratios can be written

$$N_T = \frac{1}{\sum_i p_i/N_i} \quad . \tag{13}$$

Freudenthal would obtain a reduced total number of cycles, N_R, on account of the greater damaging effect of some of the high stress amplitudes. He suggests

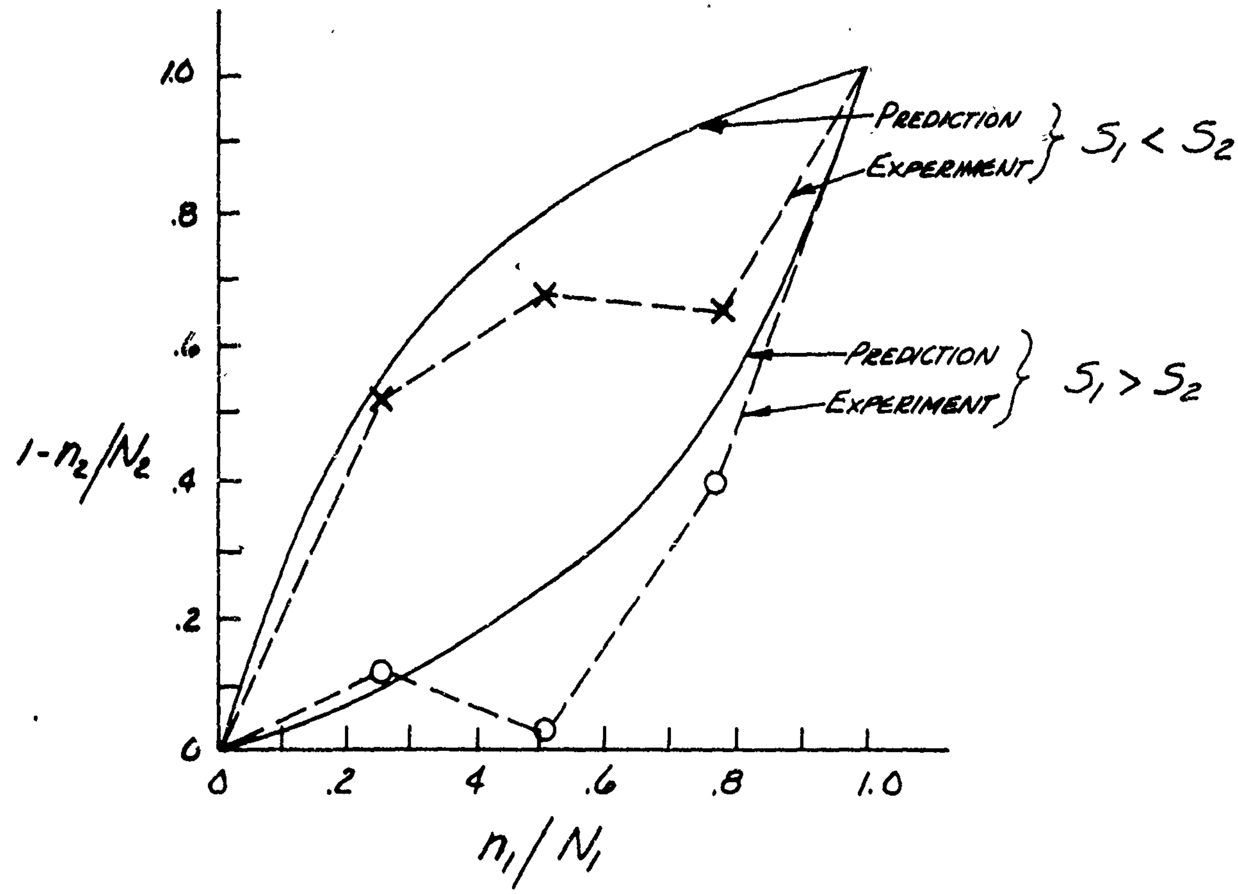

FIGURE 5. PREDICTIONS FROM HENRY'S THEORY AND EXPERIMENTAL DATA (FIG 2.)

$$N_R = \frac{1}{\sum_i \left(\frac{p_i \omega_i}{N_i}\right)^{\alpha}} . \tag{14}$$

Here, ω_i (>1) reduces the cycle ratio remaining at any stress S_i, on account of the relatively greater damage by cycle ratios at all stress levels where $S > S_i$. Unfortunately, this approach requires either much extra experimental information to evaluate the ω_i or additional analysis involving approximations. However, Freudenthal shows, by additional assumptions and analysis, reasonable fit to data from stress-spectrum tests.

In a paper to be presented (Ref 21), the present author has considered a somewhat different approach. Suppose that the fatigue process involves two steps (for example, damage to a stage of crack initiation followed by crack propagation to visible failure).

Let $N_i \equiv$ number of cycles to failure at stress S_i, and

$a_i N_i$ = number of cycles to cracking at S_i, where $a_i < 1$.

Now, assume two conditions:

$$\sum_i \frac{m_i}{a_i N_i} = 1 \text{ , and} \tag{15}$$

$$\sum_i \frac{n_i}{N_i(1 - a_i)} = 1 \quad , \tag{16}$$

$$\text{where} \quad \sum_i (m_i + n_i) = N_T . \tag{17}$$

At least for some instances (Ref 21), test results appear compatible with Equations 15 and 16, providing a_i varies with S_i in a manner suggested by recent experiments on fatigue crack propagation. In general, if S-N curves for cracking and for failure are parallel, or if loading is suitably randomized, Equations 15 and 16 reduce to the Palmgren-Miner summation; otherwise, these equations imply departures from the cycle-ratio summation and effects of order of loading. At present, this approach should be considered speculative. It has the disadvantage of requiring, for design use, more information than the S-N curve. However, information about crack-propagation dependence on stress may be of particular interest in regard to structural components.

Some general comments may be made concerning the various suggested theories in which damage rate is considered stress dependent. All involve calculations significantly more complex than the summation-of-cycle ratios. Nearly all (that by Henry being an exception) involve appreciable experimental work to evaluate the stress dependence. Most of these have been justified by some plausible mechanism; sometimes, however (compare Corten and Dolan's relations with Freudenthal's), similar analytical expressions can result on the basis of rather different assumed mechanisms. Most of the proposed mechanisms (with a possible exception for speculations concerning crack propagation) are difficult to visualize in terms of engineering data readily obtainable on structural components.

Most of these theories make allowance for damage rates higher at higher stresses, but require additional development to include strengthening by understressing or by a few cycles of overstress.

In every instance, a proposed theory has been shown to fit some data more closely than a non-stress-dependent theory. None has been shown to fit all data within significant error. (In some cases, this is rather hopeless---in view of the paucity of information concerning statistical significance.)

CUMULATIVE DAMAGE IN STRUCTURES

There are, understandably, few reported results of fatigue-damage tests upon reasonably complex structural components under conditions such that data have statistical significance. Figure 6 shows the percentage of loss of life of some components after various periods of service. These results indicate the existence of damage that is probably nonlinear in hours of service; they also show rather wide scatter. It is somewhat unreasonable to attempt conclusions from such evidence as to the relative merits of material damage theories. However, as indicated by Christensen (Ref 22), there is a good deal of total evidence of the existence of cumulative damage in structural parts.

There are relatively more data on less complex elements, such as joints (see, for some examples, Refs 23, 24, and 25). Ranges of $\Sigma n/N$, reported by Plantema, for specimens of 2024-T Alclad are shown in the following tabulation. Other instances might be quoted, but general results are: structural joints show similar ranges of $\Sigma(n/N)$ to material coupons, including wide scatter and some trends for order of loading in tests at a few levels. In some instances,

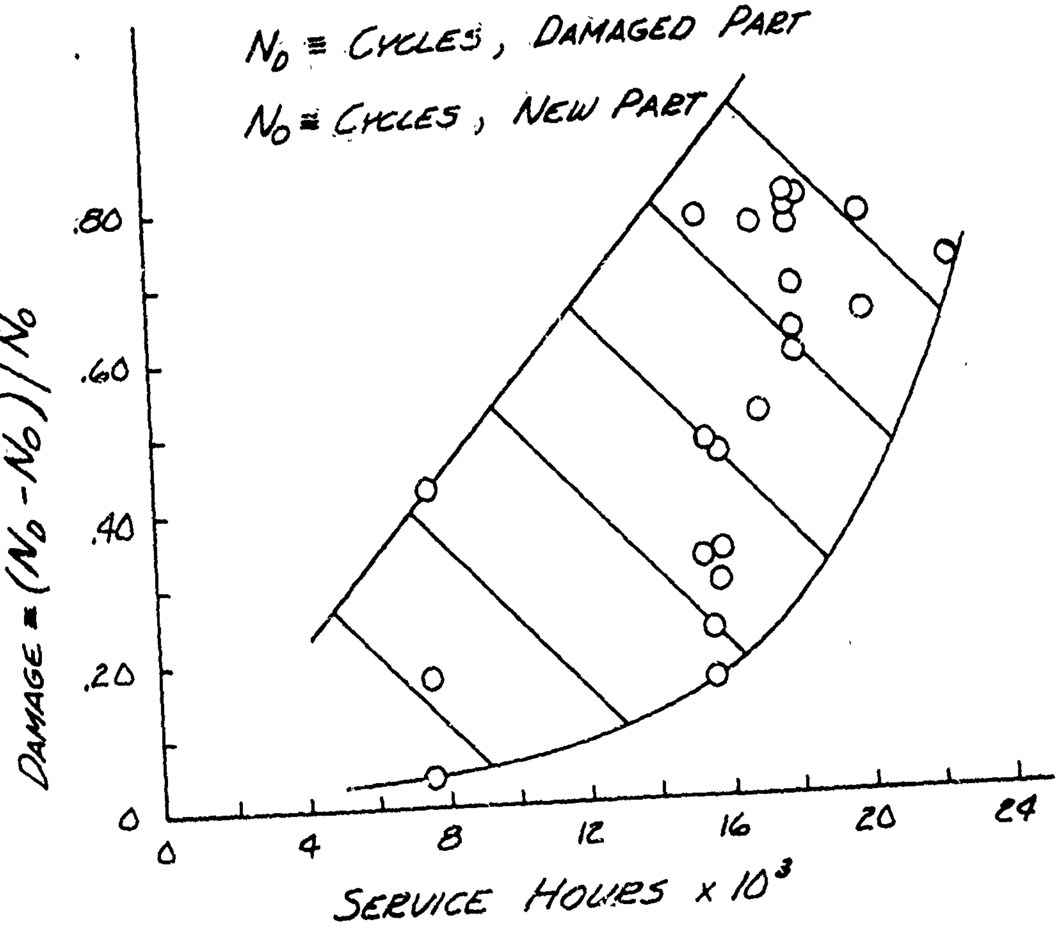

FIGURE 6. FATIGUE DAMAGE OF SERVICE-LOADED PART (REF. 26)

there appears to be a trend that, "for random load spectra", the summation-of-cycle ratios gives a value fairly close to unity.

Specimen Type	Total Number	($\Sigma n/N$) Mean
Unnotched	98	0.75 to 2.36
Riveted joint	90	0.86 to 1.31 (5.87)[a]

(a) High value, unexplained, from one group of nine specimens.

At this time, there seem to be two somewhat divergent opinions:

(1) After a great deal of study, it appears that for well-mixed loading the summation-of-cycle ratios fits observed data fairly well (say, within a factor of 2 for prediction of lifetimes). There may be other uncertainties (in load spectra, in stress-concentration factors, and in fatigue scatter) amounting to more than this (say, a factor of 10). Therefore, the later material damage theories, which are more complicated in application and usually require much additional experimental data, are scarcely justified in design.

(2) The more complex damage theories show trends which should be taken into account in design. In particular cases (especially some types of loading), these trends may have significance and, in some instances, may help avert unconservative design.

As suggested in this brief review, quantitative information to judge between these opinions is not only lacking but very difficult to obtain.

A somewhat different approach deserves consideration. The more recent theories, despite their complexities, appear to fit data on small coupons better than the summation-of-cycle ratios. If the small coupon behavior were better understood in respect to behavior of complex structures, these improved damage theories could sometimes be used to consider whether a particular load spectrum (for example, with a very few high loads early in service life) would be such that the cycle-ratio sum would be exception-

ally far from unity. The important missing item here is certainty of knowledge about the behavior of local stresses in a complex and possibly redundant structure. As an example, Smith (Ref 26) quotes values of lifetime for a riveted joint which, at the same amplitude of gross stress, varied about as follows:

No prestress	40,000 cycles
18,000-psi prestress	30,000 "
42,000-psi prestress	400,000 "

Note that this implies a slight weakening by one cycle of a moderate prestress, but very significant strengthening by one cycle of a high prestress. There have been, in numerous specific instances, observations of strengthening a part by preloading. The possibility that a few high cycles may strengthen a structure, in contrast to high-stress-damage effects in material coupons, raises questions as to the applicability of some of the recent theories of material damage behavior in situations where load spectra are known but detailed stresses are not well known. Until such strengthening and damaging factors are isolated, the often smaller effects resulting from more refined theories of material damage may be insignificant in design application to complex structures.

CONCLUDING REMARKS

The problem of cumulative fatigue damage was recognized many years ago. The use of the summation-of-cycle ratios then suggested still provides, in many instances, predictions of lifetime within scatter in fatigue data and other uncertainties in structural design problems.

Within the past few years, however, other theories have been advanced. These usually alow for higher damage, for a specified cycle ratio, at higher stress amplitudes. With some adjustment of empirical parameters, these theories provide somewhat better agreement with data on small coupons tested in the laboratory. They are also more compatible with accumulating knowledge of the formation and propagation of fatigue cracks.

At present, the more refined theories are difficult to use in engineering design. More important, there is evidence of factors in structural components which can contribute load-level effects in damage greater than the stress-level effects evident in material coupons. Enumeration and analysis of these factors seems necessary for a major advance in a cumulative damage theory for airframe structural evaluation.

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FATIGUE SCATTER AND A STATISTICAL APPROACH TO FATIGUE LIFE PREDICTION

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Without question the fatigue performance of materials and these same materials in structure have some confounding aspects. Neither testing nor analysis demonstrates the consistency in results found in the usual ultimate strength determination problem. However, it is believed that there are reasonable engineering solutions to fatigue questions. The primary hurdle is the fact that fatigue performance is far more complex than the static strength problem. While maximum loading, its direction and structural member configuration are sufficient for static strength estimates, fatigue performance not only includes these effects but also reflects the influence of the multitude of far lesser loads of ordinary usage. Local change in load magnitude rather than average or maximum magnitude itself is more important to fatigue response. Furthermore, sequence and frequency of all the loads may have a significant influence on the particular structure.

Figure 1 outlines the main areas contributing to the complexity of the fatigue problem. Physical environment, such as corrosion, temperature, etc. is an added complication to determining fatigue performance. Even detail design can mask intended improvements or controls of some of the variables. Material variables, such as their type, chemistry or manufacture and their thermal or mechanical treatment during fabrication, and other details are significant factors in the fatigue performance of structure.

Fatigue performance or fatigue-worthiness is not an everpresent quality like structural strength but is something that is expendable. On the other hand fatigue damage initial appearance in structure is not necessarily catastrophic. In the usual aircraft composite, semi-monocoque structures, the initial fatigue damage is local in nature. Repair and rework procedures can extend the service potential of structure (Reference 1).

Present day design solutions for fatigue critical structures are advancing along two possible paths of solution. These provide either a basic safe fatigue performance period or a structural capability to contain safely the growth of fatigue damage under normal loads until detected and repairs are made. Now, both crack propagation and the strength of materials in the presence of a crack in addition to fatigue crack initiation receive attention from the fatigue conscious structures engineer. Whether fatigue performance or this fail-safe philosophy are really independent solutions is not obvious. Maintenance economies encourage the judicious combination of these philosophies. However, the degree and manner of exploitation is influenced by the particular design objectives of the aircraft. With the increasing emphasis on fatigue problems, it is possible to overlook that the successful fatigue performance of many of today's aircraft is developed capability. Continued usage under reasonable structural surveillance procedures locates

the initiating fatigue damage while it still is in an incident rather than an accident stage. With this positive knowledge, improvements in the fatigue performance of the structure may be made. With given material controls, causes of fatigue damage may be divided into about three main areas: (a) the environment is different than that anticipated; (b) the discovered critical local stress concentrations were neither obvious or susceptible to analysis or (c) the fatigue performance of actual detail is different than that estimated. Although solution of the first two problem areas is aided by experience, proving tests minimize the extent of unknown failures. The last area is resolved by either reference or the performance of laboratory controlled proving tests. However, consistent fatigue performance under either laboratory or actual service conditions is not always found. Hence, the following remarks are directed towards the resolution of the variability found in the actual fatigue performance of materials and their structural applications.

FATIGUE DESIGN DATA

The structural designer, if not already aware of the problem, becomes acquainted with the problem of scatter in his reference to his "bible", the ANC-5 (Reference 2). Here the ultimate strength of materials are given in terms of either guaranteed minimum properties or a probable performance level. The latter properties are equalled or exceeded in 90 percent of the time. Furthermore, there is generally a relatively small difference in performance between the minimum or guaranteed and these probable mechanical properties. Only a few percent difference is found. On the other hand, fatigue performance shows considerable more uncertainty. First, there is no guaranteed fatigue performance of the basic material and secondly, there is a wide variation in the available data. Both notched and unnotched specimens show this large variation which may vary from one to several orders of 10 between maximum and minimum fatigue performance.

In figure 2, there is transcribed some ANC-5 data for 2024-T4, 7075-T6 and 2014-T6 notched specimen materials. In the endurance or long life range it is difficult to even assess the variation. The relative life of the high performance material is no longer numerically comparable with the low performance material in terms of life ratios. Relative stress levels have more significance. Another distressing fact is that there is apparently no really obvious difference in fatigue performance in these three alloys. This is somewhat confusing since inspection of some specific laboratory test data indicates that there is an apparent difference in their fatigue performance in terms of crack initiation, crack propagation and residual strength or tear resistance.

RANGE OF MAXIMUM TO MINIMUM LIFE

To explore this variation in fatigue performance, data on the scatter found between maximum and minimum life of laboratory test specimens varying in alloys and design is studied. In Figure 3, 357 groups of data tested at constant repeated load conditions have been summarized. These groups of data represent 2 to 5 specimens in each group and reflect both joint and basic material notched and unnotched specimens. Both aluminum and steel alloys are included in this group. While there is considerable scatter shown, it is evident that there is a centralizing effect. The median or 50 percent of the

data has a ratio of 1.6 while 95 percent of the data has a scatter factor of 4.2 or less.

Extending this same sort of comparison to 60 test groups containing 6 to 15 specimens, similar evidence is found in Figure 4 to show a median ratio of 1.9 while 95 percent of this data has a ratio of 5.0 or less.

Likewise, groups of data varying from 16 or more test specimens were examined. Figure 5, based on 43 groups of data, shows a median ratio less than 2.6 and 95 percent of the data has a ratio of maximum to minimum test life of 9.5 or less. These latter tests reflect mostly base material tests rather than joints. Gathering all of this multiple test specimen data into one collection of 460 test groups the centralizing tendency of this maximum to minimum life ratio is again evident. From Figure 6, the median scatter factor is 1.7 and 95 percent of the data has less than a 4.5 to one scatter.

From this inspection of diverse types of test specimens two principle observations are made. First, maximum scatter in materials can be great with one group showing over a 270:1 variation between maximum and minimum. Secondly, in spite of this wide variation there is a tendency towards an average or 50 percent probable value of scatter ratio. A third observation is that the greater the test group the greater the range between maximum and minimum test becomes. Hence, it appears that there is a reasonable expectation of assigning some degree of reliability in fatigue performance even though absolute extremes of performance levels may not be readily resolved. Nevertheless, this collected data has some usefulness as a means of evaluating the fatigue performance likely to be obtained from single specimen testing.

RANGE OF AVERAGE TO MINIMUM LIFE

With multiple specimen test data, another natural measure of fatigue performance is the average life of the test lot. For purposes of this comparison, the logarithmic average cycle life (i.e., the cycle life for the average for the logarithms of the test lives) has been selected. In a way, this reflects the usual semi-log graphical presentation of S-N data. Figure 7 shows a comparison of fatigue scatter in terms of the ratio of the logarithmic average test life to the minimum test life for test groups of 2 to 5 specimens. The median or 50 percent performance point shows only a ratio of 1.25 while 95 percent of the data has a ratio of 2.1.

A somewhat similar magnitude of scatter is shown by the test data with 6 to 15 specimens per test group as shown in Figure 8. The median point has a 1.4 scatter ratio while 95 percent of the data is calculated to have a variation of mean to minimum of 2.4 to 1 or less. Figure 9 shows that the 43 groups of 16 or more test specimens have median and 95 percent equal-or-less ratios of 1.7 and 2.7 respectively. Furthermore, with collecting of all 460 groups of the data together, the ratio of test logarithmic average life to minimum life has a median value of 1.3 and a 95 percent equal or less value of 2.2, as shown in Figure 10. A summary of all these scatter ratios is shown in Table 1. Here it can be noted again that there are obvious central tendencies for measured fatigue performance. Furthermore it is likely that there is some probability of obtaining a certain level of fatigue

performance. However, the specifiying of an absolute or guaranteed minimum performance is not a clear-cut possibility with the inspection of this particular group of tests.

ESTIMAT^ N OF POTENTIAL FATIGUE PERFORMANCE

In order to use fatigue test data to analyze potential fatigue performance of a particular structural detail, some rational method of analysis is obviously necessary. The method must be able to compare the severity of the applied variable load history with the allowable or potential fatigue performance of the structural detail. Considerable research effort has been pointed towards developing such a method of analysis, but with uncertain success. Of all the methods of fatigue analysis developed (for example, see Reference 8), the linear cumulative damage or cycle ratio method is believed to have some advantages as an engineering tool. Figure 11 illustrates the general principles of this theory. It can account for the variable magnitude of the mean and the alternating stress experience of aircraft structures. Furthermore, the method is simple in nature. However, it doesn't account for loading sequence and its accuracy is subject to question. Considerable laboratory testing has been done to check its accuracy with negative results in many cases. But in spite of this apparently discouraging evidence, it does merit some consideration. Figure 12 summarizes 222 axial load ordered spectrum or programmed loadings. Two or more stress levels have been used in each of these tests. It is found that the median measure of performance has a cycle-ratio summation of 1.10 while the arithmetic average or mean life-ratio summation is 1.28. Furthermore, 97 percent of these data fall above a level made safe by factor of safety of 2.

Supplementing these data are the results of testing described in Reference 3 and shown in Figure 13. These tests include 26 groups of programmed loads tests of 2024 and 7075 riveted lap joints. Each group represents seven test specimens tested to the same spectrum. It is observed that the median life-ratio summation is about 1.1 and the average is over 2. There is also an indication that the method has some degree of reliability for life estimation even with the variety of loading spectrums in this research program.

Another set of programmed-load fatigue tests, not included in Figure 12, are shown in Figure 14. While these 380 rotating-beam tests do not indicate a consistent degree of accuracy for each specific load program, collectively the tests do show some central tendencies. Hence it appears likely that the method has some potential to predict fatigue performance with a degree of reliability.

The need for some assurance of a given fatigue performance has been reflected in the development of extensive detail civil airworthiness requirements. Reference 5 outlines the British-Civil-Air-Regulations factors of safety proposed for structure with a minimum estimated safe-life performance. Here the solution has been to calculate average fatigue life by the linear cumulative damage theory and apply a factor of safety to obtain the safe-life. These scatter factors were evaluated upon the basis of inspection of actual fatigue test data. These same scatter factors have been considered for the

U.S. Civil Air Regulations but so far no specific factors have been included in CAR 4b.

Reference 6 presents some statistically derived factors for wing structure and were presented at a 1956 International Civil Aviation Organization meeting. A probability of failure of 0.001 and a standard deviation based on available experimental data were used. Table 2 summarizes these factors of safety; i.e., the scatter factors by which calculated average life must be reduced to obtain a safe life.

By the above collective inspection of actual fatigue test data, it has been shown that there are overall central tendencies in the actual and estimated fatigue performance of both single and multiple load level tests. The simple linear cumulative damage theory appears to have a general capability of prediction. Hence, an empirical resolution of the fatigue scatter problem and determination of estimated performance with a certain degree of reliability is practical with presently available engineering tools.

DEFINITION OF THE S-N CURVE

The first step towards resolution of this problem is the definiton of the allowable fatigue performance in the form of the S-N curve. Figure 15, based on data from Reference 7 reflects the scope of this task. Of course, an engineering approach requires a modicum of data, but scatter can provide hazards. Three characteristics of the measure of fatigue performance must be determined. These characteristics are the shape, life orientation and stress orientation. The analytical results of fatigue performance with the linear cumulative damage rule are quite dependent upon the integrated effect of these three factors.

Figure 16 is a plot of the same data and shows the logarithmic average of the test groups at each stress level. The range of the test data is also plotted. The parenthetical numbers indicate the total number of specimens at each test stress. It is pointed out that this data reflects basic material performance tested at a constant ratio of minimum to maximum stress ($R = 0$) rather than at a constant mean stress which is thought amenable for the usual wing fatigue calculations. It is obvious that the use of multiple test specimens at each stress level does provide a step in the definition of the S-N curve. In Figure 17, this test data is reflected in statistical form. A definite central tendency is observed in the data. While the assessment of fatigue scatter may seem a formidable task in the light of this typical scatter data, an examination of the static strength development of structures should be considered. Current practise for the development of static strength in military aircraft has utilized the full-scale static test to verify the analytical evaluation. This testing has led to the assurance of static strength equivalent to the design requirements. However, the developmental work associated with this goal is not always appreciated. For instance, Figure 18 illustrates the occurrence of first indications of failure in terms of the design loads. In one group of static tests it is observed that an initial failure occurred as low as 40 percent of the design load before the finally developed static strength goals were met. Hence it is suggested that there is some similarity in the static and fatigue strength evaluation; that is, both in the unproven state do have some variability.

ANALYSIS OF FATIGUE SCATTER

Obviously, the extent or magnitude of fatigue scatter is not resolved by a single test or two. However, since the state of knowledge on the nature of fatigue performance does not allow a fundamental approach to the problem, an empirical approach is the alternate solution. Several investigators have performed tests on a multiplicity of test specimens that permit statistical measures to be applied to add meaning to the available data. However, it must be remembered that statistical interpretation of such test data offers no miraculous insight to the nature of fatigue performance, but instead only reflects the actual input test data. Various mathematical models have been proposed for this purpose but such developments naturally add to the complication of analysis over the simple methods. Figure 19, based on the test results of 102 specimens, illustrates the fit of the simple log-normal or log-Gaussian frequency distribution curve fitted to a histogram analysis of the data. While not a precise fit, there is sufficient apparent correlation to use this method as a tool for analysis. The use of a distribution curve is reflected in Figure 20. With an analytical form of representing the distribution of fatigue failures it is possible to extrapolate meager frequency of occurrence data to reflect gains in knowledge without extensive testing and to establish likely lower limits of performance with some degree of reliability. For instance, with a normal distribution, a very few specimens will provide an estimate of the mean and the standard deviation. With this knowledge some estimate of the likely lower limit for a given level of confidence can be made.

SELECTION OF A DISTRIBUTION FUNCTION

As noted in the preceding remarks, resolution of the distribution of the frequency of fatigue damage initiation or fatigue failures has not yet been successfully accomplished. However, some relatively satisfactory approximations have been made. The use of the normal or Gaussian distribution is the easy solution. This distribution function has had considerable exploitation in various fields other than aircraft structures and has its characteristics well tabulated in the literature. Unfortunately it does have a behavior that is not aesthetically pleasing when applied to the fatigue problem. As noted in Figure 21(a), the minimum and maximum values are defined by "tails" that are asymptoptic to zero frequency of occurrence and hence can indicate potentially negative performance as a minimum at a high degree of reliability.

A cure for this feature, as illustrated in Figure 21(b), is to translate the variable into logarithms and obtain the log-normal distribution function. This transformation of the normal curve provides the normal curve symmetry with the logarithmic scale but a skewed distribution with a linear variable scale. In either form this relationship at least provides zero performance which is a bit more compatible with the observed nature of fatigue performance. However, it is still obviously conservative.

Similar performance as that offered by the log-normal distribution function can be gained by more complex formulation (Reference 9) indicated in Figure 21(c). While probably providing better theoretical fits to the data, two difficulties are associated with their usage. One difficulty lies in the extent of the data needed to define such a relationship. Another difficulty is the associated increased extent of calculations required.

A fourth type of distribution, as summarized in Figure 21(d), is that usually titled the extreme value distribution (Reference 10). This distribution is particularly attractive for use in the fatigue field because it does prove finite lower and upper limits which just naturally seem to be more applicable to the fatigue problem.

Another possible solution to the reliability problem in fatigue is that possible with the distribution-free or non-parametric forms of statistical analysis. This type of analysis eliminates the need for assumption of or demonstration of the fit of theoretical distribution function to the fatigue process. Unfortunately its accuracy in the case of fatigue is dependent upon the extent of the data. Under normal circumstances, it is doubtful that the necessary quantities can be feasibly obtained to arrive at high degrees of reliability. For the purposes of this discussion it is doubtful whether much real gains can be obtained over the simple log-normal distribution curve without considerably more developmental work.

APPLICATION OF THE LOG-NORMAL DISTRIBUTION CURVE

The log-normal frequency distribution curve is illustrated in Figure 22. Only knowledge of the mean and the standard deviation are necessary to define its properties. With a known true or population mean and a standard deviation, a specified degree of reliability can be estimated. In the case of fatigue, the interest is in the lower limit or the value which may be equalled or exceeded with a given level of reliability or confidence. For other practical applications of the normal distribution function, it has been observed that limits equal to about two or three standard deviations are generally used. However, it must be remembered that the accuracy of the log-normal curve in representing fatigue performance is conservative so it is likely that satisfactory results can be obtained with nominal degrees of confidence. In the case of fail-safe structures it is suggested that a lesser degree of confidence on the fatigue performance can be tolerated without sacrifice of realism in the analysis. Inspection controls can provide a safe-guard for any level of fatigue performance (Reference 11).

With limited test data, the normal (i.e., log-normal) distribution curve can be used to estimate probable lower limits of performance. Having no knowledge of the true or population mean and standard deviation it is possible to use data from small-size samples to compute a sample mean and a standard deviation. Then applying statistical theory which has been developed in the form of the well-known (Students') t-distribution, knowledge of the true or population mean is gained from limited sample sizes. As illustrated in Figure 23, a tolerance on the sample mean, computed by the t-distribution, defines a band or, in the case of this proposed fatigue application, a lower limit which the true or population mean will exceed with a given level of confidence. A ratio, $\overline{N}_S/\overline{N}_L$, provides a partial factor of safety to reduce the average or median fatigue performance of the sample to an average that will be equalled or exceeded with a given level of confidence. Then locating the distribution of this lower limit, a further tolerance, based on the form of the distribution (i.e., log-normal) can be calculated to provide an additional part of the total factor of safety for an arbitrary level of confidence. Thus precise statistical theory is used to define the average or median performance with a desired reliability or level of confidence. Then advantage is taken of the assumed distribution shape to calculate a further factor of safety associated with the desired reliability.

Upon superficial inspection this procedure seems reasonable enough. However, inspection of small sample sizes of fatigue test data will show quite a range of variability in their scatter. In some cases it is practically nil while in other cases it may be appreciable. Hence, application of this method can provide inconsistencies since chance may supply a different parameter to define the factor of safety for similar joints experiencing the same phenomenon. This is not a satisfactory situation even for an approximation scheme. An obvious solution for this difficulty is then to use the averages or mean of the data and an assumed standard deviation. However, regardless of the workability of this solution, there are some statistical objections since this approach infers knowledge of the true or population standard deviation and casts some technical doubts on the propriety of using the t-distribution.

With a known standard deviation for a normal or Gaussian distributed population the procedure would be the same as described above except for the use of normal instead of t-distribution data. As a result, the indeterminate characteristics of the t-distribution for one specimen are avoided. Furthermore, use of the normal distribution characteristics tends to show a little more effectiveness in the definition of the range or tolerance relative to the mean value. Figure 24 illustrates the application of a known standard deviation. In establishing the upper and lower limits on the sample mean k is the factor from the normal distribution curve tabulated properties for the relative number of standard deviations associated with a particular level of confidence (i.e., k = 1.28 when 90 percent of the data will equal or exceed a given level of confidence as shown by Figure 22).

Table 3 tabulates the median and several other levels of confidence for which the minimum life has been estimated by various procedures. Because this particular lot of data is rather well-behaved as far as scatter is concerned, there is no real difference in actual lives for any of the assumed distribution function procedures.

SCATTER OR VARIABILITY FACTORS FOR LOG NORMAL FREQUENCY DISTRIBUTION CURVE

Using the assumption of log-normality for the character of the fatigue damage initiation frequency distribution curve with a known mean life, the magnitude of the variability or scatter factor can be computed for any degree of reliability. In Figure 25, this factor is computed for the log-normal distribution over a range of reliability or probability from the median or 50 percent point to one part in 100,000 parts for three values of the standard deviation. With a fatigue performance that will be equalled or exceeded with a probability of .95, a fatigue variability or scatter factor of about 2.1 is found when the true or population standard deviation is 0.20. If 99.99 percent of the parts are desired to equal or exceed a given performance, it is seen that an 0.02 standard deviation requires a life reduction by a factor of 5.5 for the life calculated by the use of the average and median fatigue performance. However, it should be remembered that this scatter or safety factor is merely a calculated one and does not have supporting test evidence. Obviously in full scale structures or even simple specimens, the proof of extremely high degrees of reliability by test evidence is a formidable task. With an estimated mean life and assumptions of a population and a sample standard deviation, the magnitude of the mean life variability factor is

illustrated in Figure 26. The effectiveness of a statistical analysis based on log-normality and a known population standard deviation is shown by comparing the solid line curves of Figure 26 with the curved dash lines. The dashed curves reflect the use of the t-distribution. It is observed that the two groups of curves diverge rapidly as reliabilities over 95 percent are reached. Comparing Figure 25 with Figure 26 it is observed that the mean life variability factor is significantly less than that for the "distribution" effects even for 3 specimens. Only with one test specimen would the tolerance for the same degree of reliability be the same for the mean and for the distribution. Hence, from limited sample data it is supposed that the average life can be estimated with more efficiency than a minimum life.

Extending this statistical analysis to allow for a tolerance on the accuracy of the mean due to small sample sizes and also to account for population scatter, Figure 27 illustrates the two approaches to minimum life estimation with the assumption of log-normality and the use of a known or an estimated standard deviation. A reference point of the previously noted safe-life factor is drawn on the chart. It is apparent that the suggested simplified statistical systems do not match the safe-life factors based on engineering knowledge of observation and judgment. Hence some degree of conservatism is present in this proposed statistical analysis system for higher degrees of reliability. In Figure 27 it is further observed that knowledge of both the log-normal nature of the distribution and a population standard distribution controls more effectively the magnitude of the total variability factor.

In calculating the variability factors it is observed that the reliability of the mean is closely associated with the number of test specimens. In Figure 28 there is plotted the relationship between the number of test specimens and their effect on the magnitude of the scatter factor for a range of probabilities of equalling or exceeding a given minimum life. It is observed that for a given standard deviation the use of the t-distribution produces the expected greater variability. Furthermore, the t-distribution is not capable of recognizing a test sample size equal to one specimen. This of course suggests that a multiplicity of specimens be always tested; however, the usual situation finds one in the position of testing one specimen or possibly none -- a difficult choice at best!

THE STANDARD DEVIATION

In the preceding remarks, it is evident that the standard deviation is a significant feature of this proposed statistical analysis. A review of the calculated sample standard deviations of the 460 groups of multiple test specimens has been made. A histogram of this data is shown in Figure 29. Central tendencies for this parameter are also observed. A median or 50 percent probability value of the standard deviation is found to be only 0.10 (base 10 logarithms). The average deviation is 0.12 while over 95 percent of the deviations were calculated to be less than 0.25 to 0.30 interval.

SUMMARY OF SCATTER FACTORS AND THEIR APPLICATION

As a summary of the previous development of the scatter or variability of fatigue performance, Table 4 has been prepared. Within this table

has been collected the factors which are applied to reduce the mean life of a structural element to a reliable level of safety. Factors from Reference 5 as well as the two conditions of statistical analysis developed in this paper are included. The two statistical measures of reliability have been shown for a 95 percent reliability and three values of the standard deviation. Again it is obvious that knowledge of the distribution function only is not sufficient to provide a good feel for the degree of reliability in the one or two specimen test sample size. Conversely, as the number of samples increases beyond three, the gains in knowledge of the reliability are not really great as compared to the added test effort.

The development and extension of this approach to the estimation of the fatigue performance of aircraft is graphically illustrated in Figure 30. The first step is the determination of the mean fatigue performance and the lower limit of the true mean with a given level of reliability or confidence. Using the lower limit to orient a frequency of fatigue damage initiation or occurrence, an integration of this curve is possible to obtain a cumulative probability distribution curve. This latter curve illustrates the probability of fatigue damage initiation with time or cycles for an individual detail. With a multiplicity of details subject to the same environment, the probability of one or more cases of fatigue damage existing at the same time should be considered. With P(t) taken as the number of failures equal to or less than the maximum failure goal, a probability relationship for "N" details is also shown in Figure 30. Further remarks on this subject will be found in Reference 11. The necessary point here is only to recognize that with a given "time-probability" fatigue damage performance level, the likelihood of existence of one or more cases of fatigue damage is greater than that set for the individual performance. Of course the nature of the individual part frequency distribution curve is important to the conclusions to be derived from this analysis.

In conclusion, Table 5 outlines the suggested approach of this paper for the establishment of the useful fatigue life with a given degree of reliability. Of the points enumerated only the variability due to basic fatigue scatter has been examined in complete detail in this paper. The reader will recognize the need for careful consideration of the other facets of the problem.

In closing these remarks, it is desired to express an appreciation to the various members of the Fatigue Studies Group who have performed the detail computations and collected the extensive data necessary to compile this paper.

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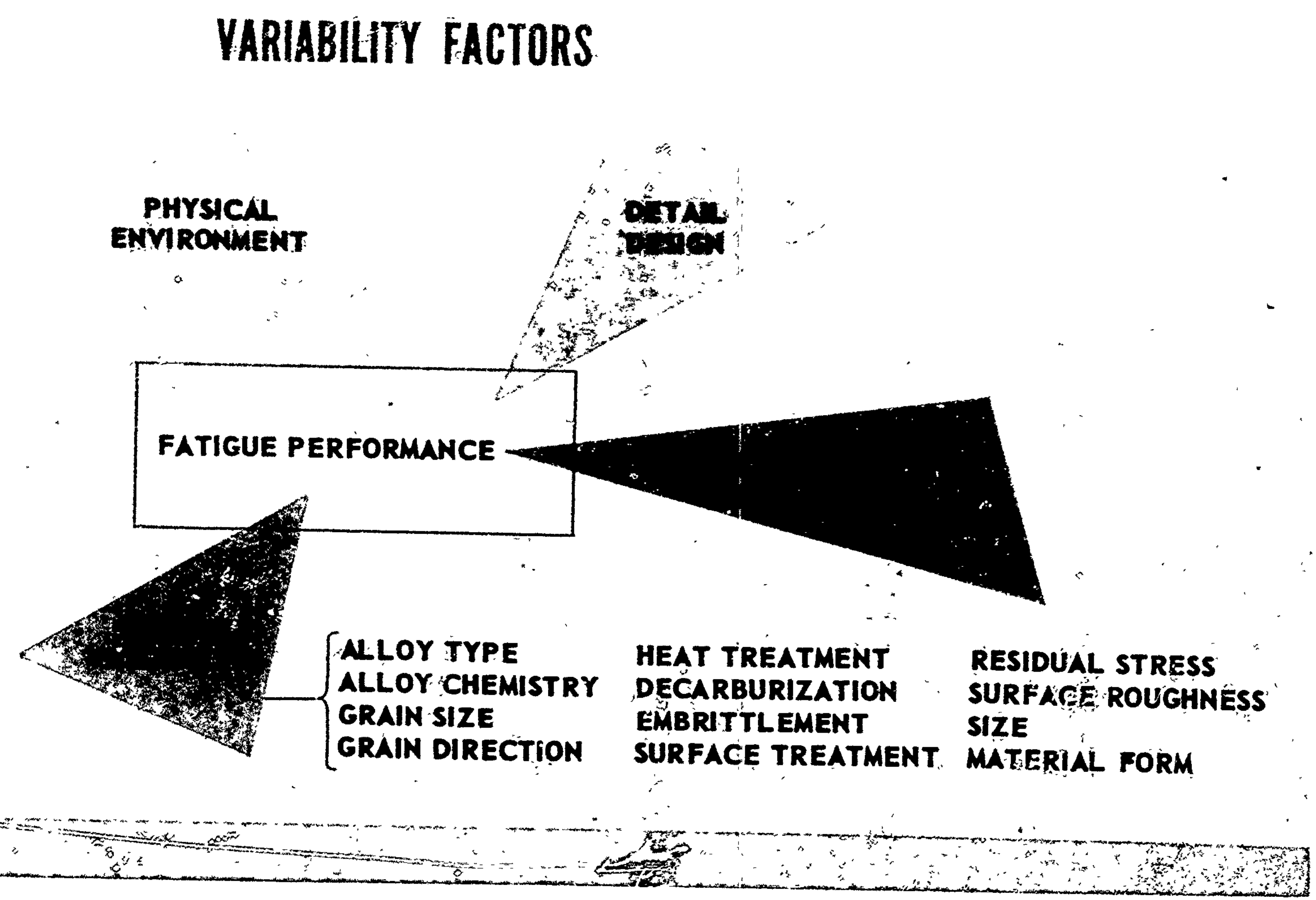


Figure 1 - Variability Factors In Fatigue Performance

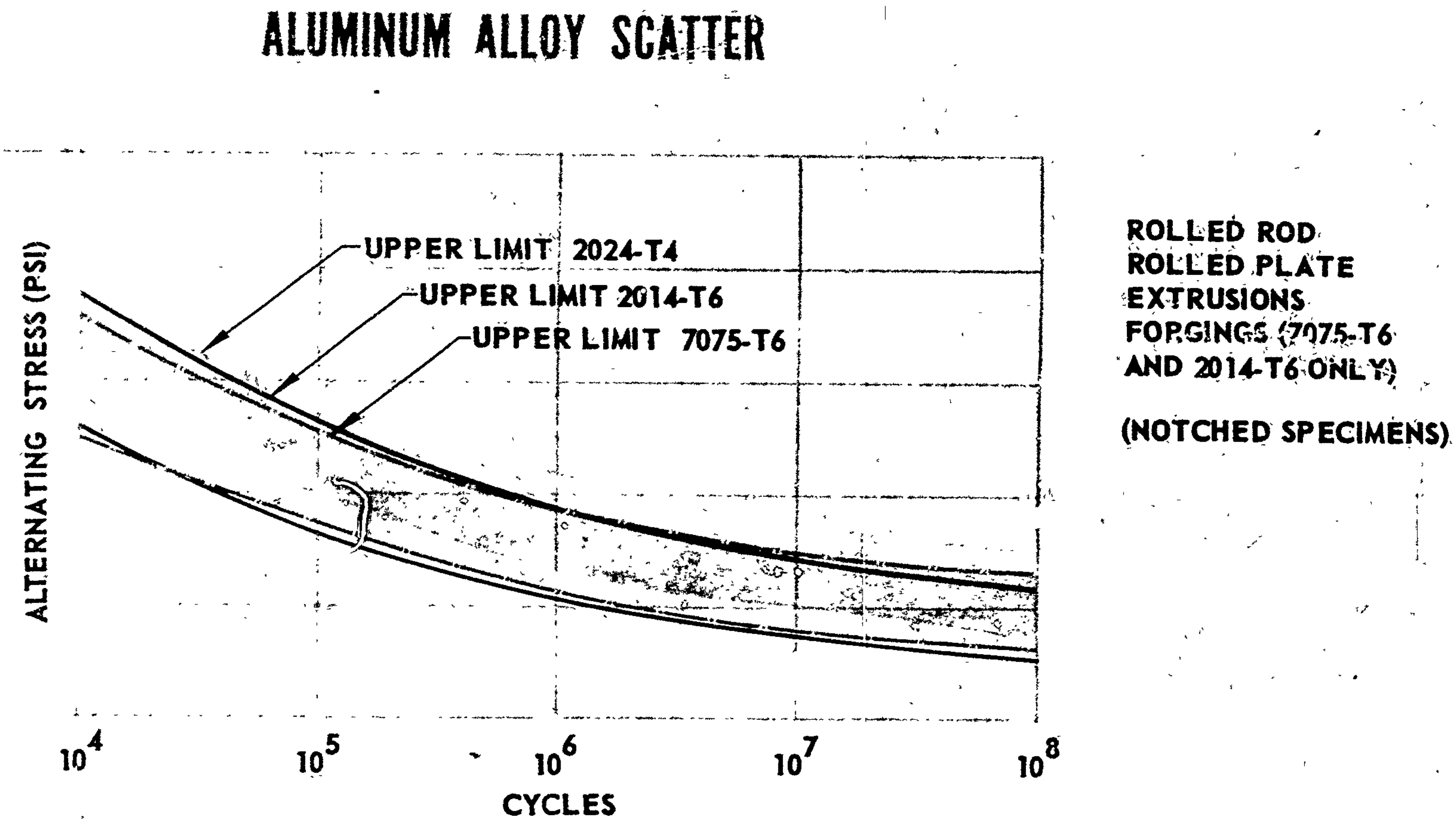


Figure 2 – Fatigue Scatter Between Aluminum Alloys (Taken From ANC-5)

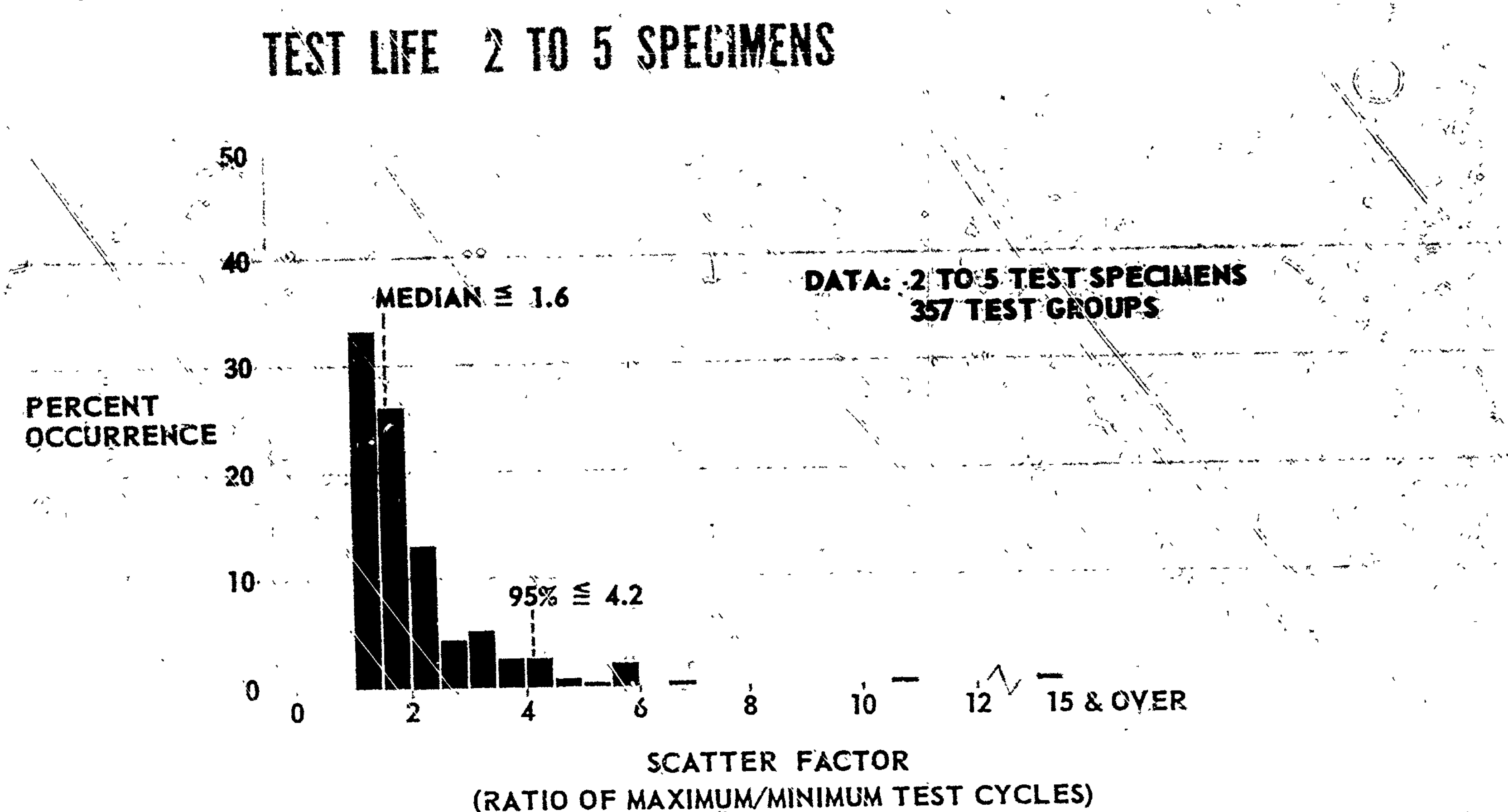


Figure 2 - Ratio Of Maximum To Minimum Test Life For 2 To 5 Test Specimens

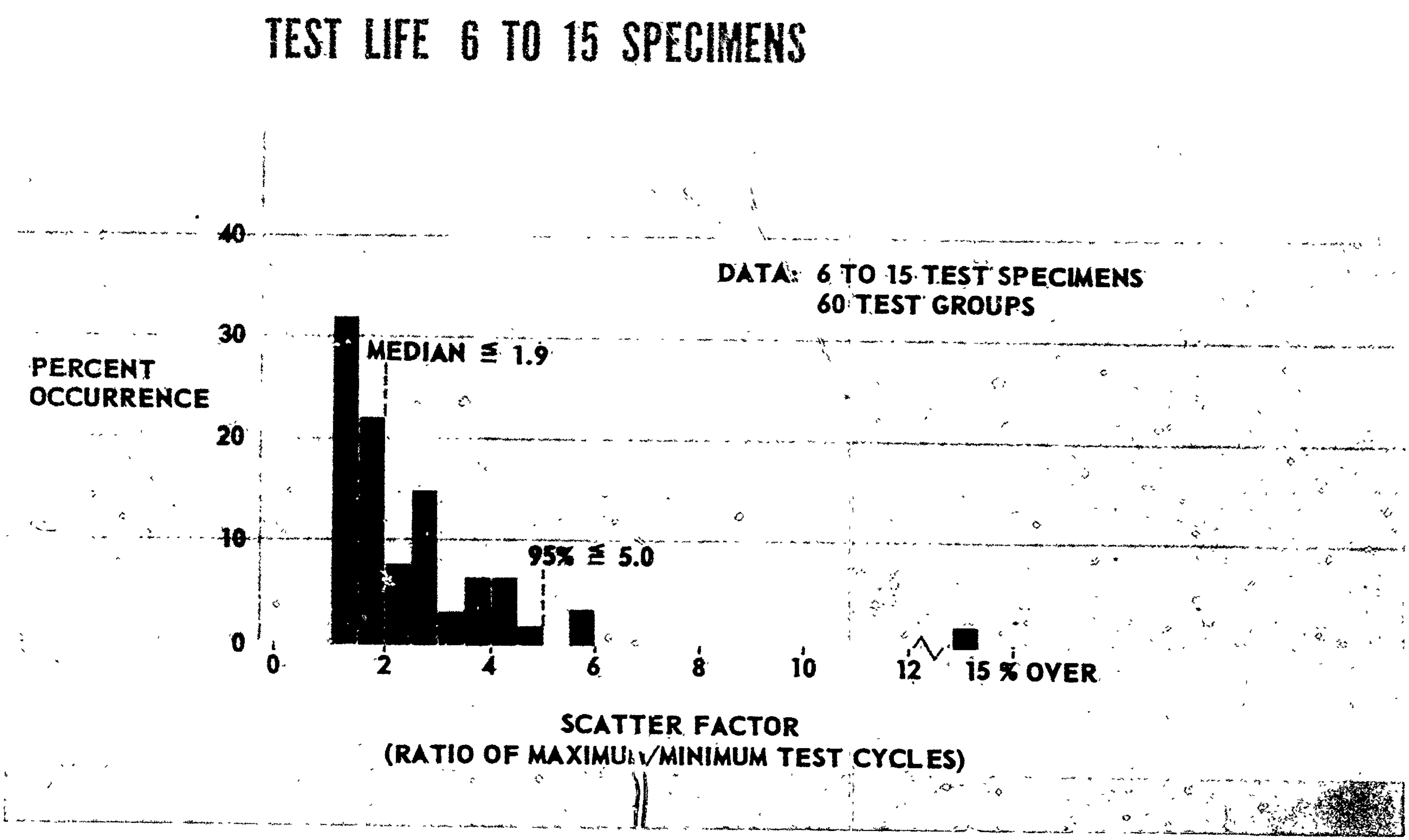


Figure 4 - Ratio Of Maximum To Minimum Test Life For 6 To 15 Test Specimens

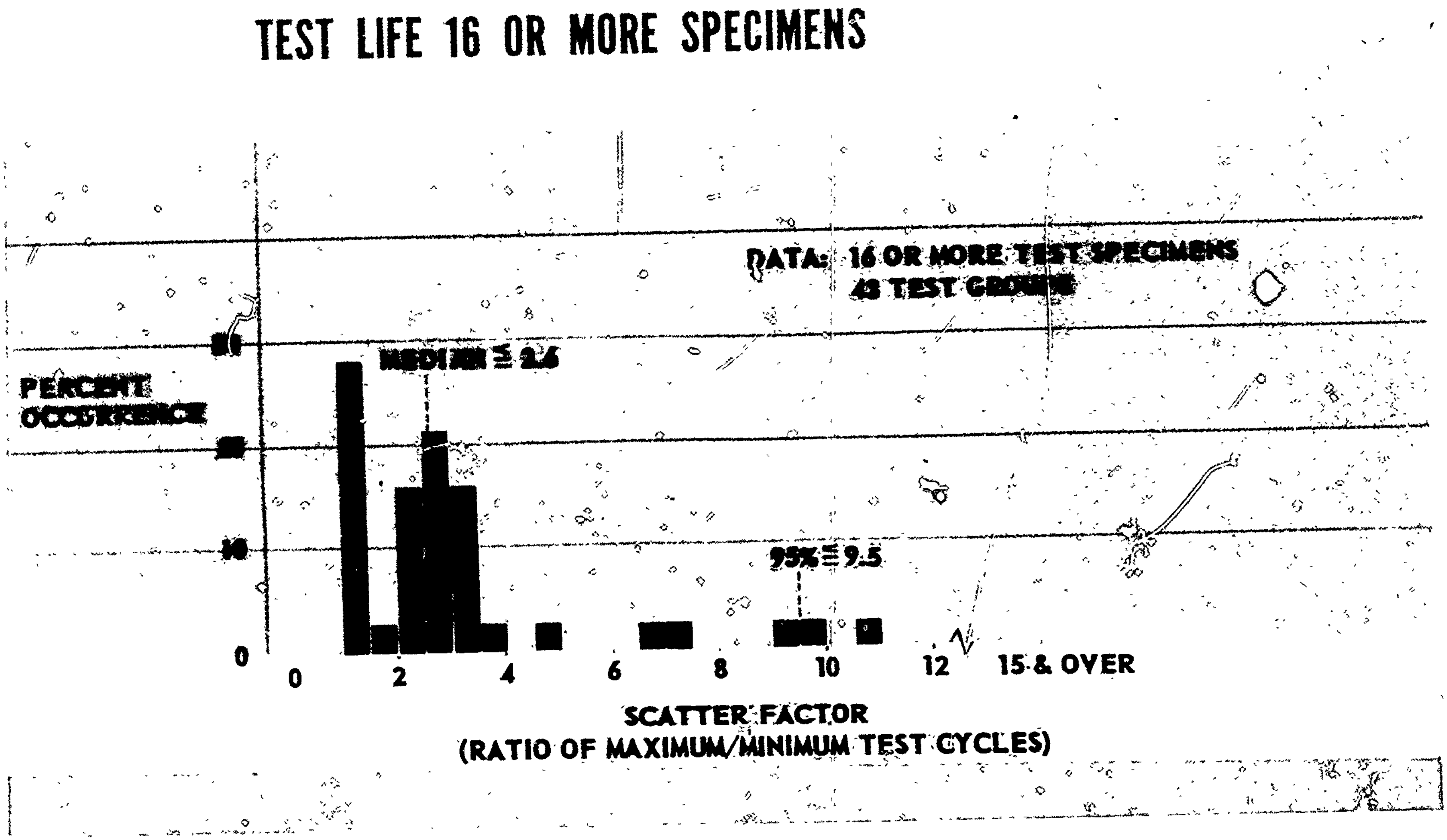


Figure 5 - Ratio Of Maximum To Minimum Test Life For 16 Or More Test Specimens

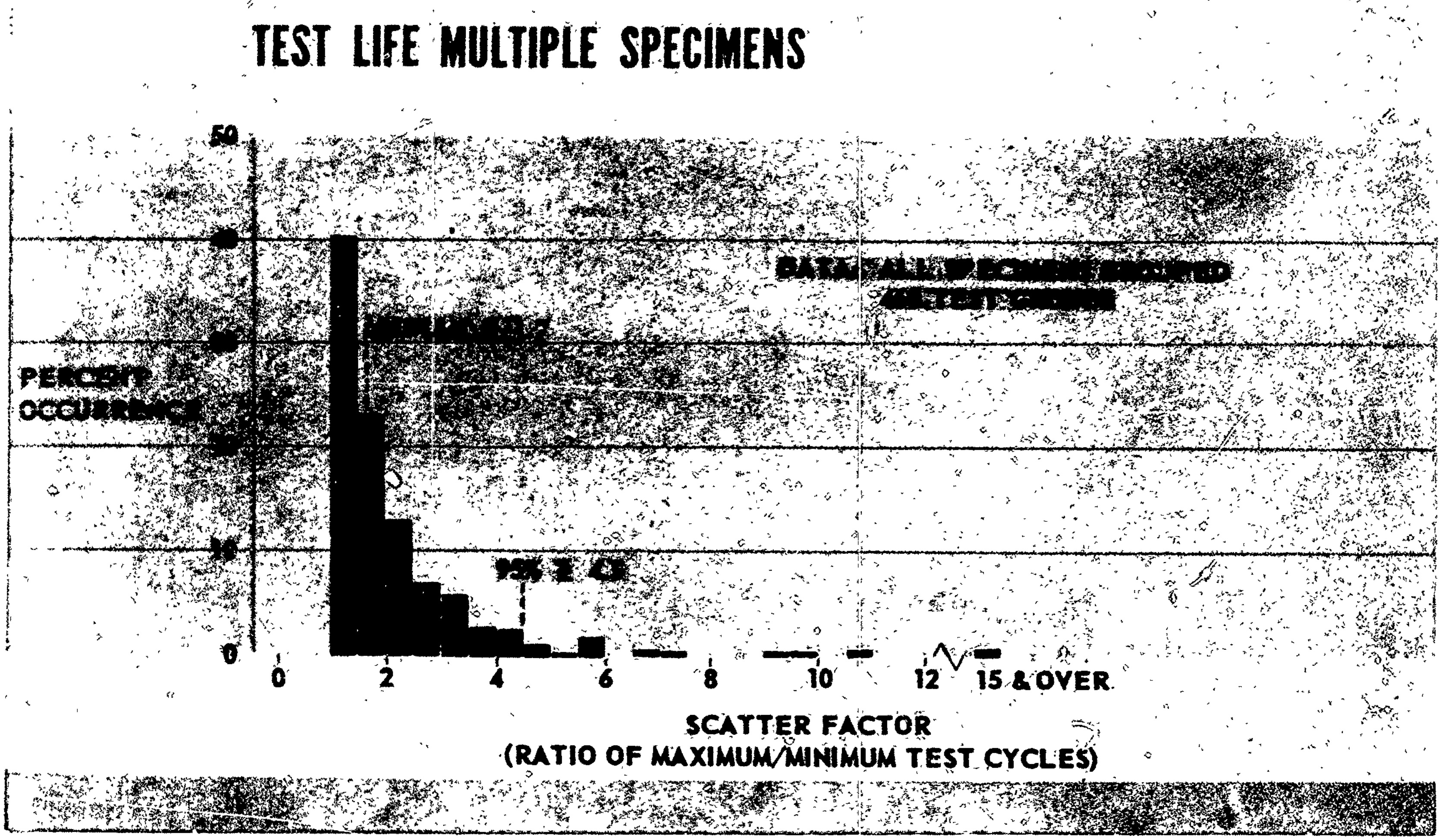


Figure 6 – Ratio Of Maximum To Minimum Test Life For Multiple Test Specimens

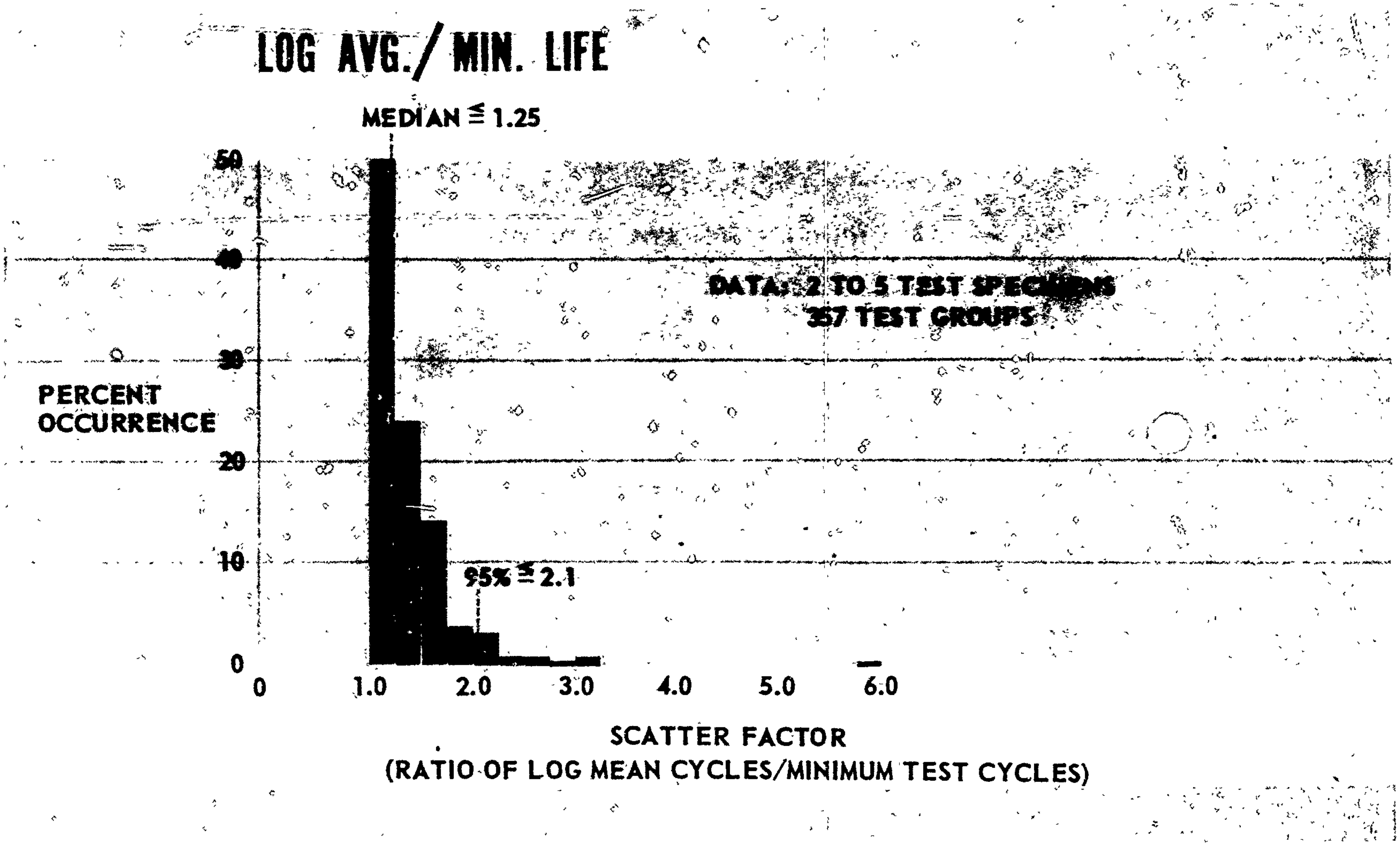


Figure 7 - Ratio Of Logarithmic Average Life To Minimum Life For 2 To 5 Test Specimens

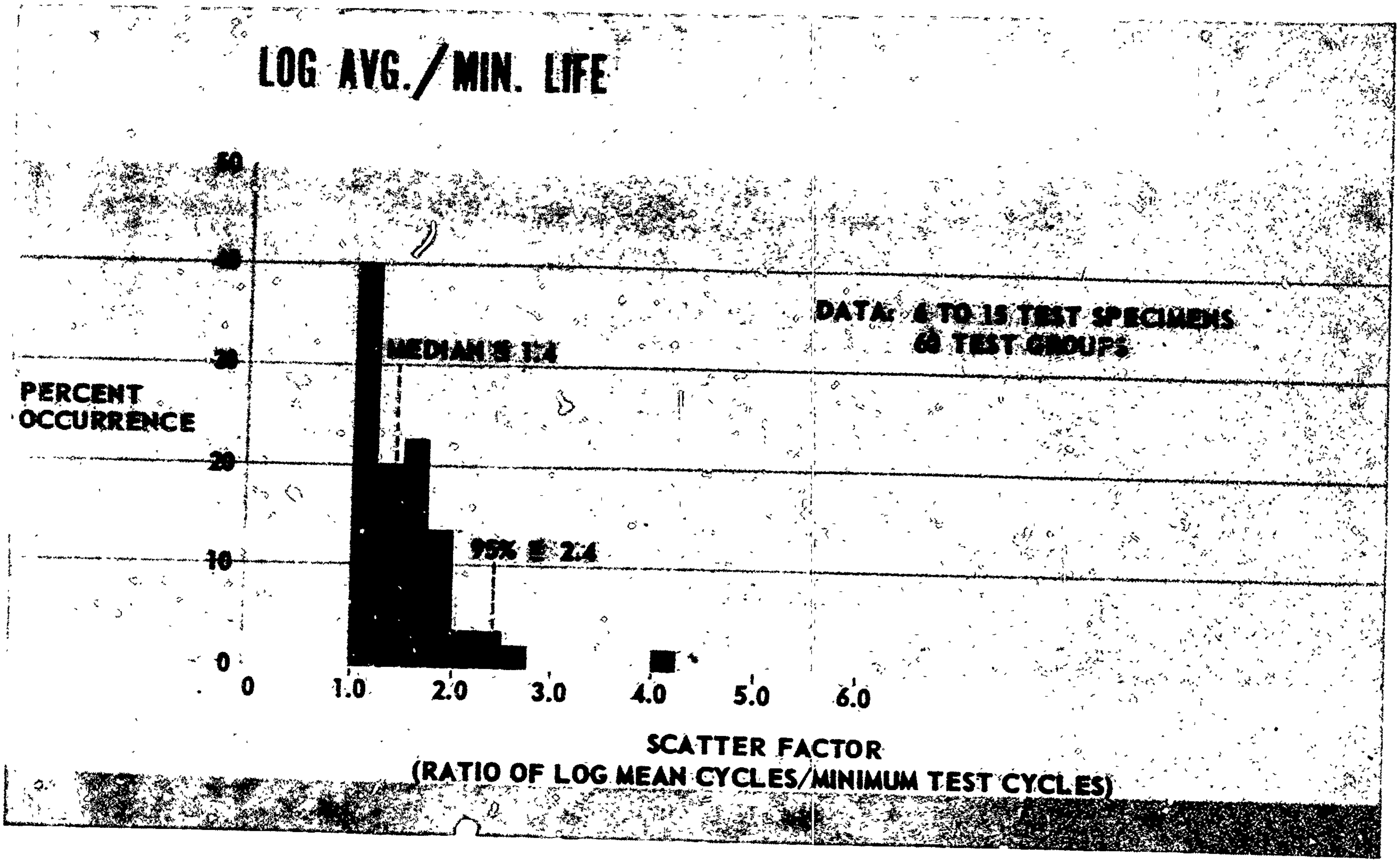


Figure 8 - Ratio Of Logarithmic Average Life To Minimum Life For 6 To 15 Test Specimens

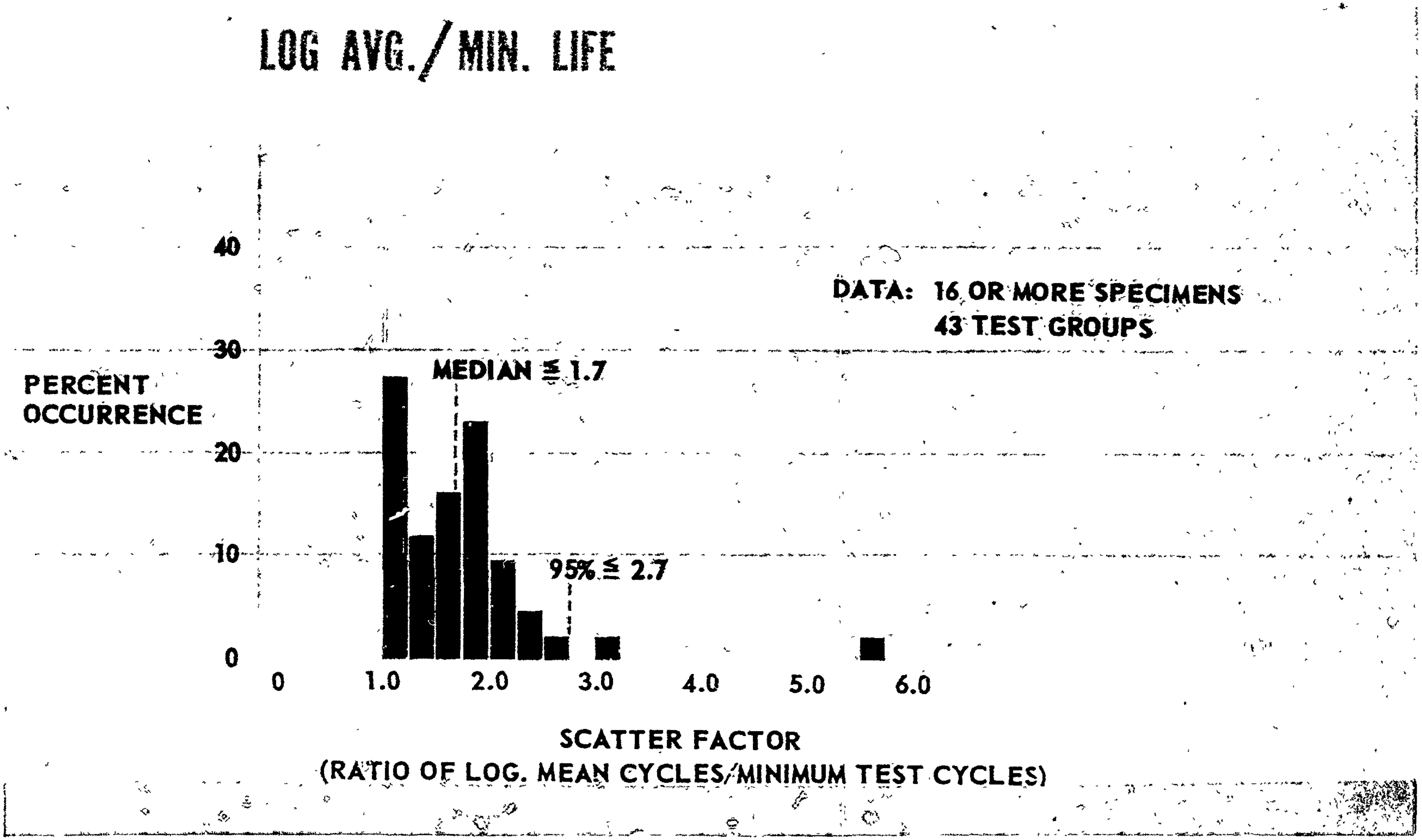


Figure 9 - Ratio Of Logarithmic Average Life To Minimum Life For 16 Or More Test Specimens

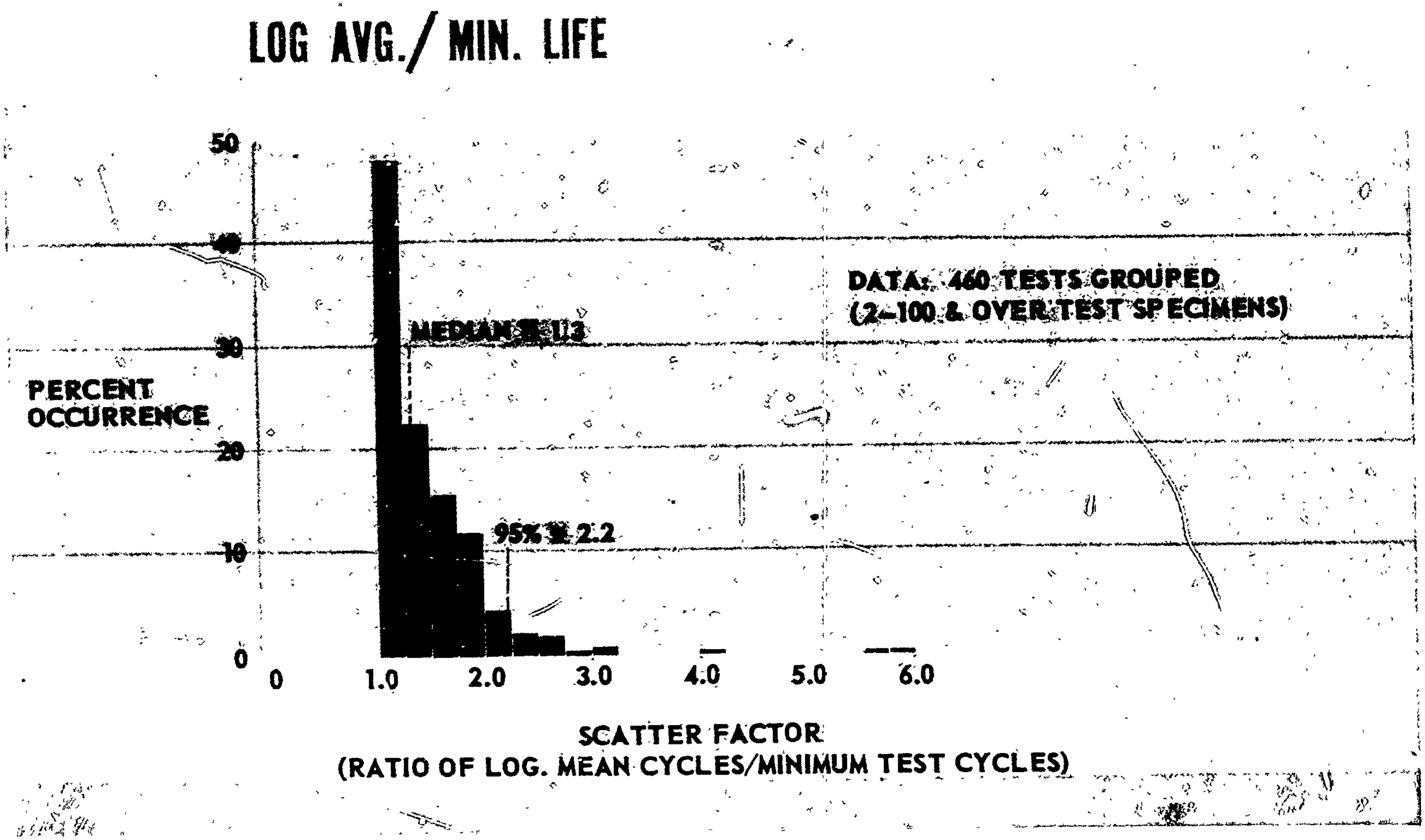


Figure 10 - Ratio Of Logarithmic Average Life To Minimum Life For Multiple Test Specimens

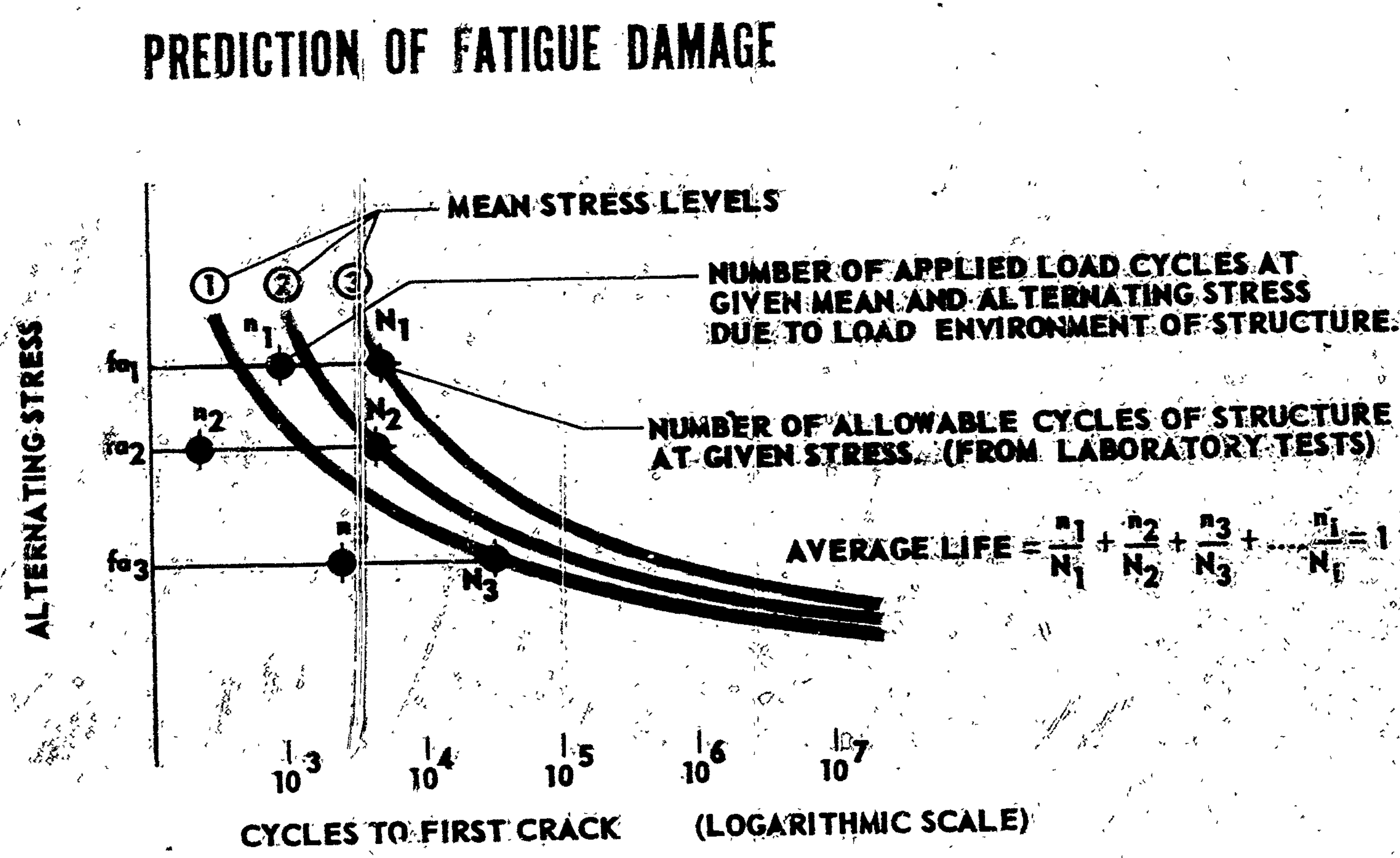


Figure 11 - Prediction Of Fatigue Damage

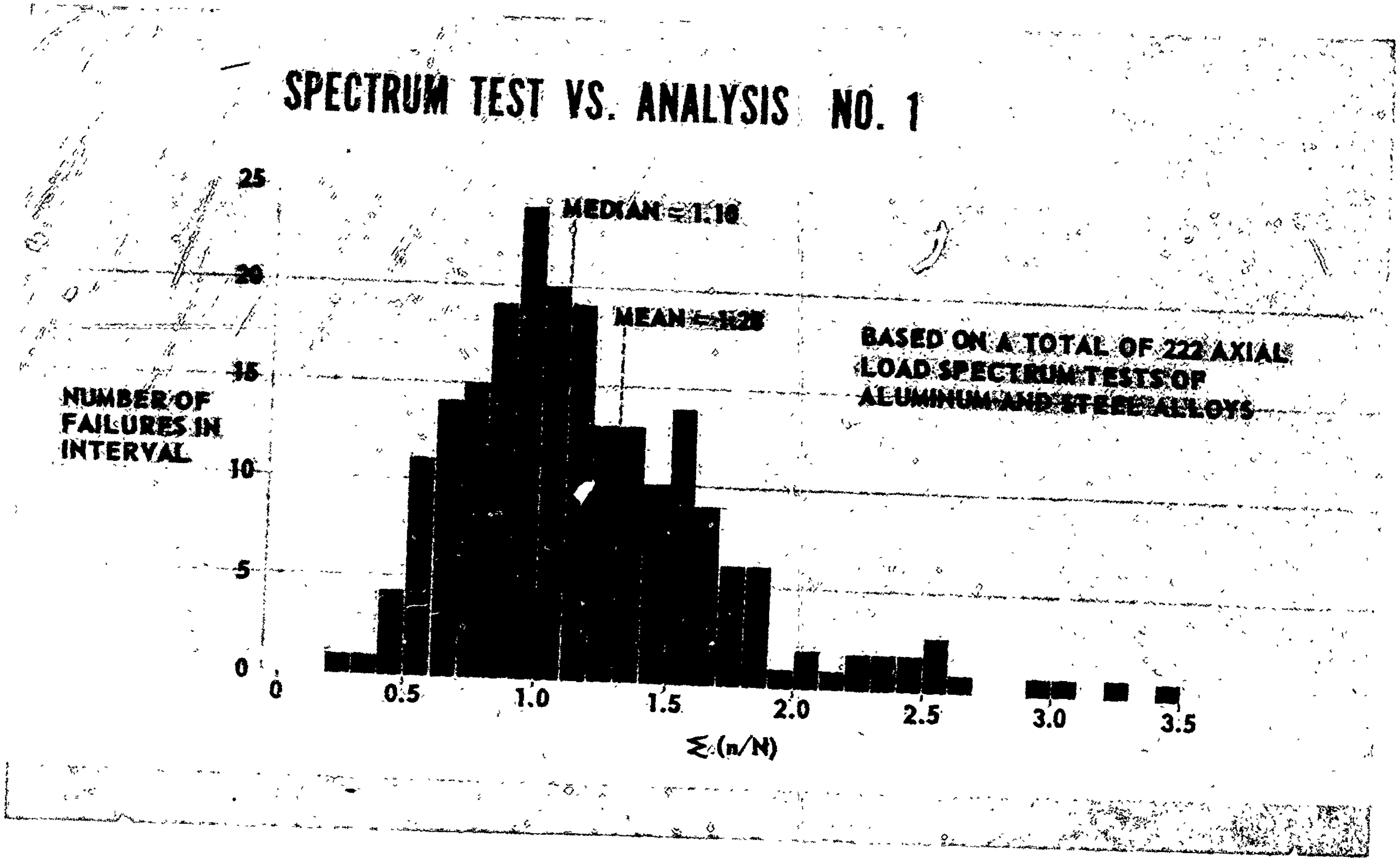
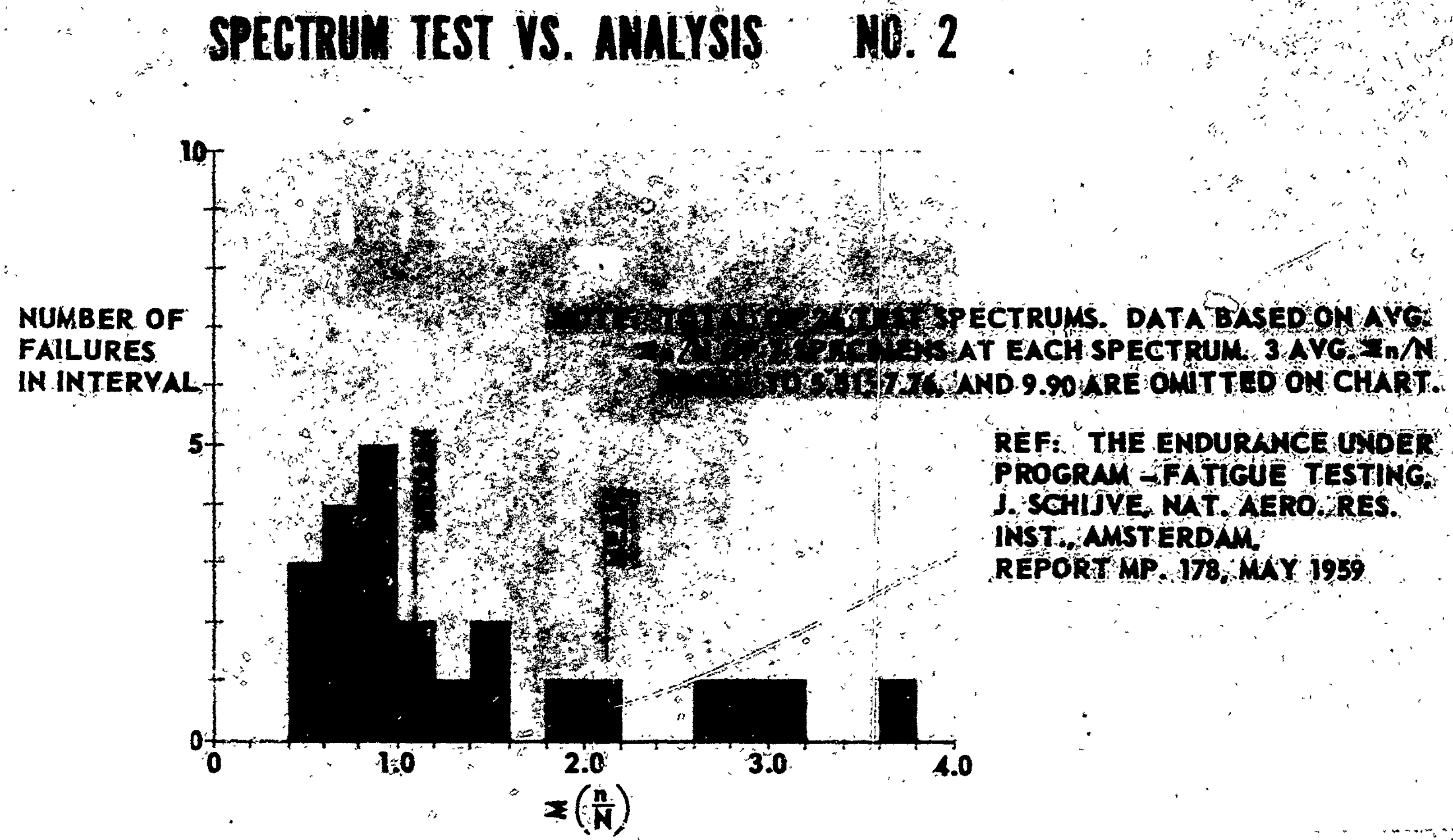


Figure 12 - Correlation Of Fatigue Test Data With Analysis Of Fatigue Damage



Fi_ure 13 - Correlation Of Program:ed Fatigue Test Data 'ith Analysis Of Fatigue Damage

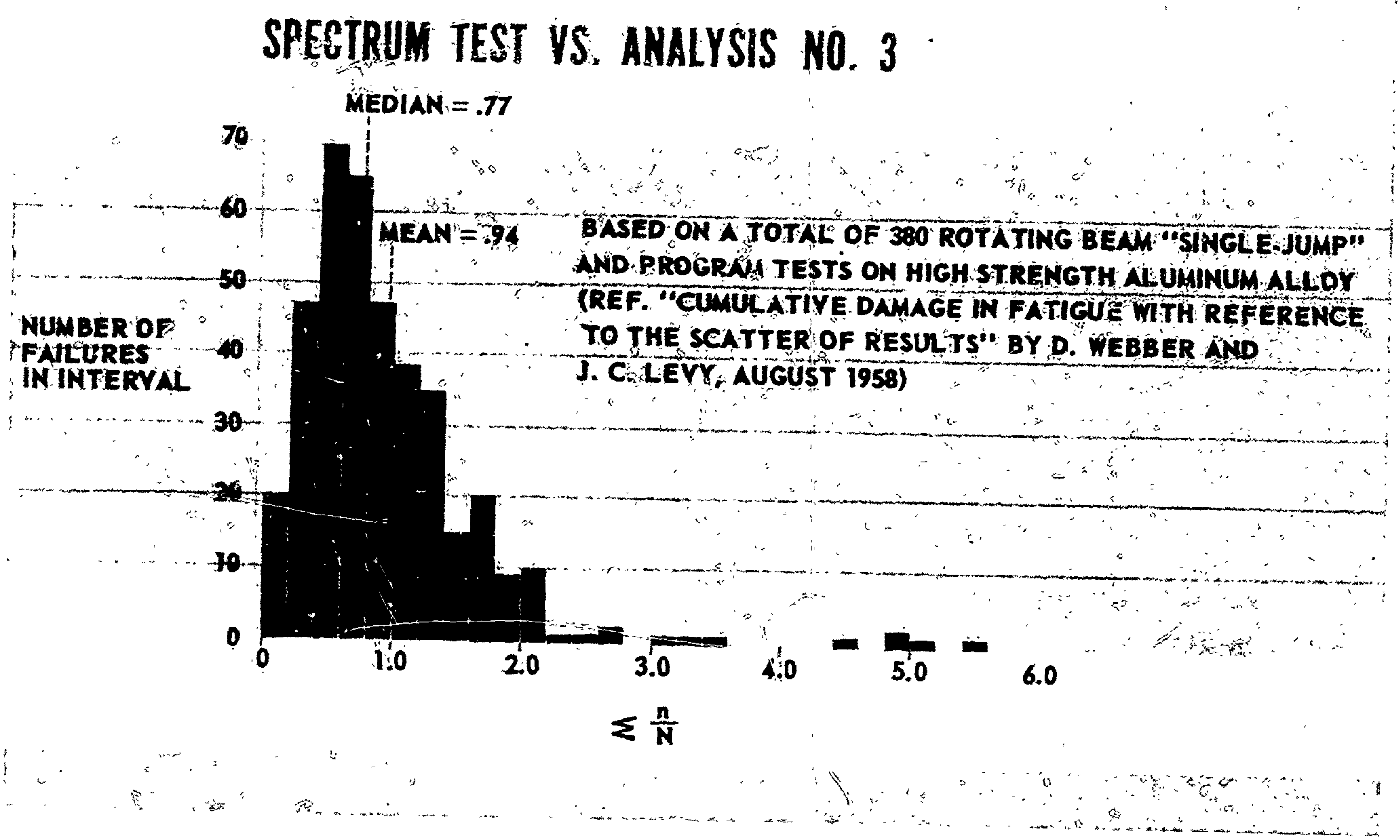


Figure 14 - Correlation Of "Single-Jump" And Two And Three Stress Level Spectrum Fatigue Test Data With Analysis Of Fatigue Damage.

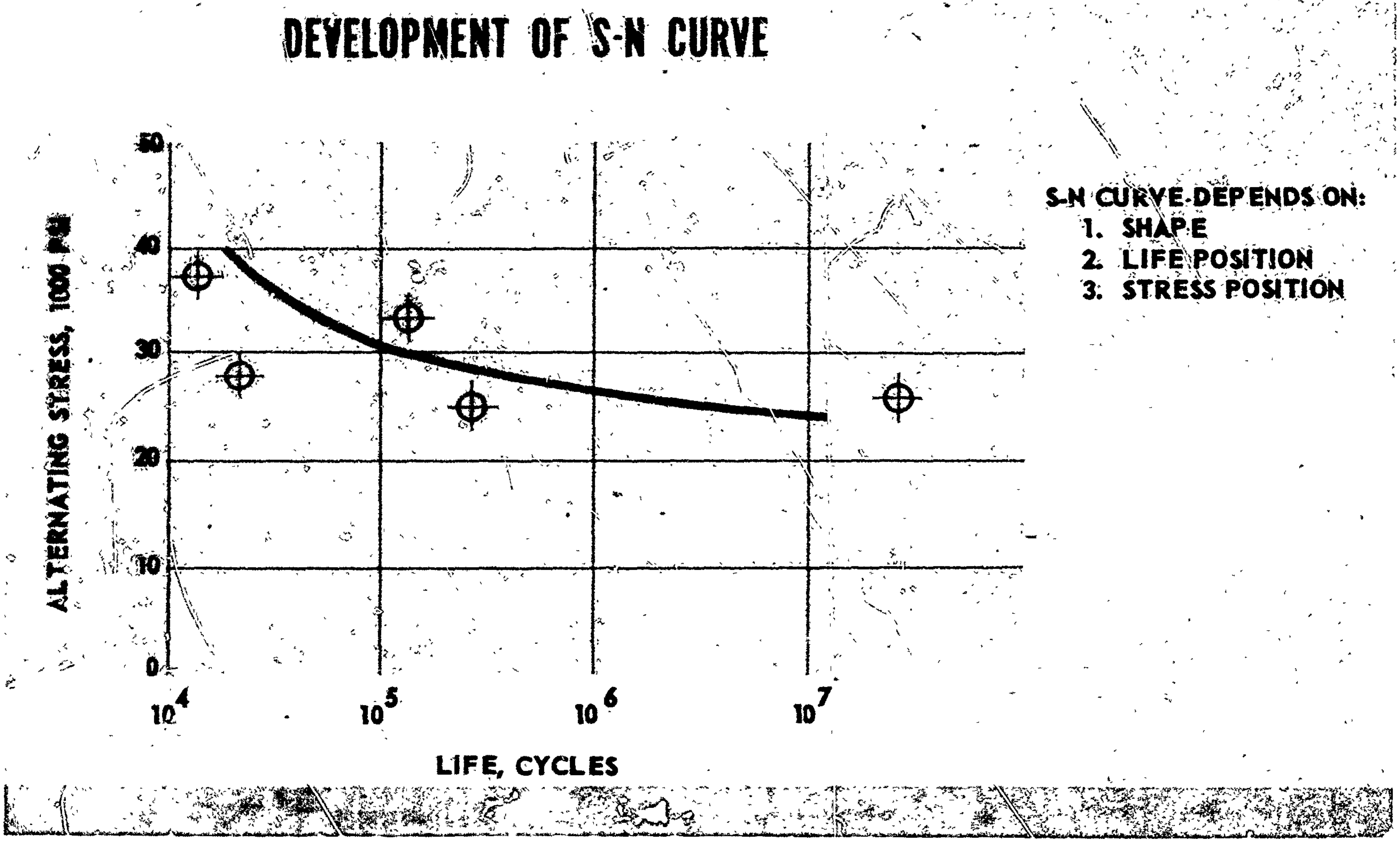


Figure 15 - Development Of S-N Curve

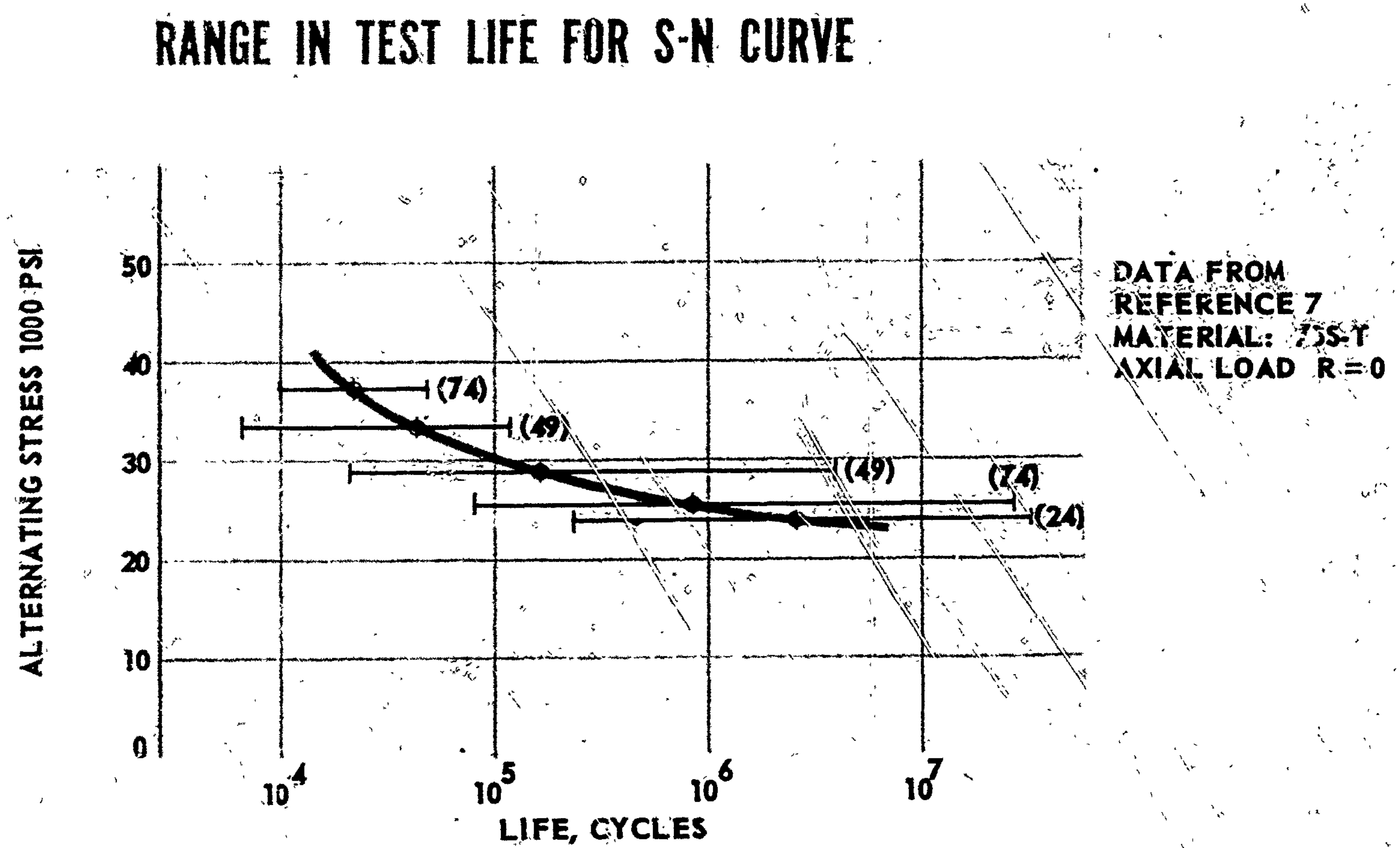
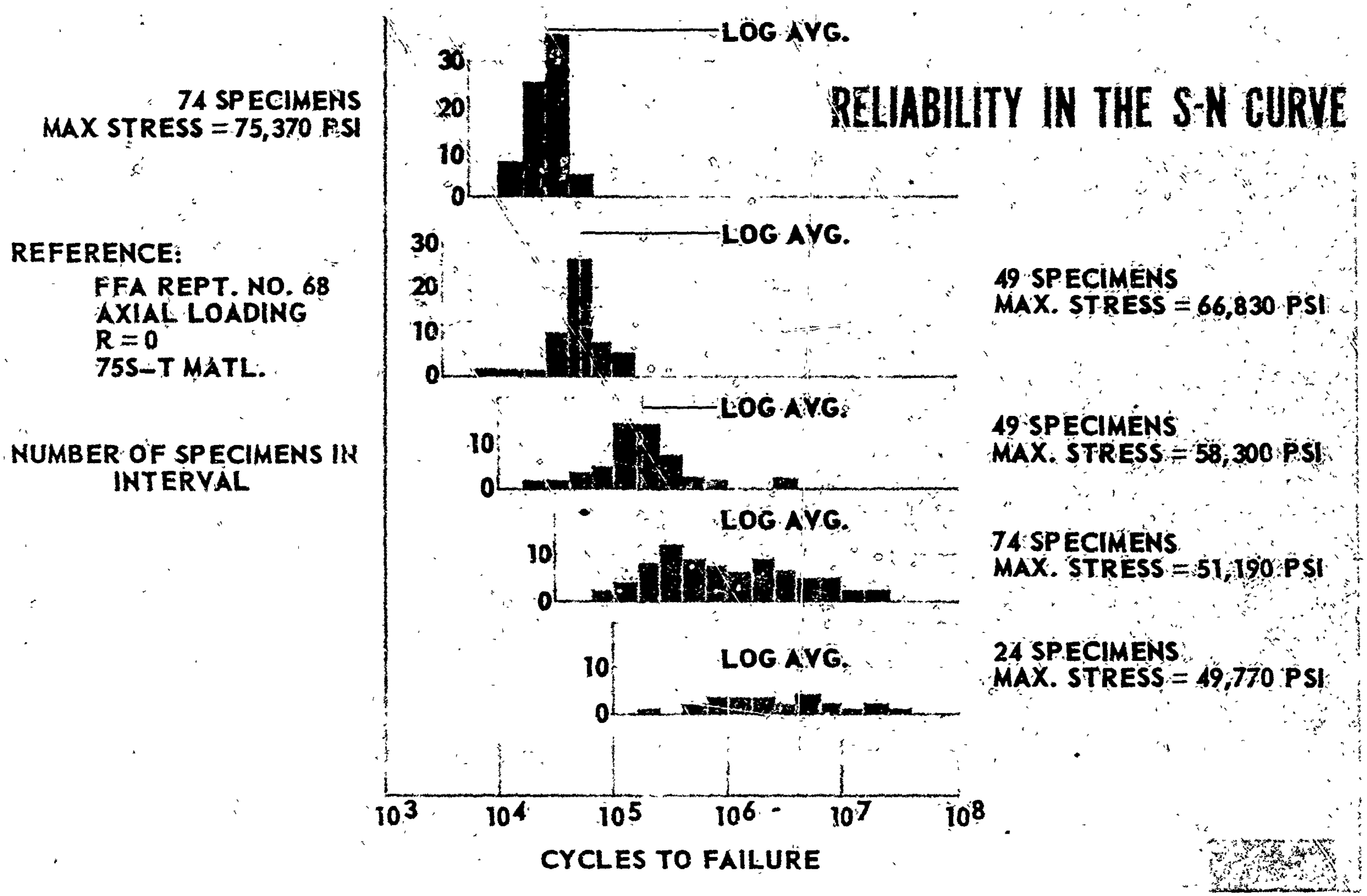
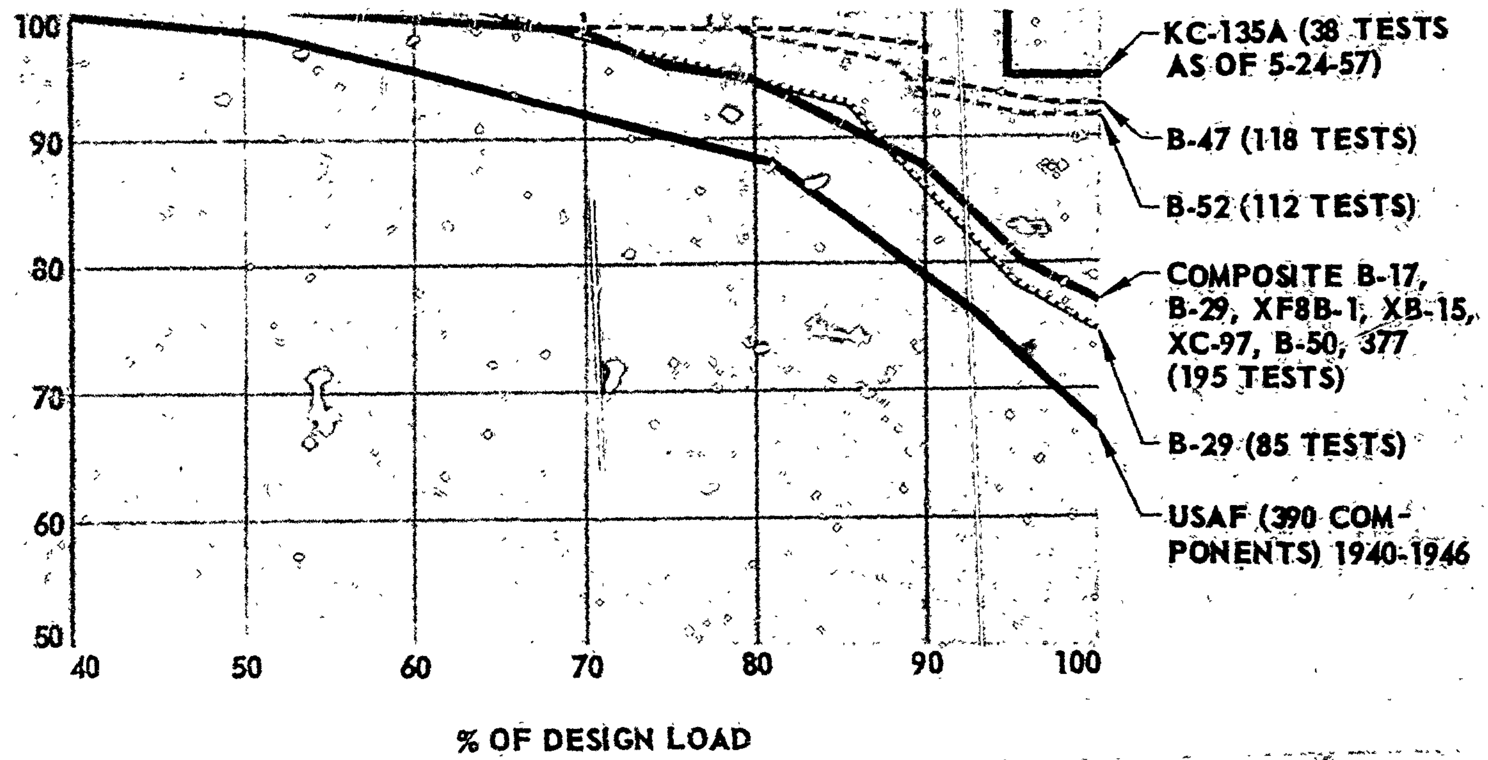


Figure 16 - Range In Test Life For S-N Curve

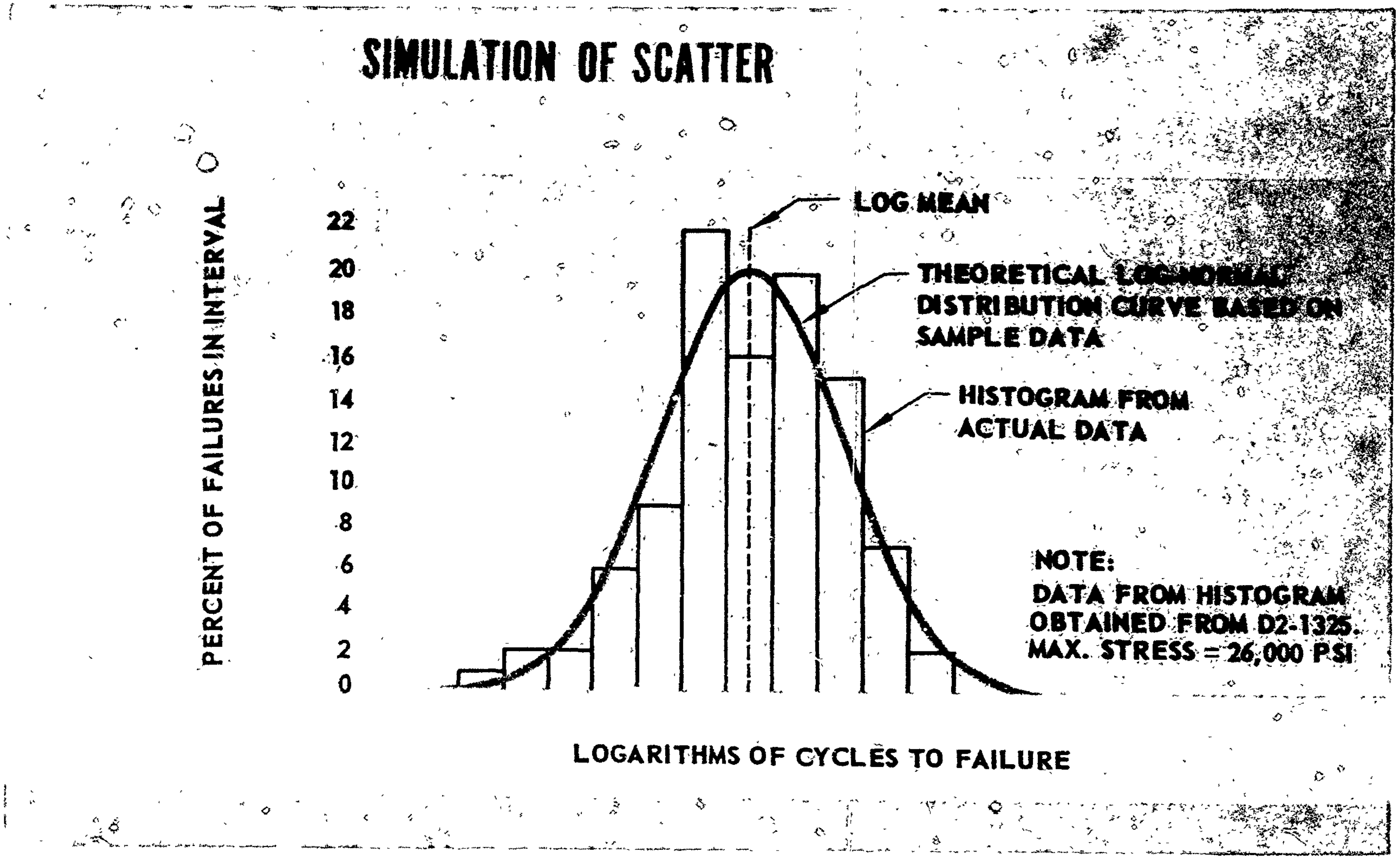


Fi~ure 17 - Reliability In The S-N Curve

SCATTER IN STATIC TESTS



…re 1… - Scatter In Static Tests



igure 19 - Simulation Of Scatter

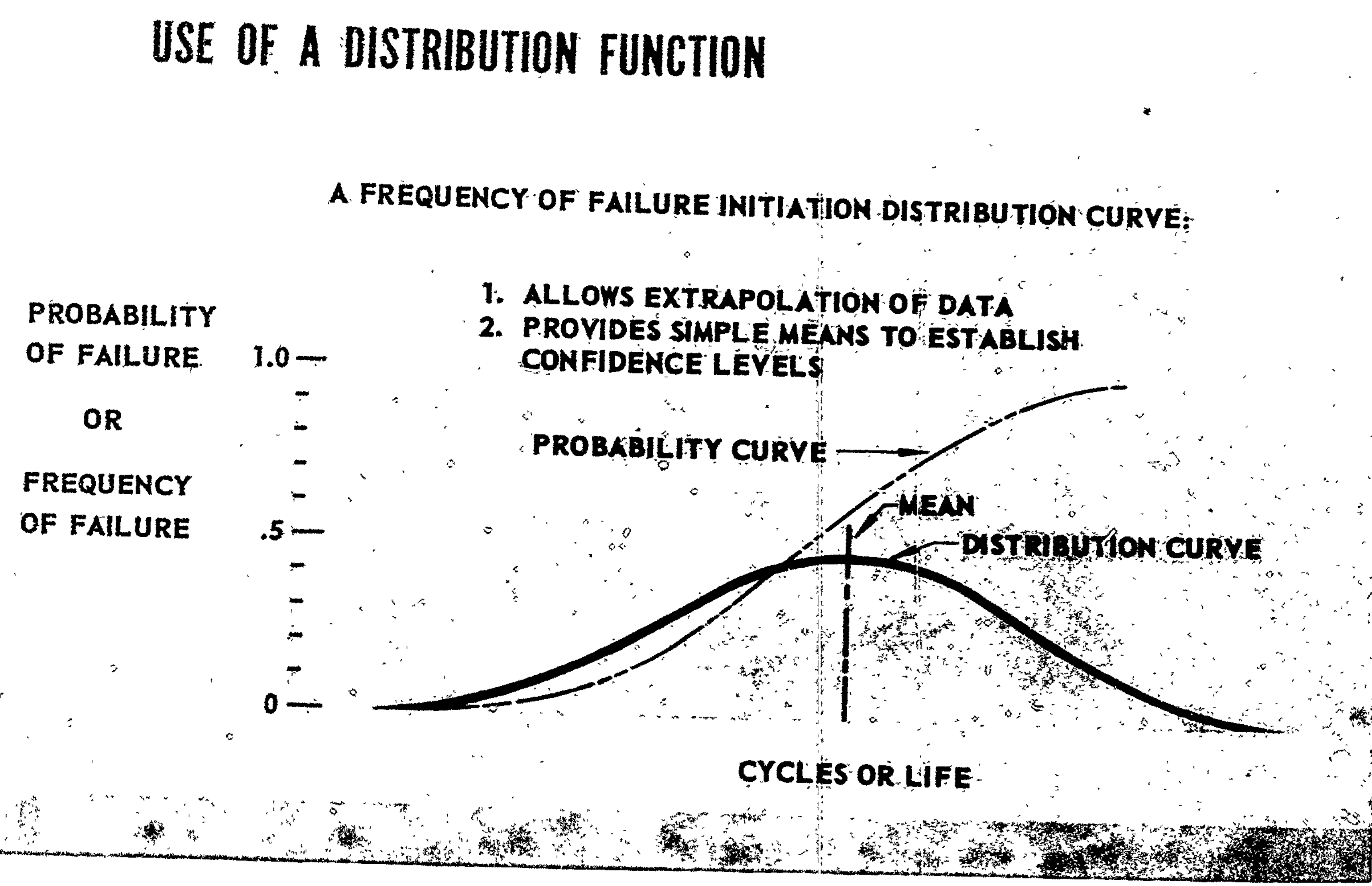


Figure 20 - The Use Of A Distribution Function

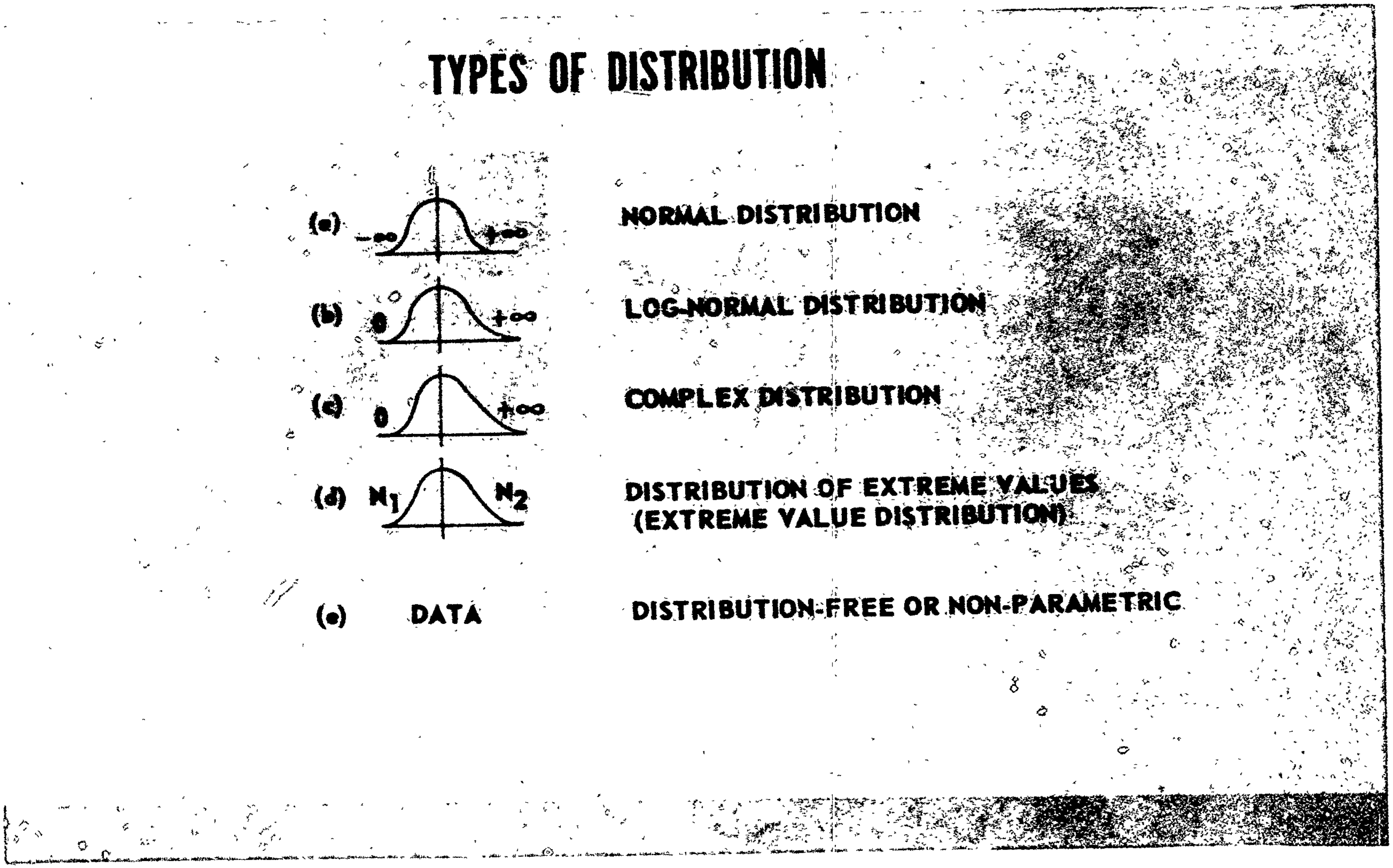


Figure 21 - Types Of Distribution

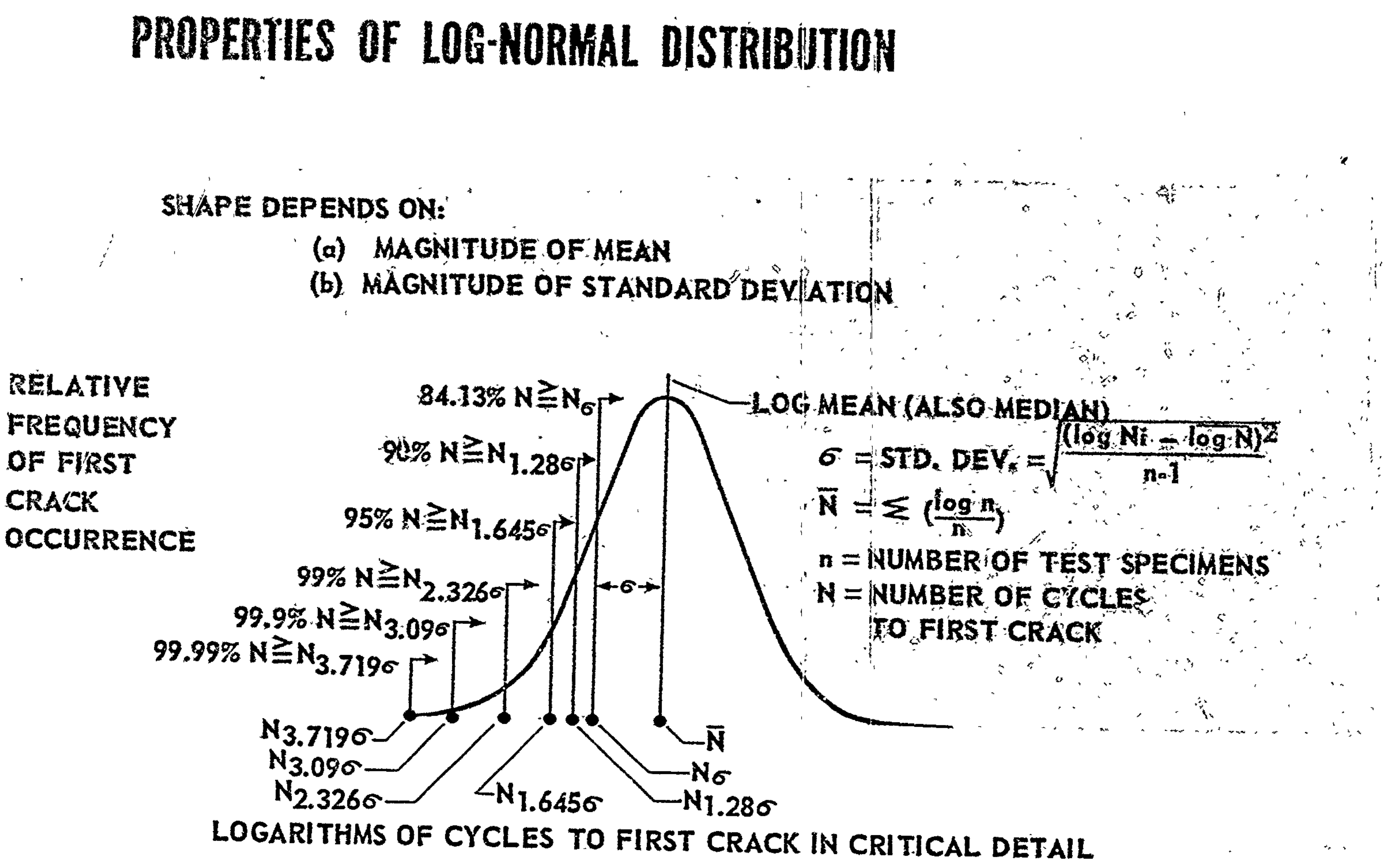


Figure 22 - Properties Of A Log-Normal Distribution

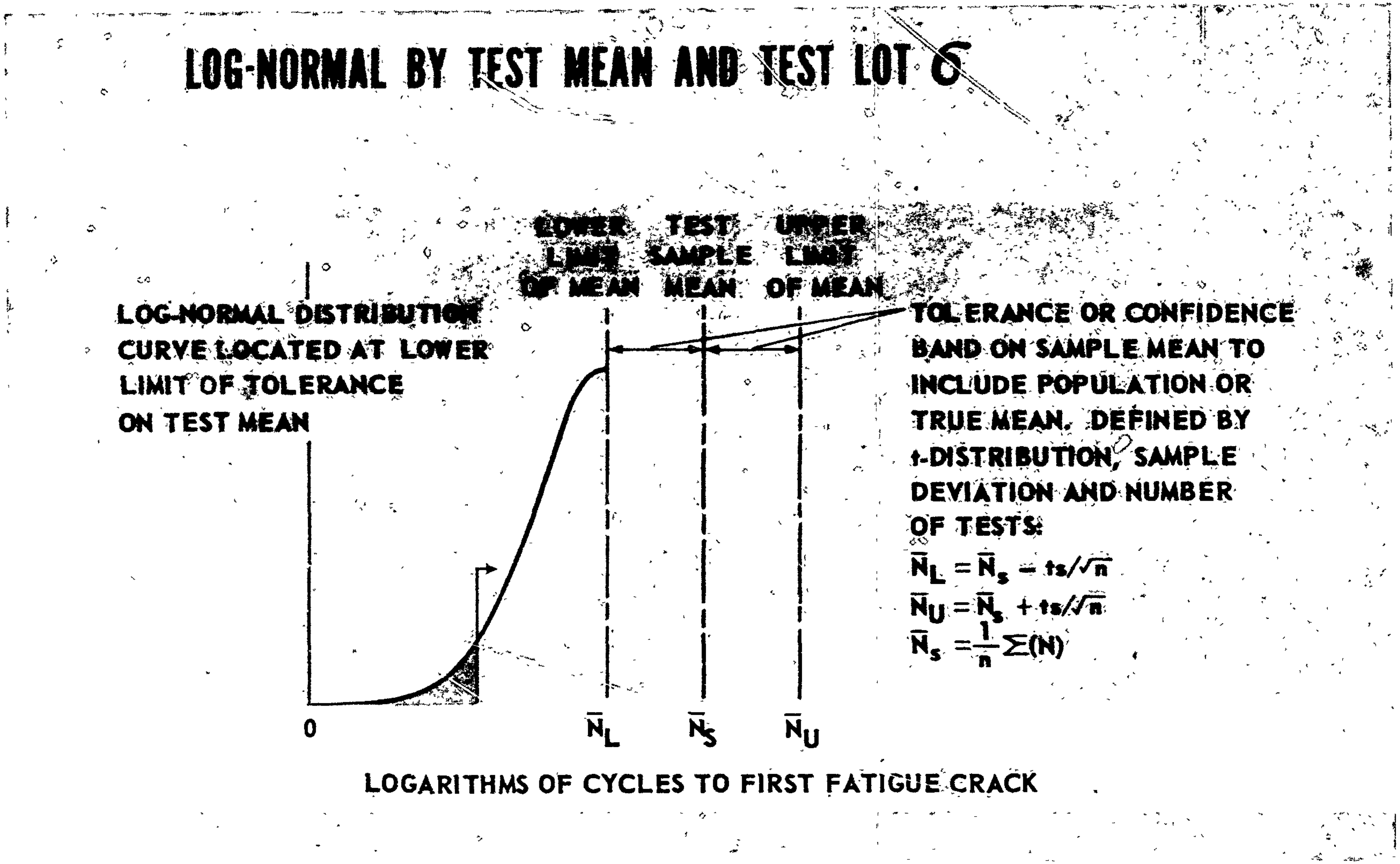


Figure 23 - Definition Of A Log-Normal Distribution Curve By Test Mean And Test Lot Standard Deviation.

LOG-NORMAL BY TEST MEAN AND KNOWN POPULATION σ

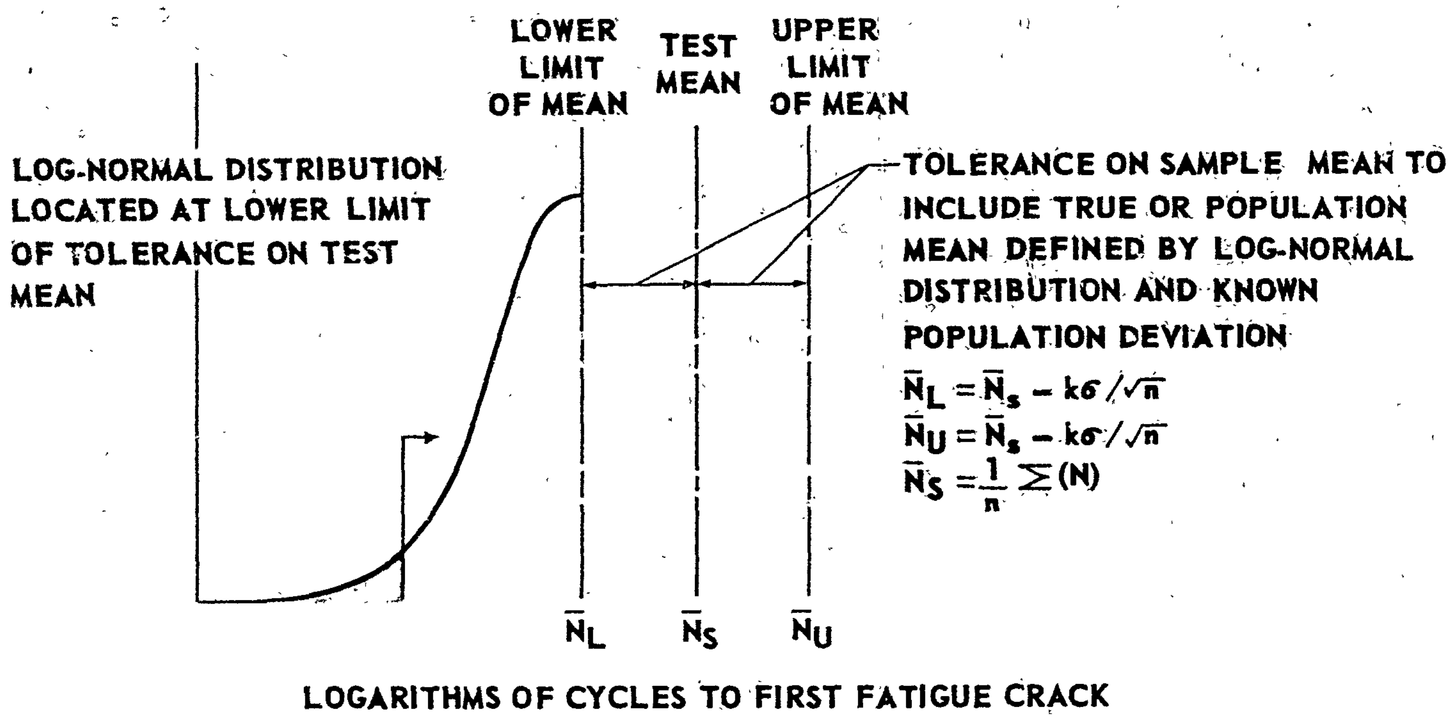


Figure 24 - Definition Of A Log-Normal Distribution Curve By Test Mean And Known Population Standard Deviation.

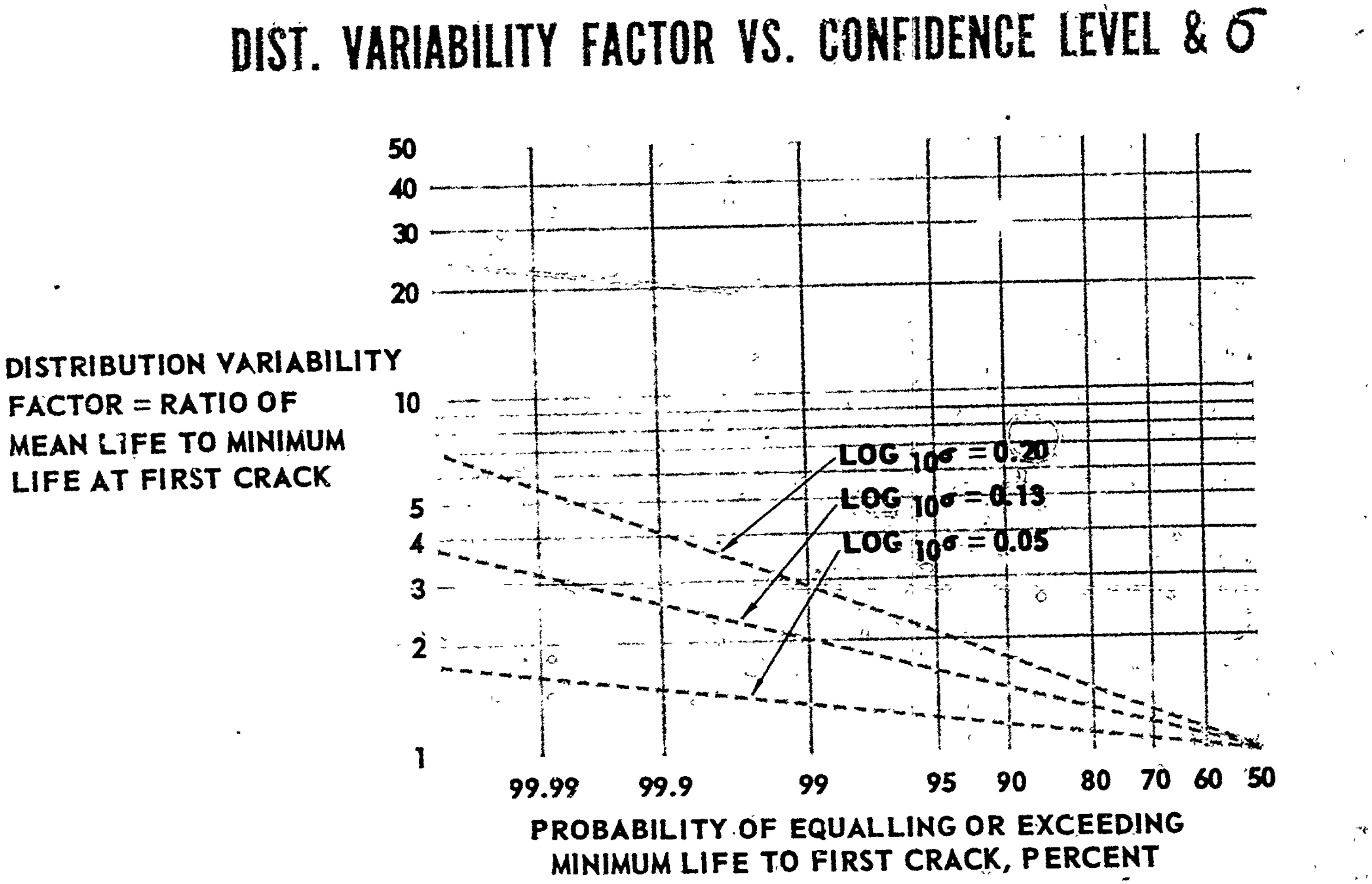


Figure 25 - Variation Of The Ratio Of Mean To Minimum Life With Confidence Level For Several Standard Deviations When Population Mean Is Known.

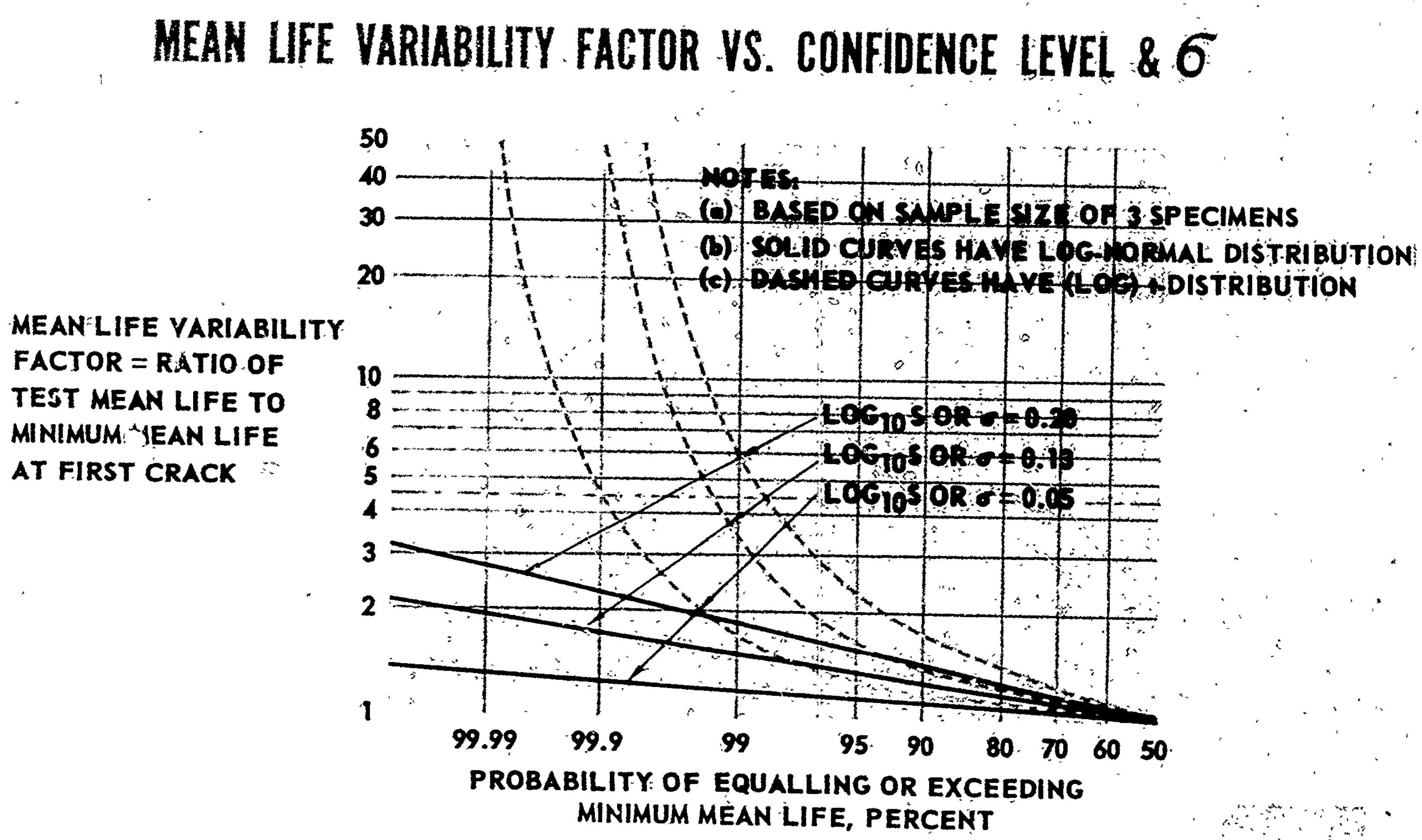


Figure 26 - Variation Of The Ratio Of Test Mean Life To Minimum Mean Life With Confidence Level And Standard Deviation.

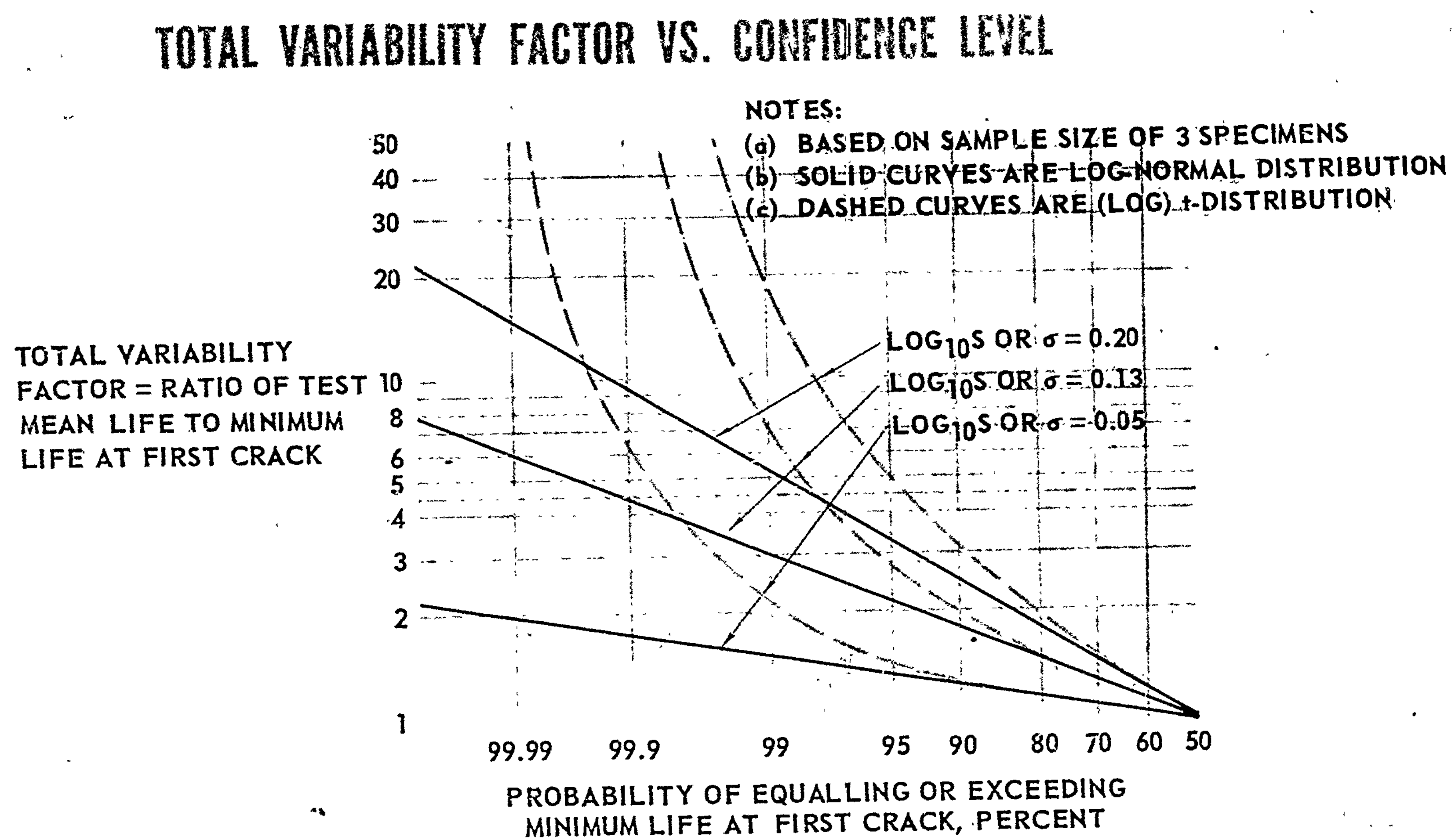


Figure 27 - Variation Of Ratio Of Test Mean Life To Minimum Life With Confidence Level And Standard Deviation.

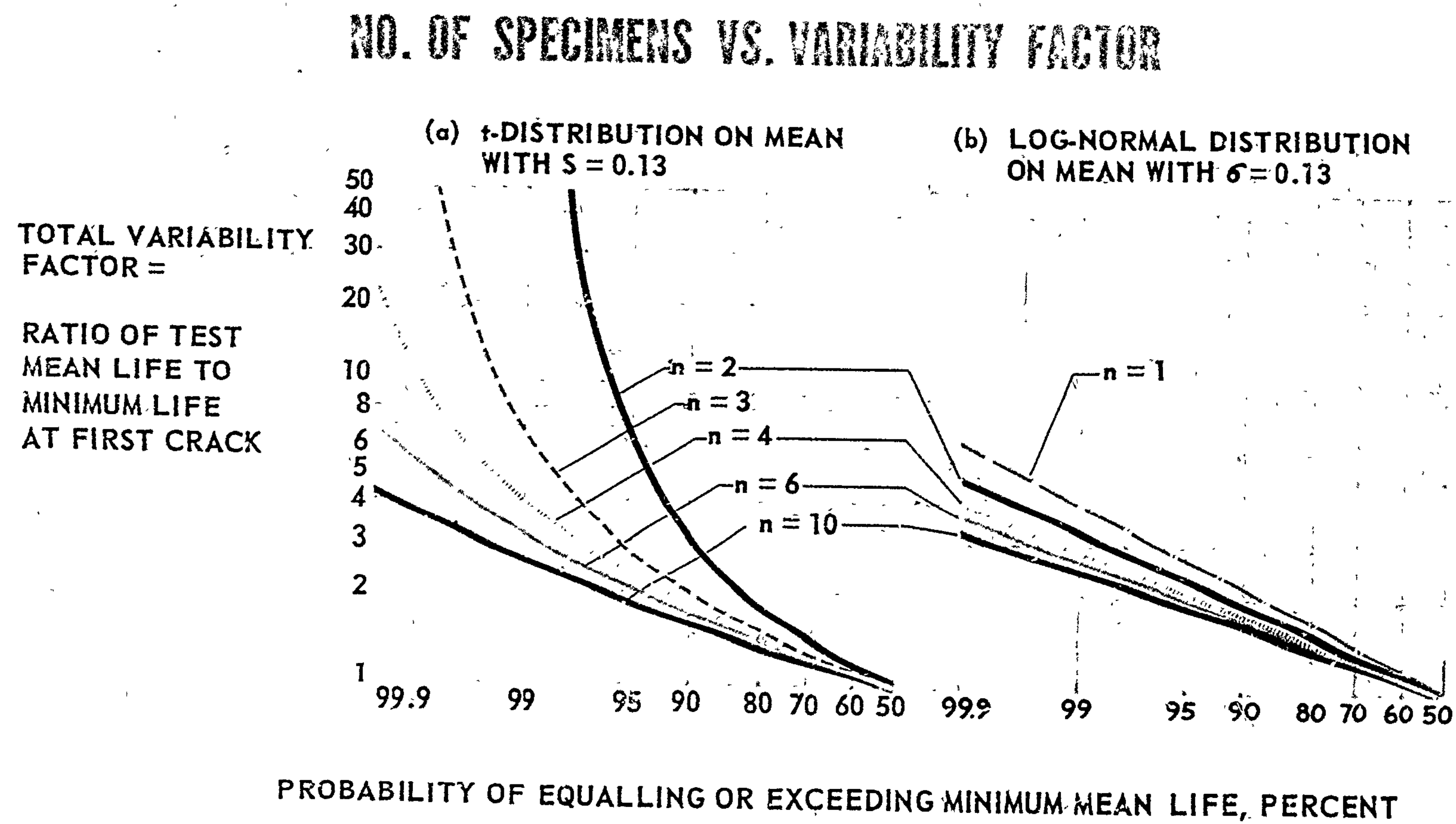


Figure 28 - Effect Of Number Of Test Specimens On Total Variability Factor

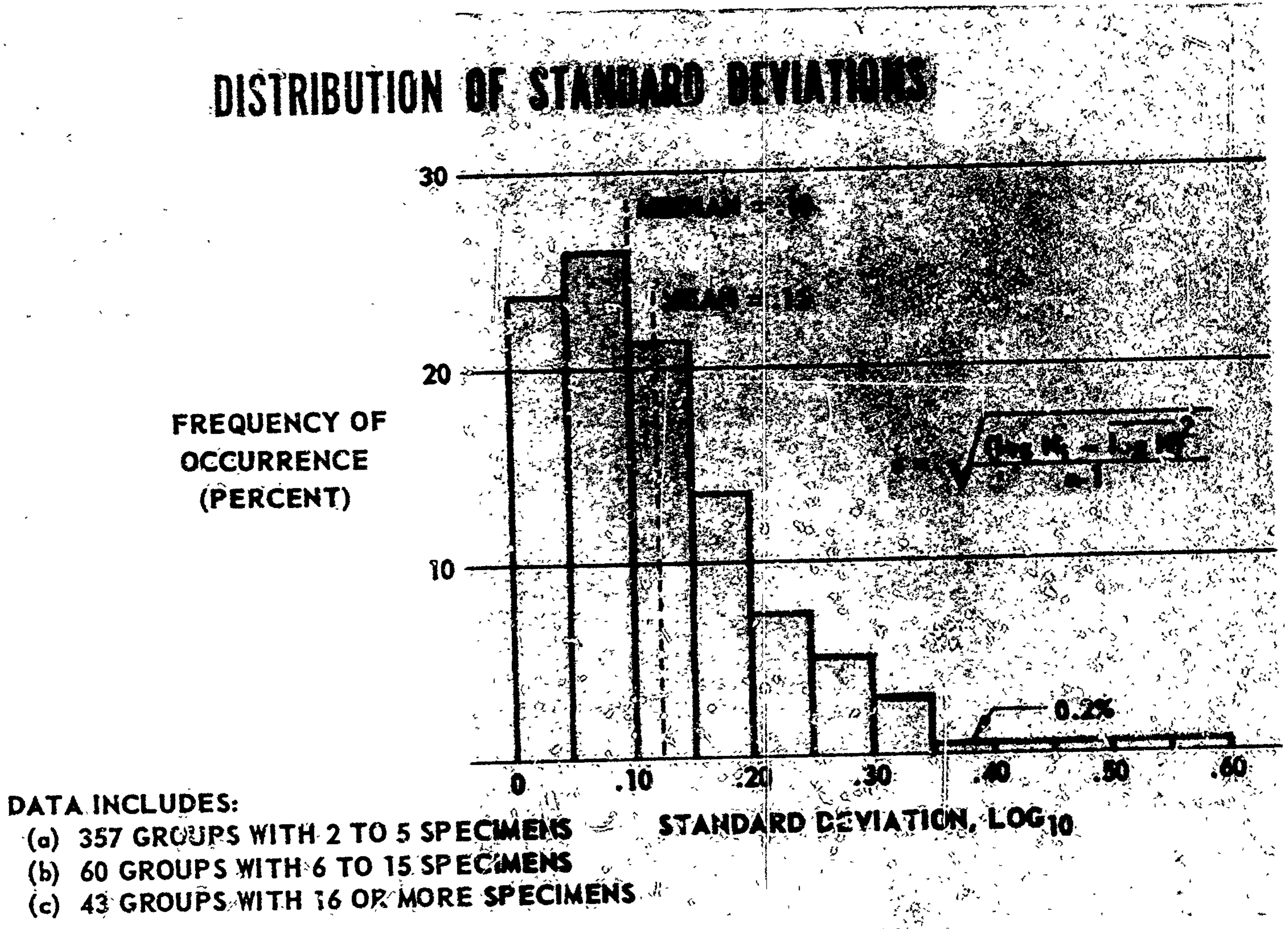


Figure 29 - Distribution Of Standard Deviations From Test Data

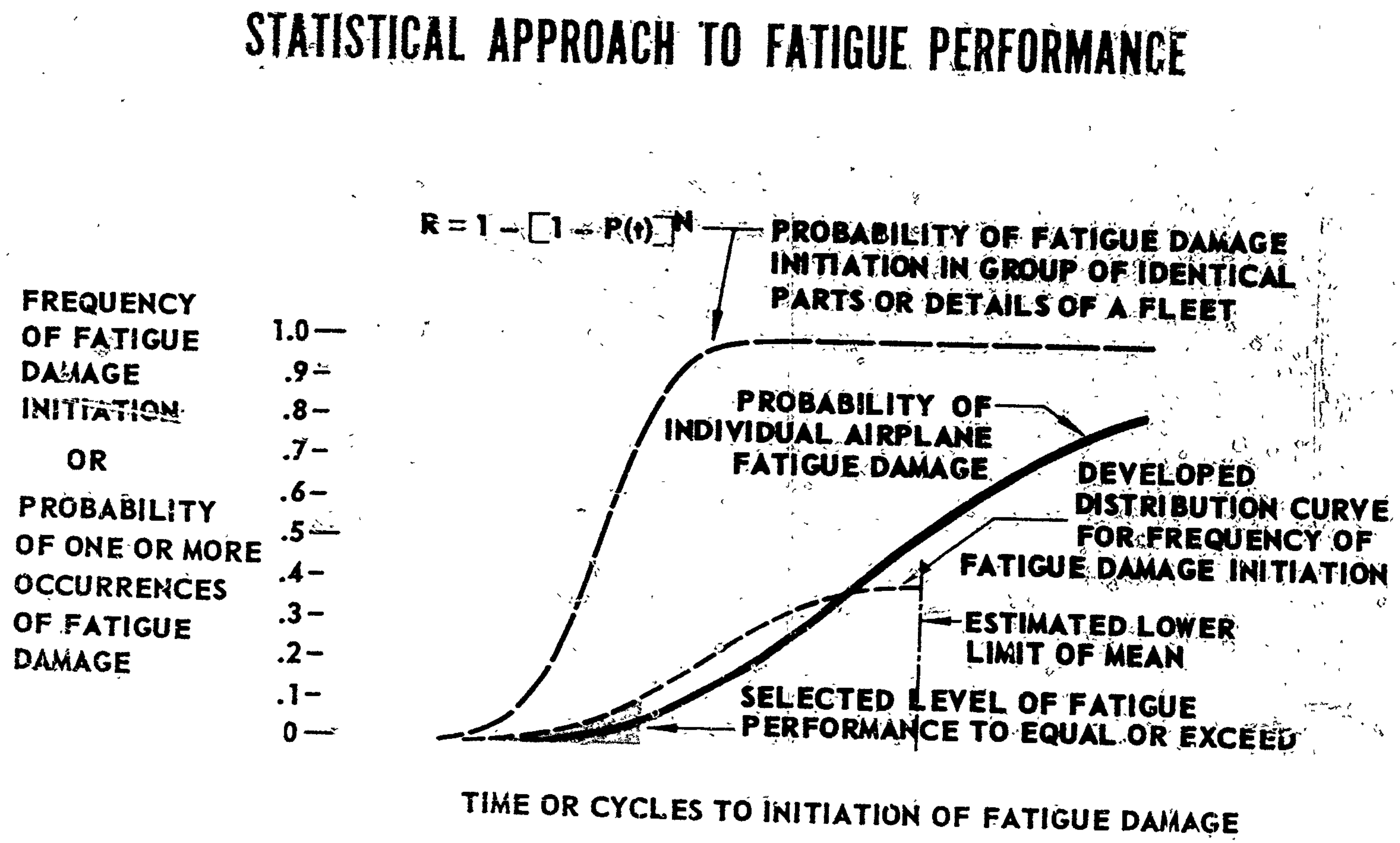


Figure 30 - Use Of Statistical Approach For Estimating Fatigue Performance Of Structure

SUMMARY OF FATIGUE SCATTER RATIOS

NUMBER OF SPECIMENS PER TEST GROUP	RATIO: $\frac{\text{MAXIMUM TEST LIFE}}{\text{MINIMUM TEST LIFE}}$		RATIO: $\frac{\text{LOGARITHMIC MEAN TEST LIFE}}{\text{MINIMUM TEST LIFE}}$	
	MEDIAN (50%)	95 PER CENT EQUAL OR LESS	MEDIAN (50%)	95 PER CENT EQUAL OR LESS
2 TO 5 SPECIMENS PER GROUP 357 GROUPS	1.6	4.2	1.25	2.1
6 TO 15 SPECIMENS PER GROUP 60 GROUPS	1.9	5.0	1.4	2.4
16 OR MORE SPECIMENS PER GP 43 GROUPS	2.6	9.5	1.7	2.7
ALL GROUPS	1.7	4.5	1.3	2.2

Table 1 – Summary Of Fatigue Scatter Factors For Various Groups Of Multiple Specimen Tests

SAFE-LIFE SCATTER FACTORS

PROPOSED FATIGUE SAFE-LIFE OR SCATTER FACTORS OF SAFETY

NUMBER OF TEST SPECIMENS	FACTOR OF SAFETY (REFERENCE 5)	FACTOR OF SAFETY (REFERENCE 6)	
		SINGLE WING	WING OF TWO IDENTICAL HALVES
1	6.0	7.50	6.41
2		5.70	5.46
3	4.5 (OR 1.5 ON MIN.)	5.17	5.15
4	—	4.90	5.02
5	—	4.75	—
6	3.5 (OR 1.5 ON MIN.)	4.62	—

Table 2 - Summary Of Scatter Factors For Calculating Safe-Life

MIN. LIVES VS. DISTRIBUTION FUNCTION

PROBABILITY OF EQUALLING OR EXCEEDING GIVEN LIFE (IN PERCENT)	50	90	95	99	99.9	99.99
LOG NORMAL MEAN (LOG) = 393 STD. DEV. = 0.0702	393	319	301	270	238	214
TWO PARAMETER $\hat{k} = 39$ $\hat{a} = 10.202 \times 10^3$ (MEAN = 398)	394	318	299	270	229	203
EXTREME VALUE CHARAC. LIFE = 420 MIN. LIFE = 193	398	315	293	257	227	211
LOG-NORMAL WITH t-DISTRIBUTION TOLERANCE ON MEAN	393	312	293	260	227	203
LOG-NORMAL WITH TOLERANCE ON MEAN	393	313	293	260	227	203

NOTE: CALCULATED LIVES GIVEN IN 1000 CYCLES. DATA REPRESENTS 102 SPECIMENS FLEXURALLY TESTED AT ± 26,000 PSI.

Table 3 - Comparison Of Minimum Lives Calculated By Several Distribution Functions

SUMMARY OF FATIGUE SCATTER FACTORS

NUMBER OF TEST SPECIMENS	CIVIL FACTORS OF SAFETY FOR SAFE-LIFE (ARB)	SUMMARY OF STATISTICALLY DERIVED SCATTER FACTORS					
		POPULATION ASSUMED LOG-NORMAL WITH 95% CONFIDENCE ON MEAN AND DISTRIBUTION			POPULATION ASSUMED LOG-NORMAL WITH KNOWN STANDARD DEVIATION AND 95% CONFIDENCE LEVEL		
		$S = 0.05$	$S = 0.13$	$S = 0.20$	$\sigma = 0.05$	$\sigma = 0.13$	$\sigma = 0.20$
1	6	——	——	——	1.5	2.7	4.5
2	——	2.0	6.5	16.0	1.4	2.3	3.7
3	4.5	1.5	2.7	4.7	1.4	2.2	3.2
4	——	1.4	2.3	3.7	1.4	2.1	3.1
5	——	1.4	2.2	3.4	1.3	2.1	3.0
6	3.5	1.3	2.1	3.1	1.3	2.0	2.9

Table 4 - Summary Of Fatigue Scatter Factors

SUMMARY OF PROCEDURES FOR APPLICATION OF AN ENGINEERING STATISTICAL APPROACH TO FATIGUE ANALYSIS

1. ESTIMATION OF LOAD HISTORY.
2. ESTIMATION OF AVERAGE FATIGUE – PERFORMANCE BY S-N CURVE.
3. COMPUTATION OF AVERAGE FATIGUE LIFE (LINEAR CUMULATIVE DAMAGE THEORY)
4. SELECTION FACTOR OF SAFETY TO OBTAIN DESIRED RELIABILITY OF INDIVIDUAL FATIGUE PERFORMANCE.
5. PROVISION OF ADDITIONAL FACTORS OF SAFETY TO ACCOUNT FOR VARIABILITY IN LOAD HISTORY AND PHYSICAL ENVIRONMENT.
6. EVALUATION OF EXTENT OF PROBABLE FATIGUE DAMAGE IN TOTAL GROUP OF PARTS (OR AIRPLANES).
7. ESTABLISHMENT OF INSPECTION PROCEDURES TO CONTROL EXTENT OF FATIGUE DAMAGE.

Table 5 - Summary Of Procedures For Application Of An Engineering Statistical Approach To Fatigue Analysis

ANALYTICAL APPROACH TO STRESS CONCENTRATION EFFECT IN FATIGUE OF AIRCRAFT MATERIALS

by

R. E. Peterson

Westinghouse Research Laboratories
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Available data on stress concentration effects in aluminum alloys and steels have been reviewed and compared with theoretical stress concentration factors. An attempt has been made to provide notch sensitivity factors which can be utilized in design to estimate the effect of geometrical variations for which specific data do not exist.

INTRODUCTION

This presentation has the following aims: 1) to bring the published summaries of fatigue notch sensitivity values up to date, 2) to attempt an analytical interpretation of the results, and 3) to seek a simple formula and constants which can be used in design. This latter objective is being pursued in a number of technical fields as a phase of utilization of computers in design and optimization of design.

DEFINITIONS

Many readers will be sufficiently familiar with the subject[1,2,3,27] to skip this section.

Designers calculate nominal stresses from given loads, moments and torques. These are the simplest possible calculations, based on net section and uniform or linear stress distribution. The presence of a hole, groove, or other geometrical variation introduces a localized stress peak (Fig. 1a). A measure of the intensity of this

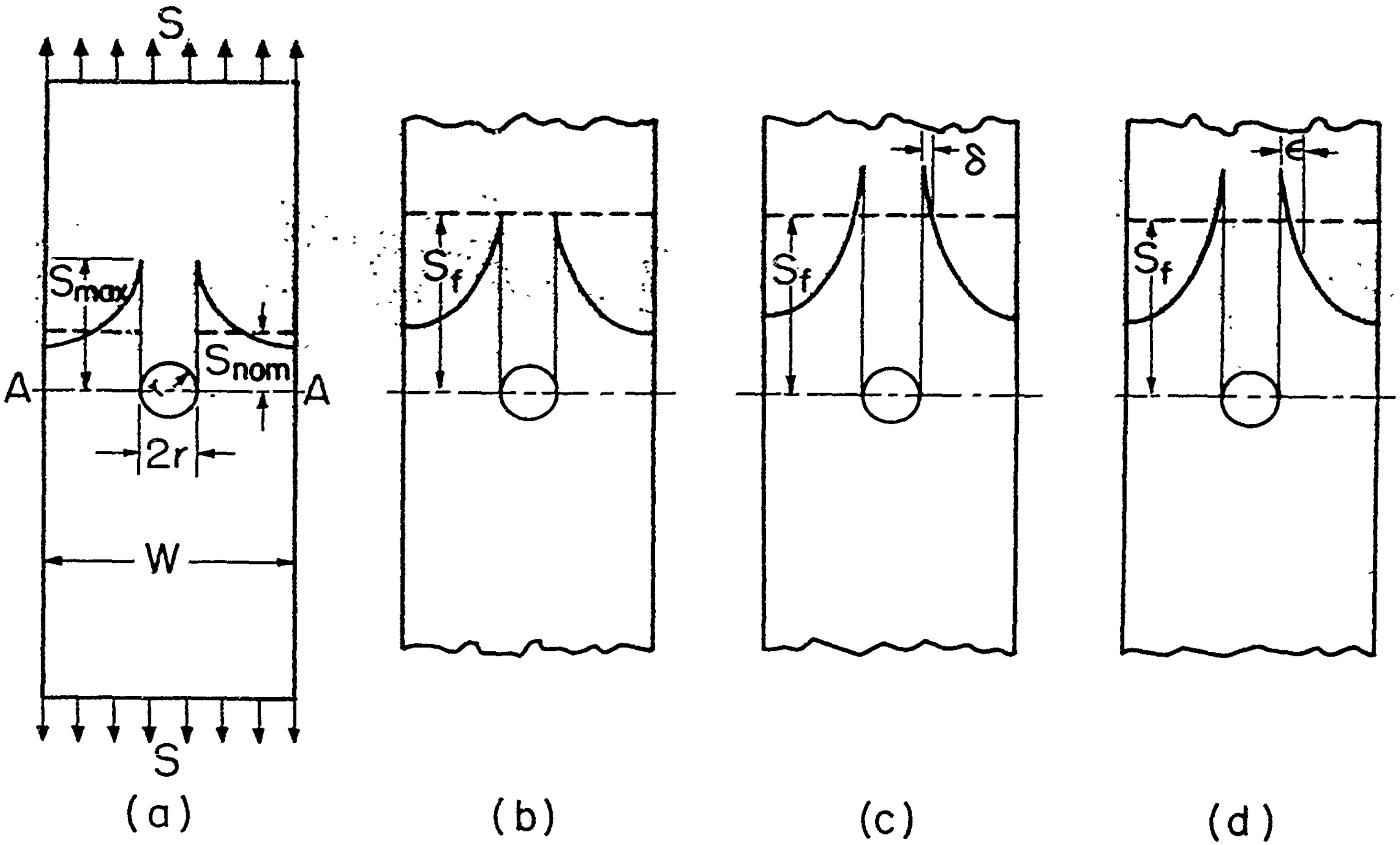


Fig. I — Nomenclature and failure criteria for notched specimen

"disturbance" of the simple stress field is the stress concentration factor, which is defined as the maximum stress over the nominal stress, $K_t = S_{max}/S_{nom}$, where S_{nom} is the average net stress acting on the section A - A, or $S_{nom} = S/(1-2r/W)$. This factor is based on the theory of elasticity and is therefore determined by geometry and type of loading.

Now, turning to fatigue testing, the effect of a geometrical disturbance (the generic term is "notch") is expressed as the fatigue notch factor, K_f , defined as the fatigue strength of an unnotched specimen over the fatigue strength of a notched specimen. If there is no effect of the notch, $K_f = 1$, and if full theoretical effect is obtained, $K_f = K_t$.

Notch sensitivity is defined as follows: $q = (K_f - 1)/(K_t - 1)$. The reason for subtracting unity is to provide a scale where for no notch effect, or for a notch effect small compared to theoretical, $q = 0$, and for full theoretical effect, $q = 1$. The condition $q = 0$ will be explained in more detail since it is not correct to say that this always corresponds to no notch effect. For a hole or a notch of constant shape (depth to radius constant), a condition of no notch effect is obviously reached as r goes to zero since the hole or notch vanishes. However, where the notch depth is constant and the notch radius is decreased a finite K_f is reached while at the same time K_t is increasing toward infinity. In this case $q = 0$ must be regarded as a <u>limit value</u> which is continuously approached with decreasing r. But, regardless of geometry, it can be generally stated that <u>notch sensitivity is a measure of the degree to which theoretical stress concentration effect is realized in a corresponding fatigue test.</u>

FATIGUE DATA

In Figs. 2, 3 and 4, the points represent values of q for fatigue tests of aluminum alloy* specimens containing holes, grooves or shoulder fillets.[4-14] All tests were of completely alternating type, i.e., mean stress zero. The basic fatigue strength used was that of the smallest specimen where a range of sizes was tested.[21] Due to the considerable inherent scatter of fatigue data, better accuracy is obtained for tests corresponding to higher values of stress concentration. In the present analysis only the results of test specimens with K_t of 2 or greater were used. Results for aluminum alloys correspond to values at $500\ (10)^6$ cycles, or in other instances the largest number of cycles available from the data.

* The material designations 24 S-T and 75 S-T, used in most of the references, have also been used in this paper. The corresponding present designations are 2024-T3, 7075-T6.

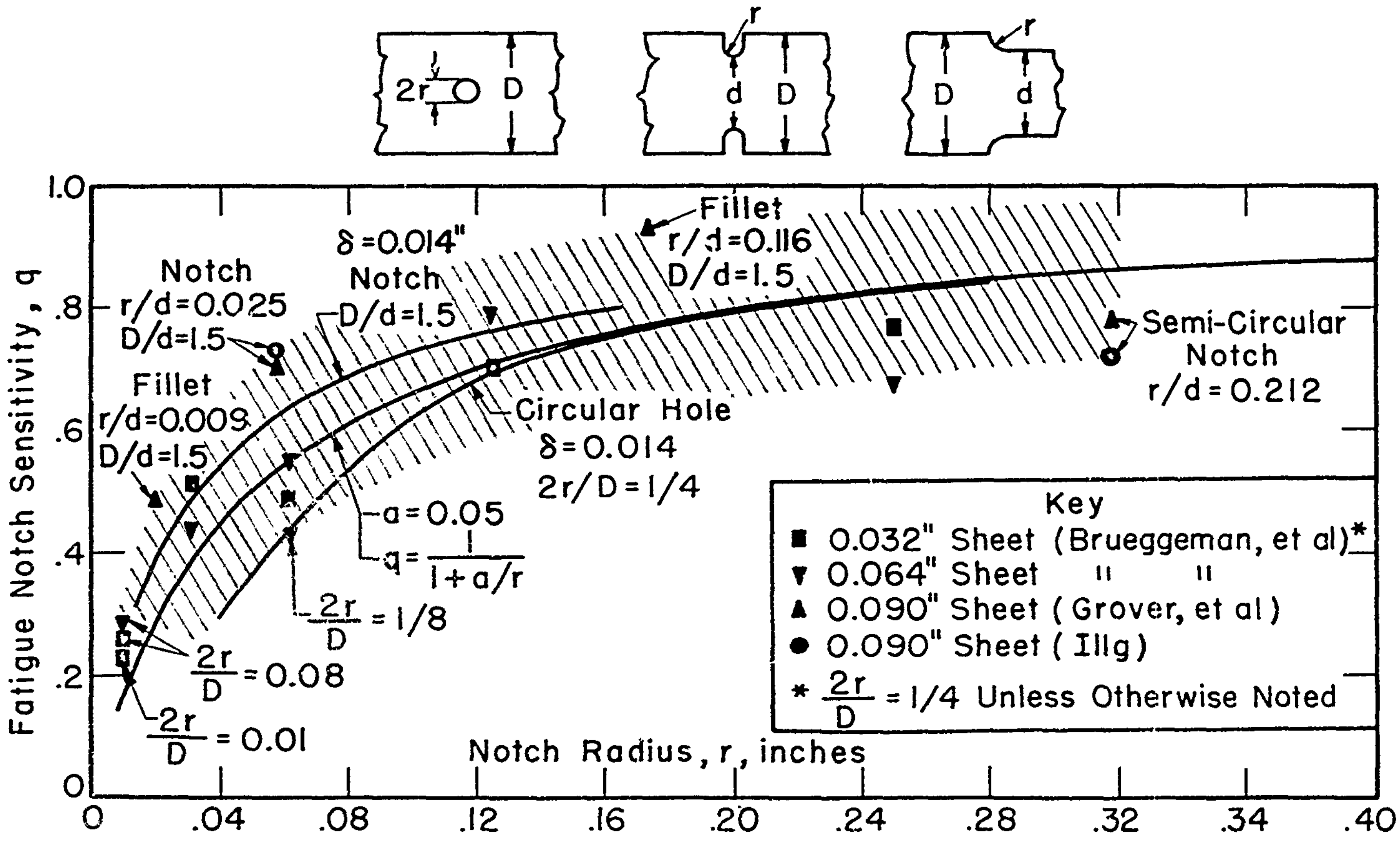


Figure 2- Fatigue notch sensitivity
24S-T aluminum alloy sheet
Axial test; zero mean stress

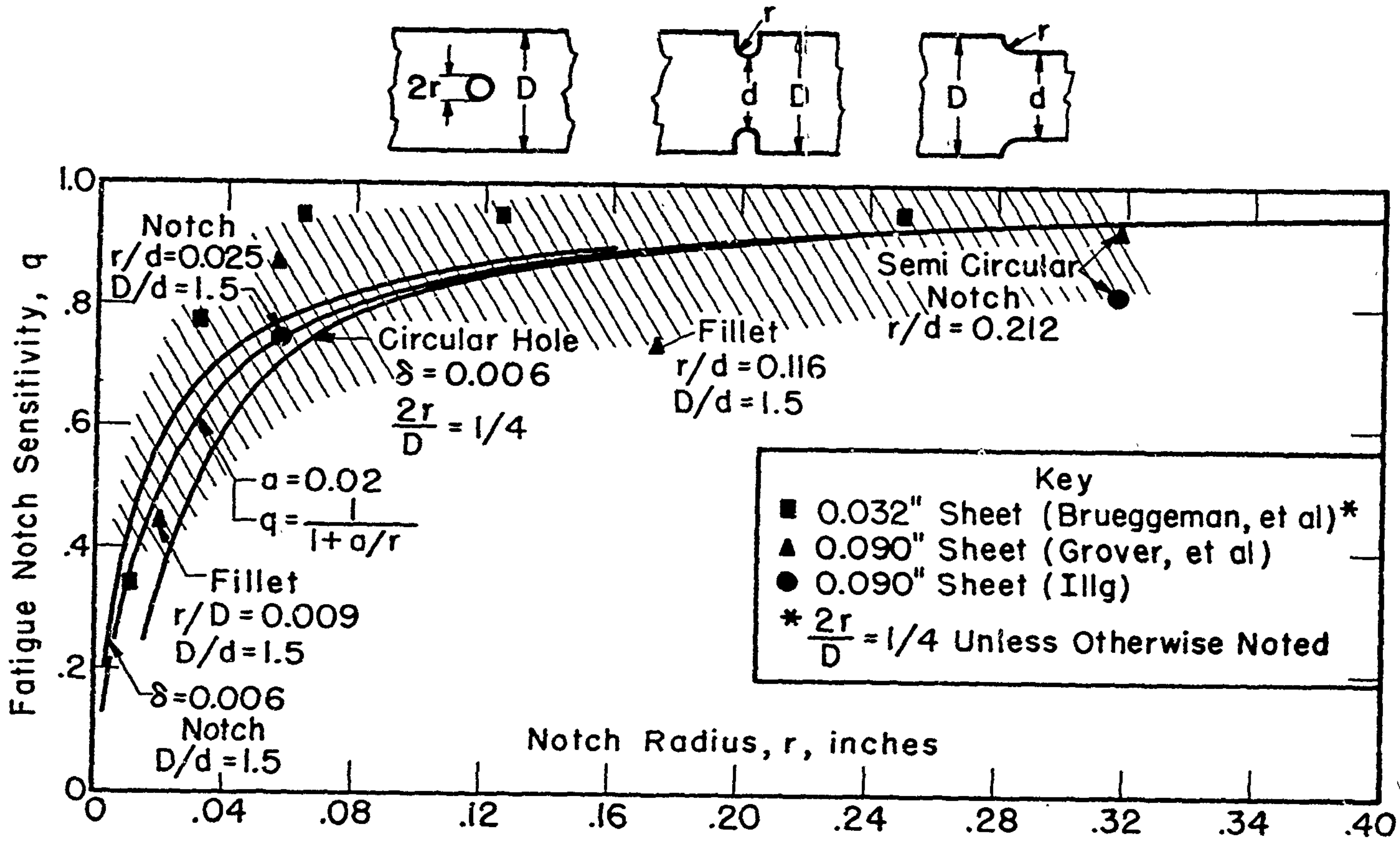


Figure 3- Fatigue notch sensitivity
75S-T aluminum alloy sheet
Axial test; zero mean stress

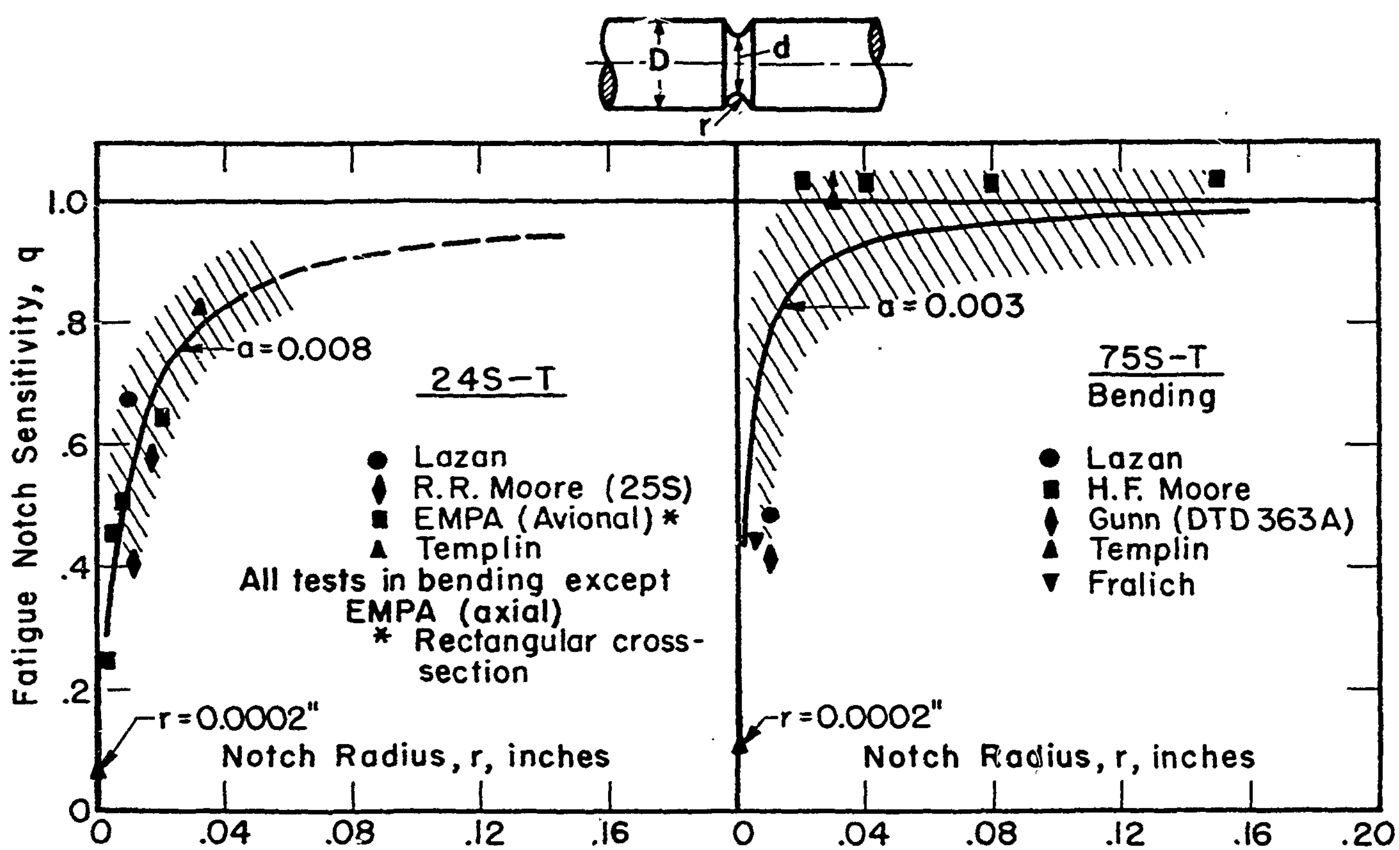


Figure 4 - Fatigue notch sensitivity
Aluminum alloy
Round specimens
(zero mean stress)

For steels considerable test data exist for the lower strength steels,[15] but very little has been available for high strength steels. Recently, a very extensive investigation of the fatigue of high strength steels has been made by Curtiss-Wright for WADC.[16] Data were available for 4340 steel at tensile strengths of 150, 190, 230 and 260 ksi and for 4350 at 300 ksi. Typical data for the latter case are shown in Figs. 5 and 6.

ANALYTICAL INTERPRETATION

Referring to Fig. 1a, from elastic theory we obtain the result that the maximum stress is about three times the applied stress. What happens when we make a fatigue test? If the theoretical factor applied we would expect the fatigue strength of the specimen with the hole to be 1/3 that of a specimen without a hole. Usually the fatigue strength is higher than the 1/3 value. If we think about it, this result seems reasonable.

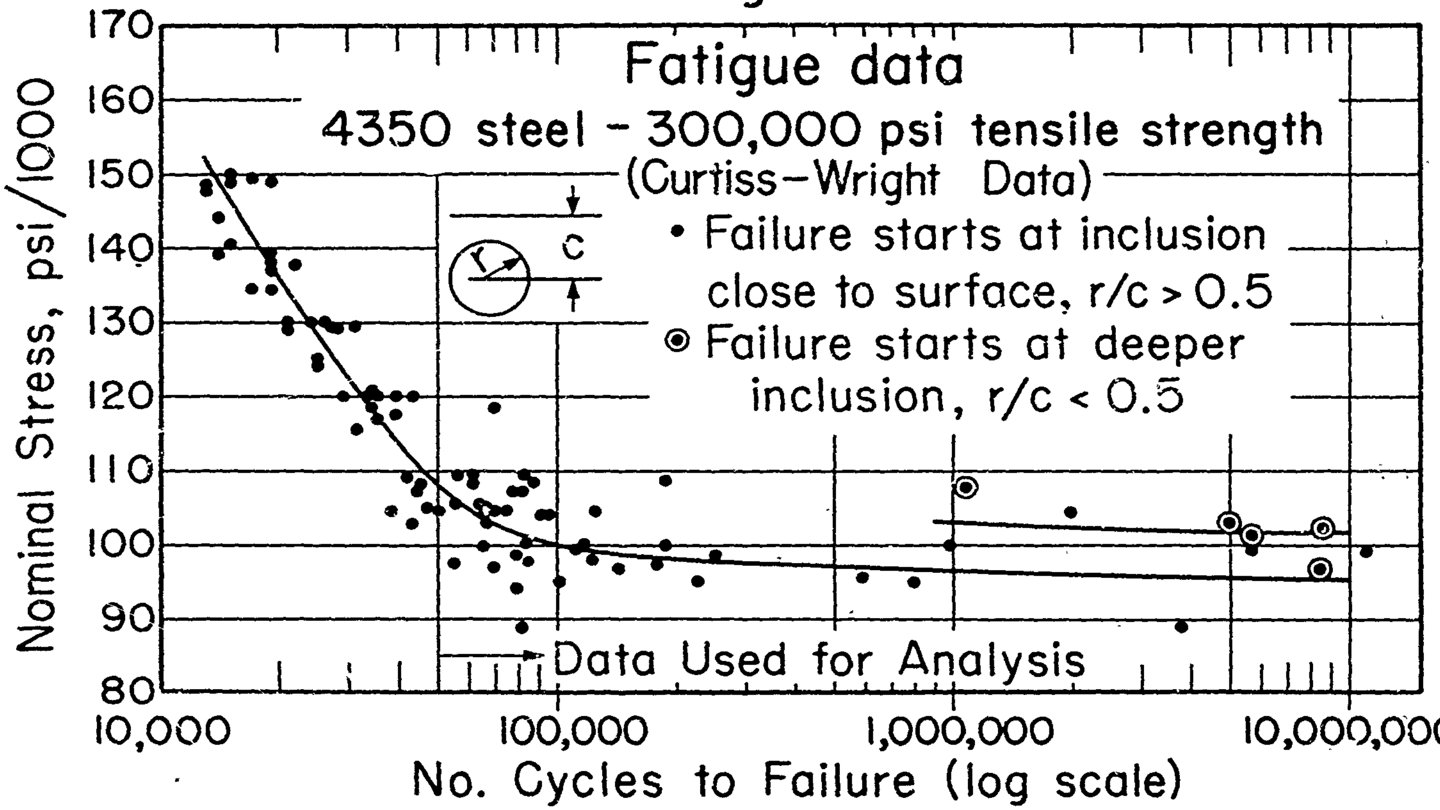
In Fig. 1b let us represent the inherent (unnotched) fatigue strength by S_f. Let us now imagine that we have applied just enough load to bring the peak stress S_{max} up to the inherent fatigue strength of the material S_f. Since we are dealing with a material which has a grain structure, it would be unrealistic to expect that fatigue failure would occur under conditions where the maximum stress is applied only at a "point" (or line if we consider thickness). It would seem that it should be necessary to apply more load so that S_f will be reached at some depth δ as shown in Fig. 1c. Or viewed in a related way, one might expect that failure occurs when the average stress over a distance ϵ reaches S_f (see Fig. 1d).

To put this idea to work we start by making the simplest possible assumption, namely that δ, or ϵ, is a constant[17,19] for a particular material,* and then try this out to see how the relations fit the data. The two above assumptions (Fig. 1c, 1d) differ mainly near the bottom part of the stress peak, but it turns out that for practical purposes the difference is unimportant. Since it is easier to use the δ method of Fig. 1c, this forms the basis of the curves of Figs. 2, 3, 4 and 6. Assuming a constant δ, the shape of these curves are not arbitrary, but are determined by the actual stress distribution. The positions of the curves are of course such as to fit the data.

Considerable information exists on stress distributions for infinitely wide members,[17, 19b, 25] but for finite members very little

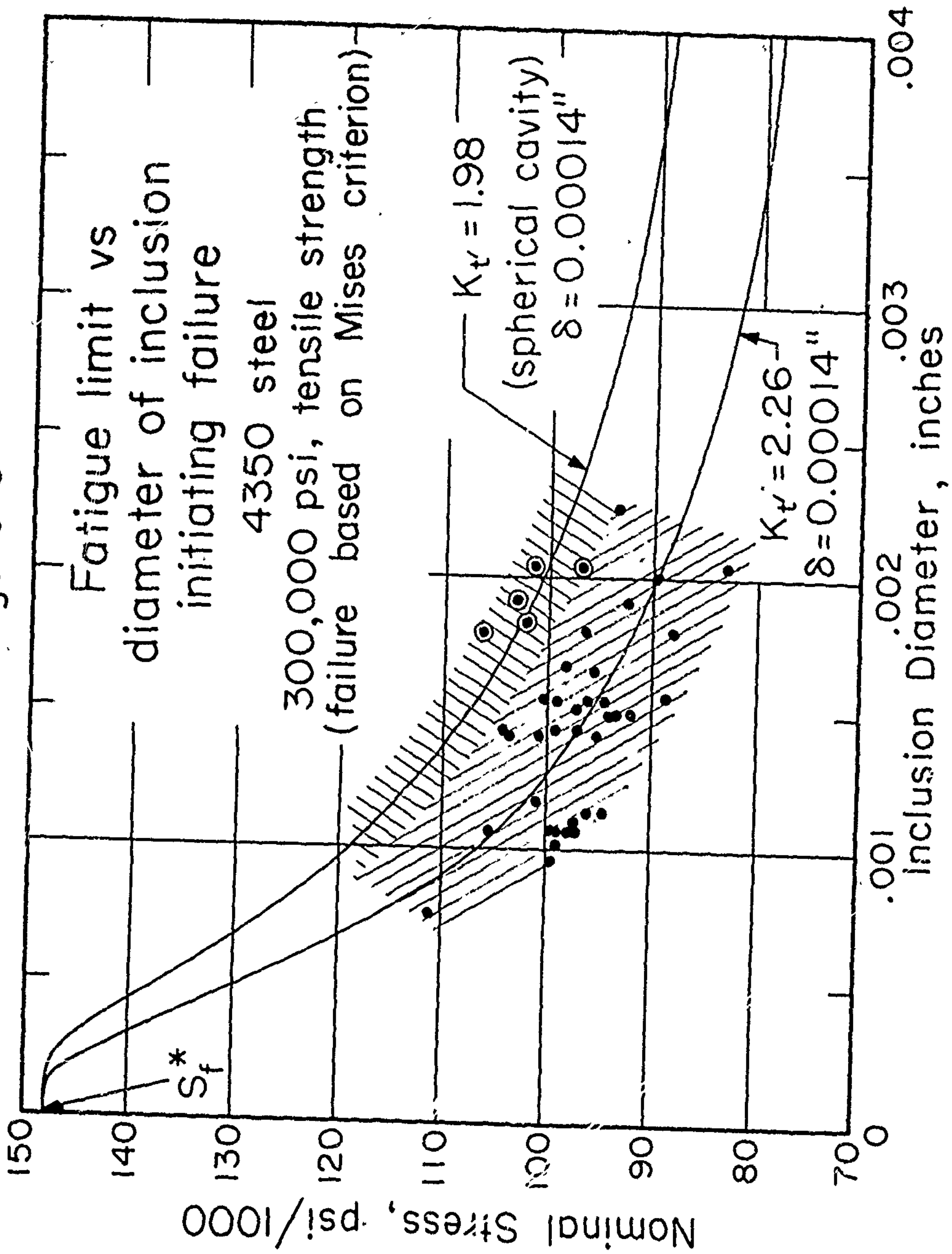
*Any heat treatment or other processing resulting in different physical properties is regarded as creating another material in the sense used herein.

Figure 5



Fatigue data
4350 steel - 300,000 psi tensile strength
(Curtiss-Wright Data)
• Failure starts at inclusion close to surface, r/c > 0.5
◉ Failure starts at deeper inclusion, r/c < 0.5
r
c
Data Used for Analysis
170
160
150
140
130
120
110
100
90
80
Nominal Stress, psi/1000
10,000
100,000
1,000,000
10,000,000
No. Cycles to Failure (log scale)

Figure 6



Fatigue limit vs diameter of inclusion initiating failure
4350 steel
300,000 psi, tensile strength
(failure based on Mises criterion)

information is available. The case of a tension strip with a hole has been worked out by Howland.[20] From Neuber's work, approximate distributions can be worked out for finite members. These results can also be expressed as maximum shear stress[3] curves and as octahedral shear stress[3] curves. Such curves have been developed for the test specimen and loading conditions referred to under "Fatigue Tests," but it is not possible to give the details in this paper.

It will be noted from Fig. 2 for 24 S-T sheets that $\delta = 0.014$ provides curves having reasonably good fit. This is based on maximum shear stress; for octahedral stress (not shown) $\delta = 0.008$ results in best fit of data. For 75 S-T sheets, the corresponding values are $\delta = 0.005$ and 0.004. For round specimens (Fig. 4) the q values are somewhat higher than for the sheets, although the data are insufficient for firm quantitative conclusions. In general it can be seen that the fatigue results are characterized by considerable scatter and, therefore, it would be difficult to make meaningful comparisons on the basis of individual S-N curves. By looking at an assemblage of results, it is believed that certain trends can be distinguished.

An important point should be mentioned at this time. Note that for all tests of a strip having a hole of diameter 1/4 of the strip width (Figs. 2 and 3), the theoretical stress concentration factor is the same, 2.43. However, the actual effect depends on the absolute <u>hole diameter</u> (or radius as shown). This means that one cannot determine a unique notch sensitivity value by making tests with specimens of only one given set of dimensions. Also the result may be unsafe when applied to a larger member. Furthermore, comparisons of different materials can be misleading.

The high strength steels mentioned in the previous section contained small inclusions of round cross-section and the dimensions and location of the nucleating inclusion had been carefully measured on the fracture surface. This information* provided the possibility of an analysis of the foregoing type, on the assumption that the inclusion acts primarily as a cavity. On Figs. 5 and 6 the top curve is for a deep seated spherical cavity ($K_t' = 1.98$); the lower curve is for the remaining cavities close to the surface, in which case K_t' turns out to be 2.26 for the same value of $\delta = 0.00014$. Kuhn and Hardrath[18] suggested plotting a material parameter in terms of tensile strength, and this has been done for δ in Fig. 7.

The third objective mentioned at the outset is to seek a simple design formula and appropriate material constants. In Figs. 8 to 11 the formula $q = 1/(1 + a/r)$ has been used,[21] with the values of "a" as listed in Table 1.

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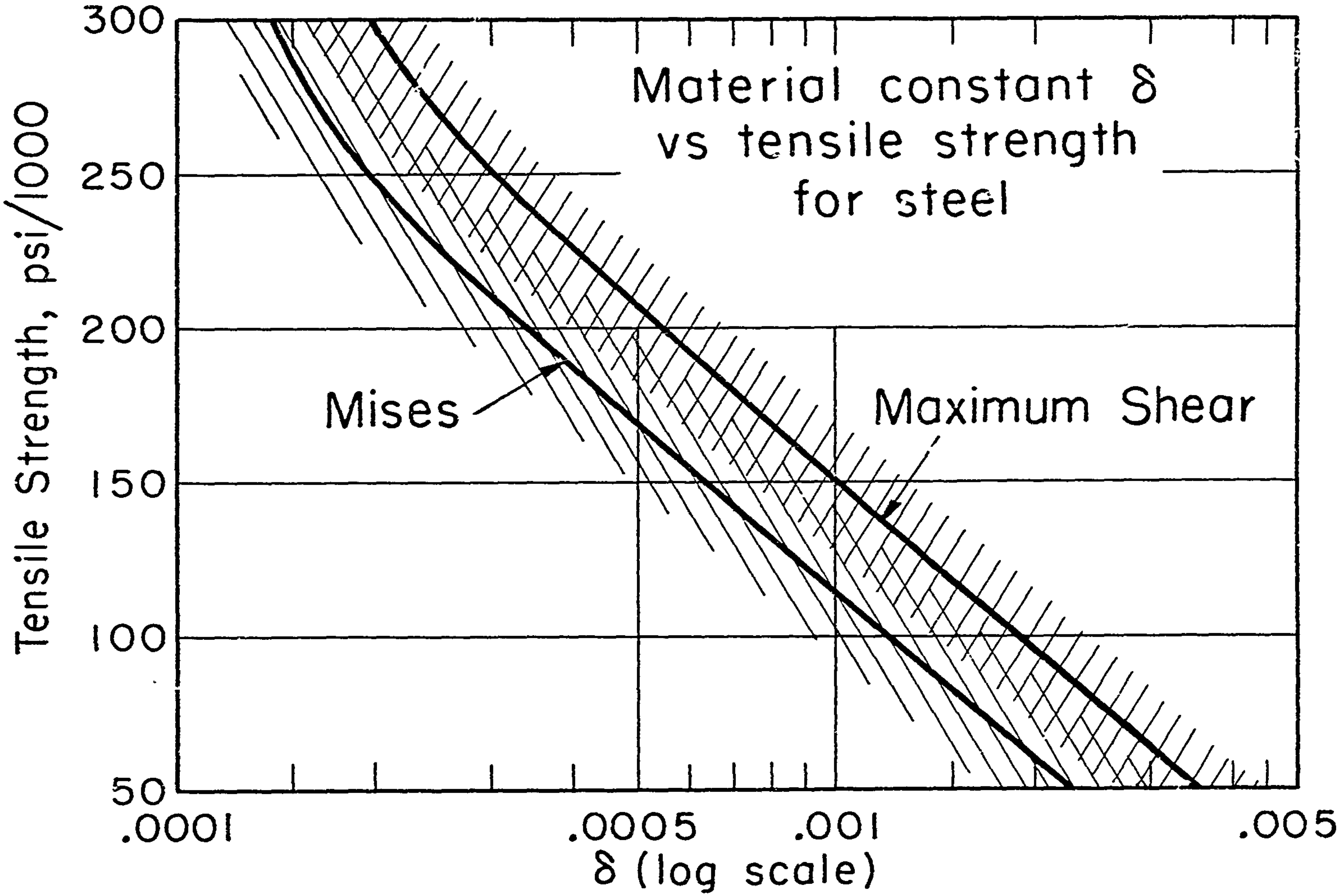
*The writer is grateful to Mr. F. E. Stulen of Curtiss-Wright for his cooperation in supplying supplementary information.

Figure 7


Material constant δ
vs tensile strength
for steel
Tensile Strength, psi/1000
300
250
200
150
100
50
.0001
.0005
.001
.005
δ (log scale)
Mises
Maximum Shear

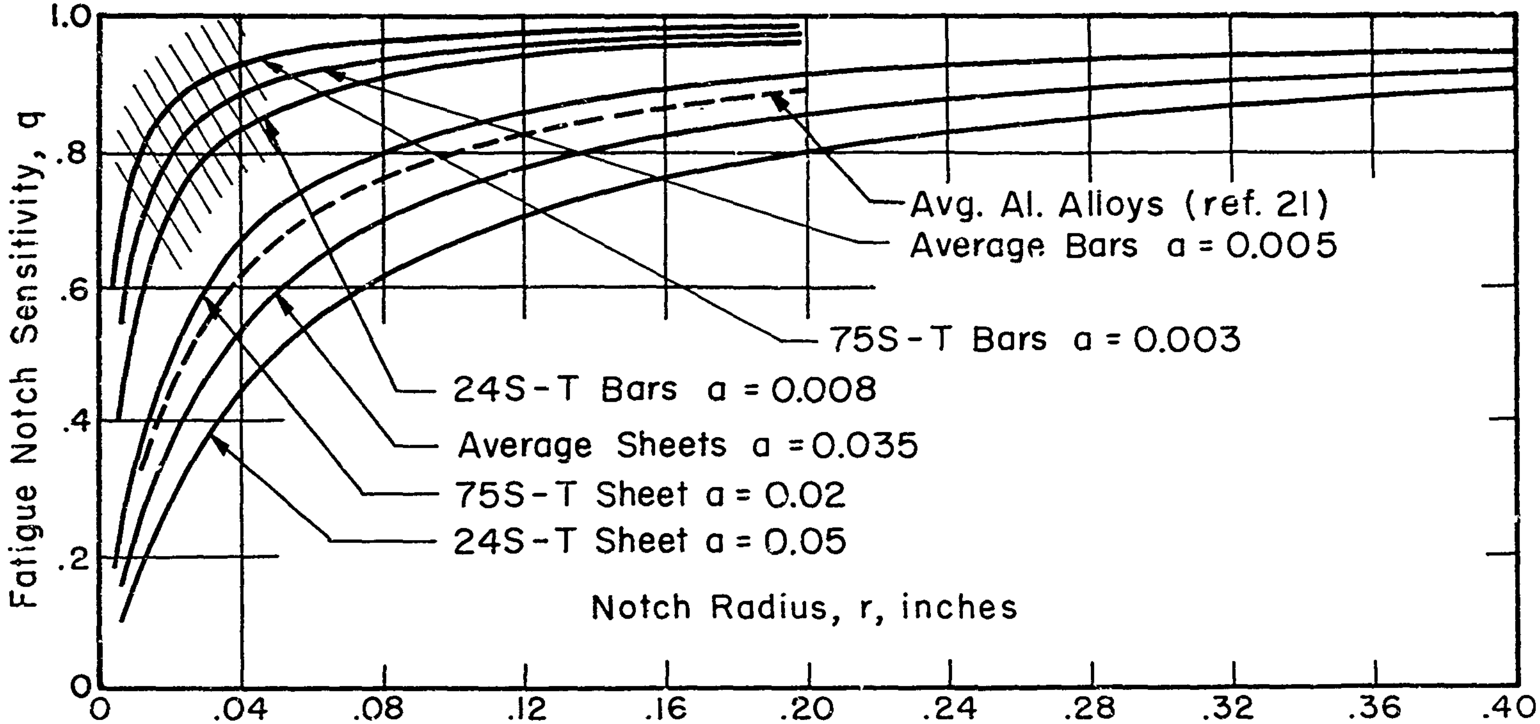


Figure 8 — Average Values of "a"
Aluminum alloys 24S-T and 75S-T

$$q = \frac{1}{1 + a/r}$$

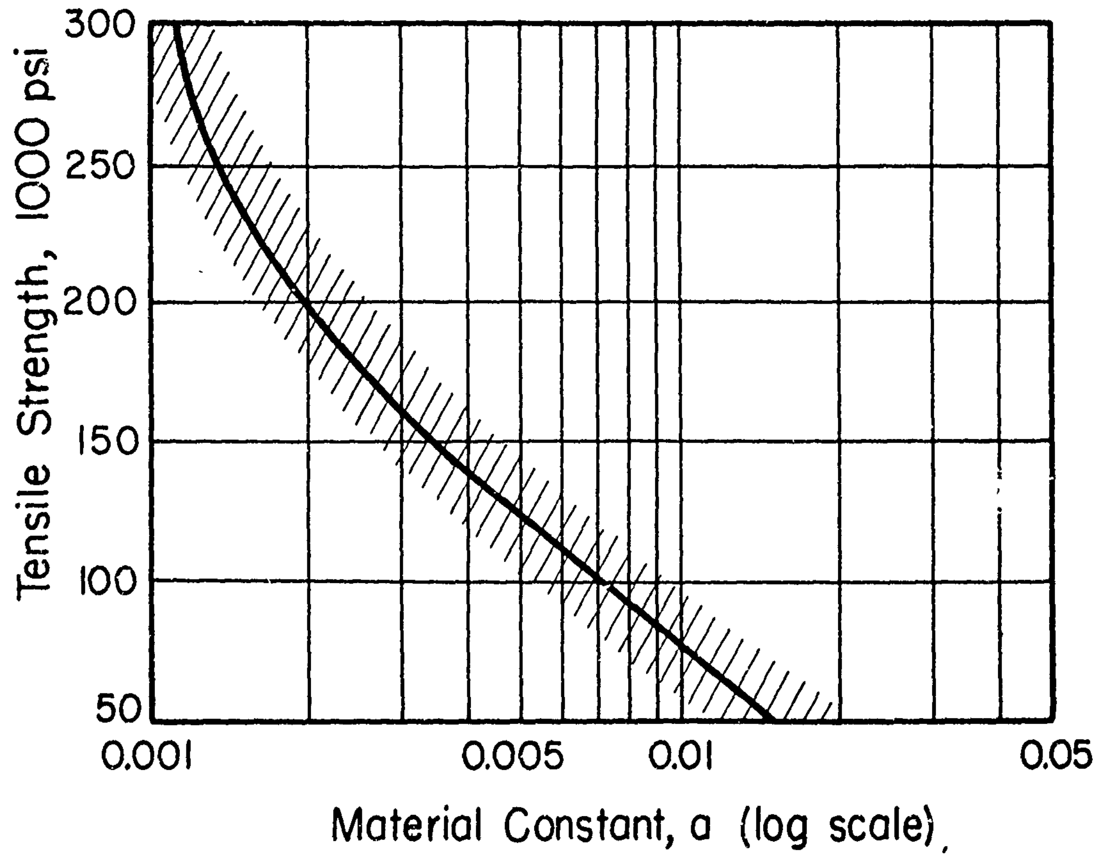


Figure 9-Values of "a" for steels of various tensile strengths.

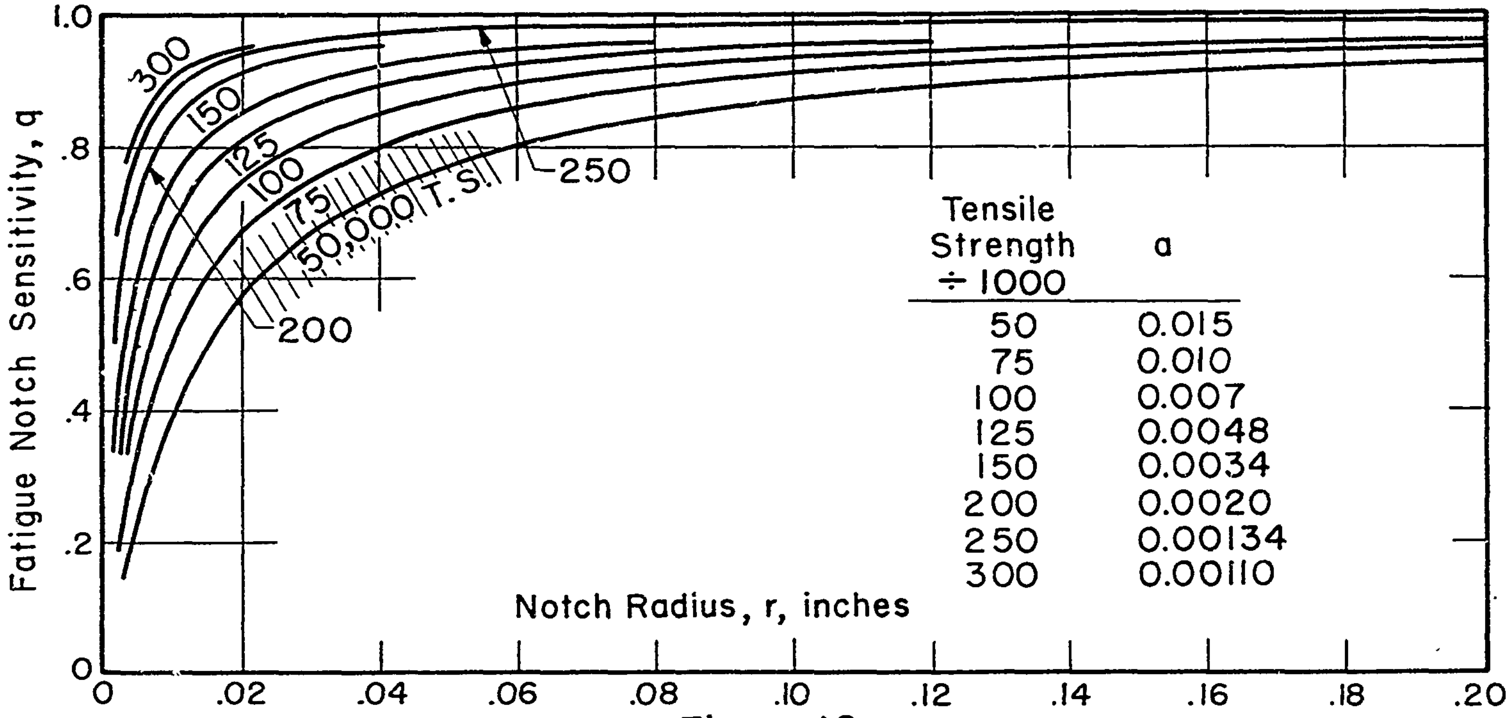


Figure 10

Average fatigue notch sensitivity curves for steel.

$$q = \frac{1}{1 + a/r}$$

The general averages for steels and for aluminum alloys (Fig. 11) are intended only to give a broad comparison of widely different materials. The general averages are not intended for design, since the more specific values are given in Table 1.

DISCUSSION

From Fig. 12 it can be seen that only for very high strength steel could one expect an appreciable gain in fatigue strength by eliminating all inclusions.*

It is not certain that the assumption that an inclusion can be treated the same as an empty cavity from a fatigue standpoint is of sufficient accuracy to be used in developing a general δ vs. r curve. Further tests should be made with grooved specimens where the notch radius is of the order of 1 to 5 mils, with the groove depth about 1/8 of the specimen diameter. Incidentally, the same kind of tests are needed for round aluminum alloy specimens, particularly for 75 S-T.

From the "q vs. r" curves it is not possible to compare strengths of alternative designs directly, since the basic fatigue strengths also are involved. A direct comparison of this kind is shown in Fig. 13 for 24 S-T and 75 S-T sheet specimens with a hole 1/4 of the width.** Note that for no hole or a very large hole 75 S-T is possibly slightly stronger in fatigue. For a hole diameter of about 1/2" the two materials are about the same, whereas for smaller holes, 24 S-T is stronger in fatigue. The maximum difference is at about 0.04" diameter where 24 S-T is about 1/3 stronger than 75 S-T in fatigue on the average. Rivet and bolt holes are larger in diameter, 1/8" or greater, and here the slight difference in favor of 24 S-T is hardly significant.

The main interest in connection with the hole tests, as far as aircraft design is concerned, is their role in eventually developing an analytical procedure for design of riveted or bolted joints. It is not obvious that the ratio of fatigue strengths of riveted joints of 24 S-T and 75 S-T should be the same as the ratio of fatigue strengths of strips with a hole for the same two materials. For example, differences in fretting corrosion could conceivably affect the relative joint strengths. Comparative tests of various riveted joints of 24 S-T and 75 S-T, bare and Alclad, are shown in Fig. 14

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* Since completing the paper a report[26] has been received of work at Armour Institute on processes aimed at varying or reducing inclusions. The results seem to be in rather good agreement with the above comment made with regard to Fig. 12.

**The smooth curves do not correspond to those of Fig. 3 where both holes and notches were considered.

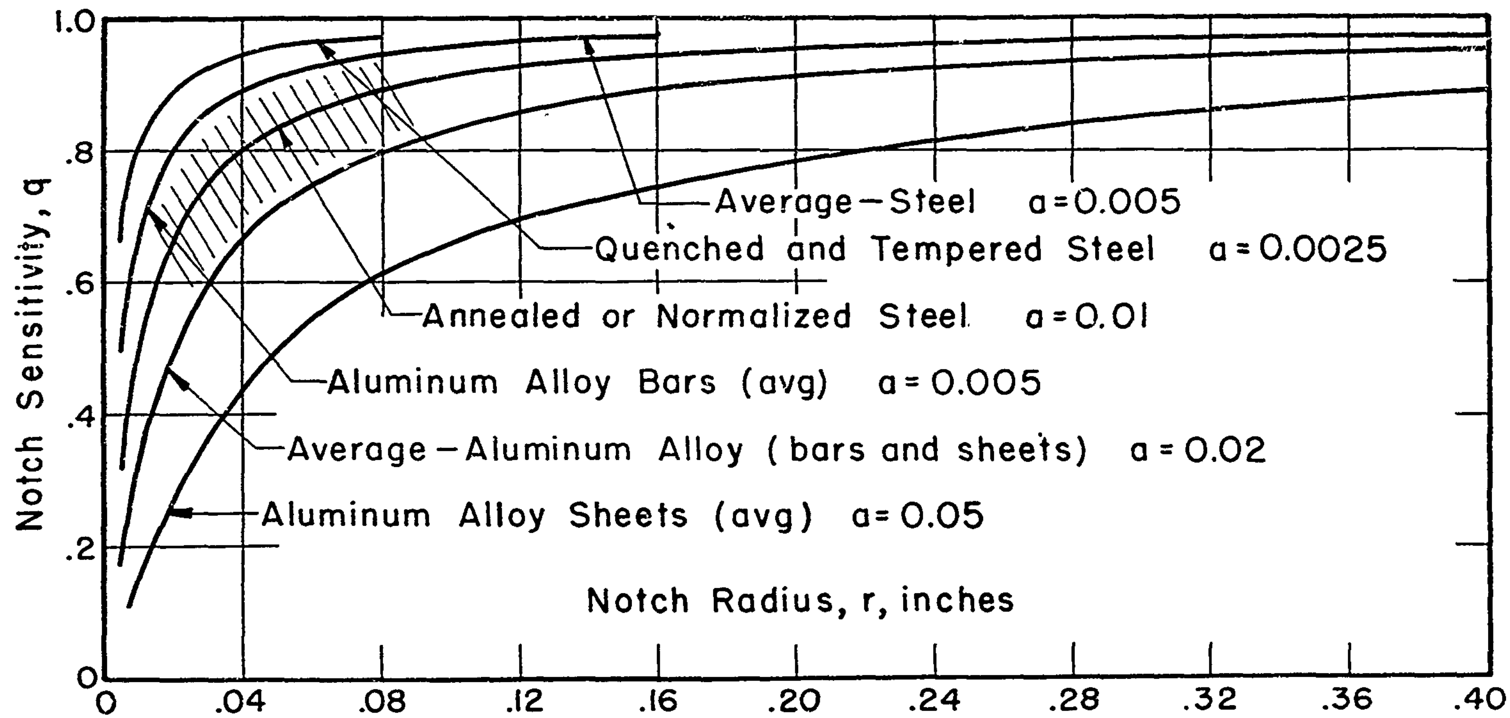

Figure 11 - Average fatigue notch sensitivity

$$q = \frac{1}{1 + a/r}$$

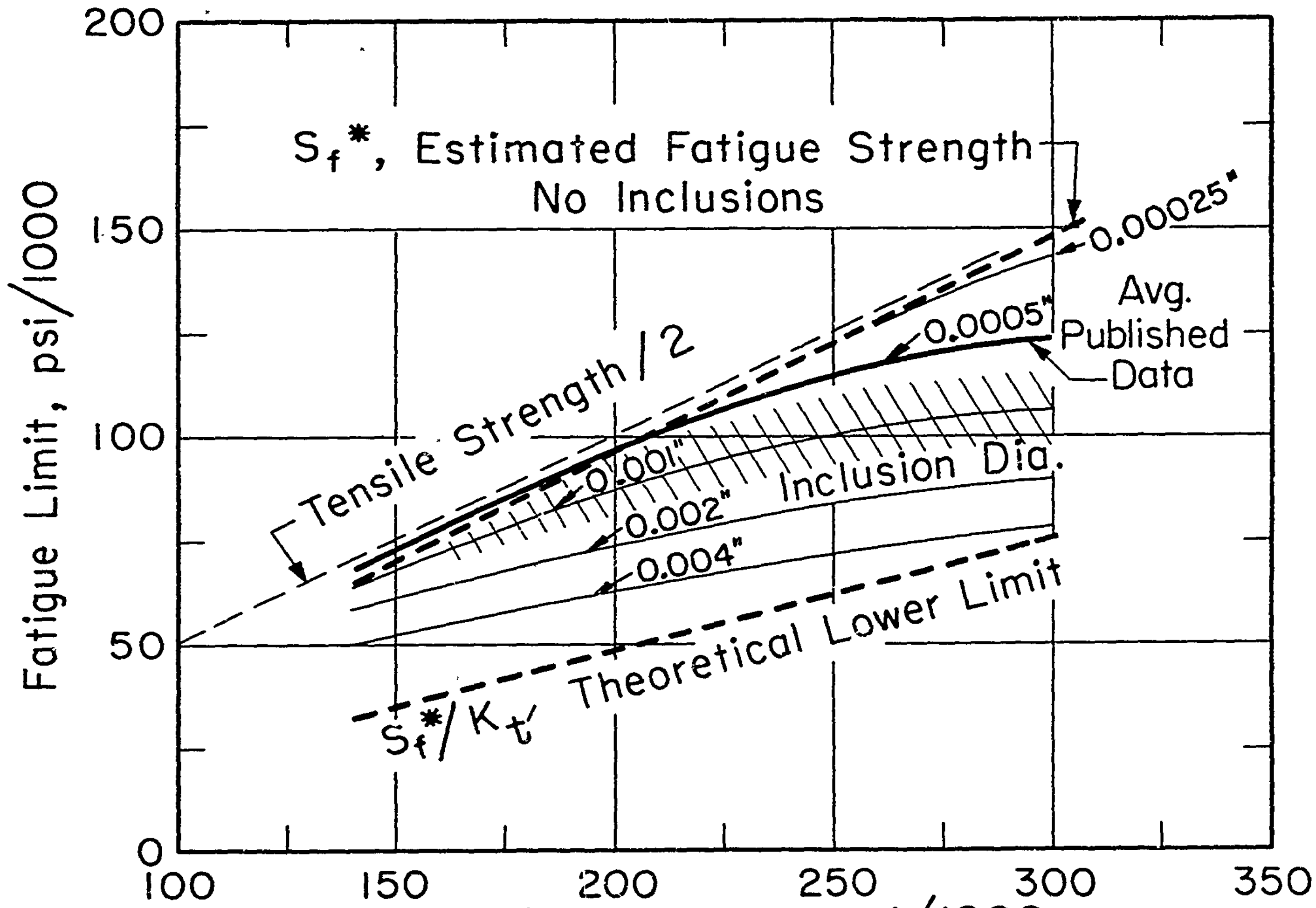

Fig. 12-Fatigue strength vs inclusion diameter
4340-4350 steel

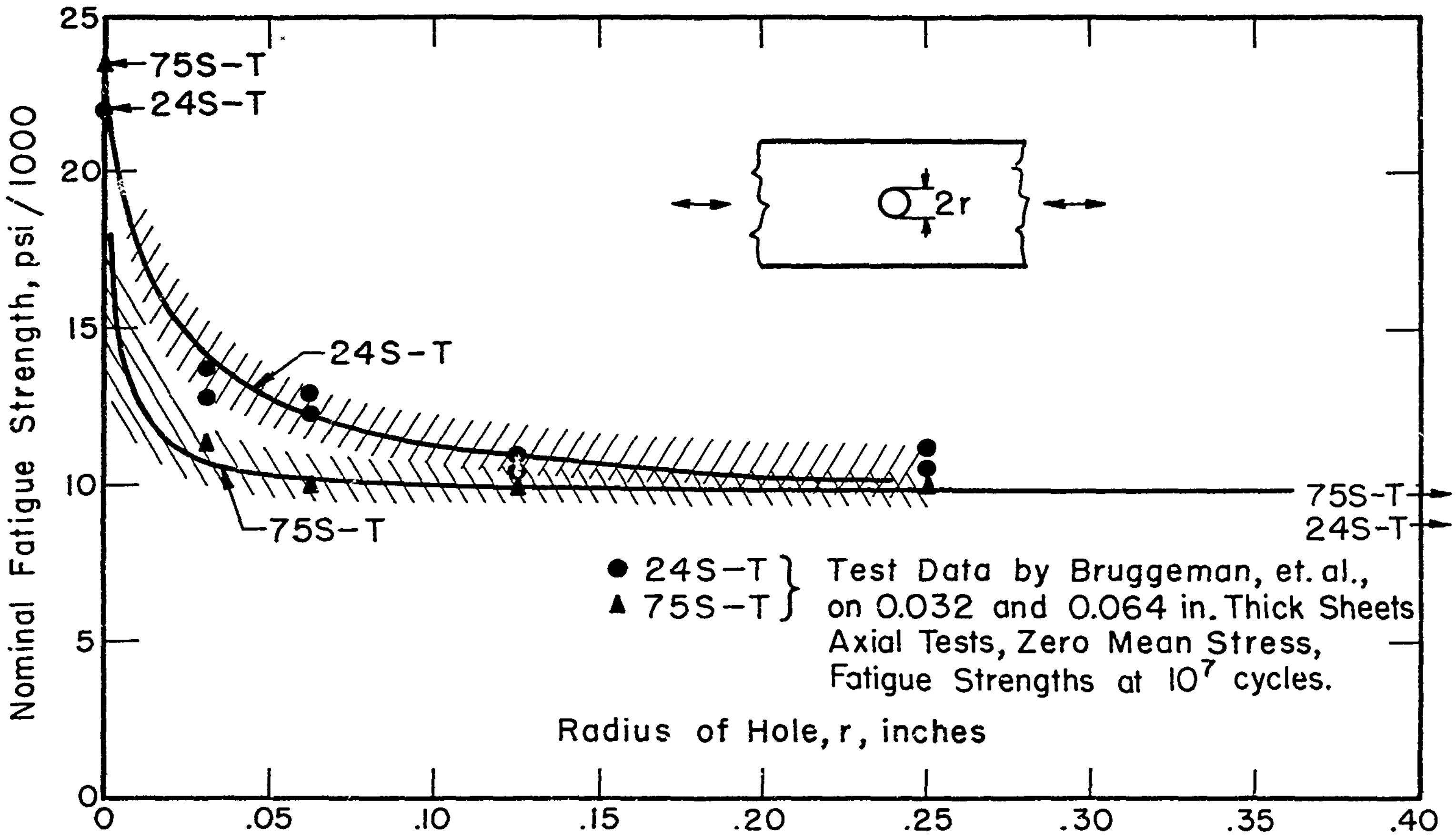

Figure 13— Fatigue strength of aluminum alloy sheet with hole diameter 1/4 of width.

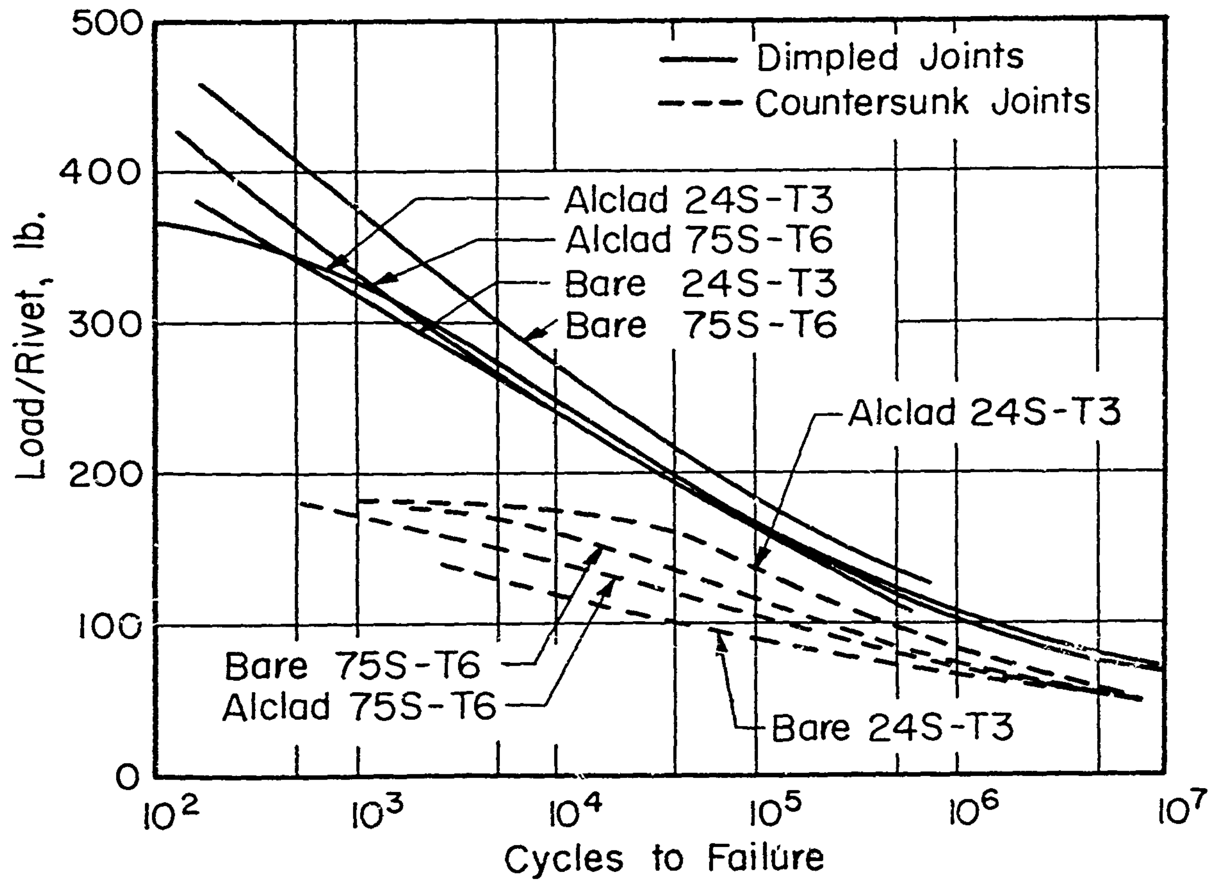

Fig.14-Comparison of lap joints of 24S-T3 and 75S-T6 strips 0.032" thick.
(Howard and Smith)

for alternating stress[22] and in Fig. 15 for pulsating stress.[23] These show no significant difference between the two materials at 10^7 cycles. Pulsating fatigue tests of bolted joints[24] showed no consistent difference between the extruded aluminum alloys tested. Referring to Fig. 13, based on sheet specimens and zero mean stress, we see that at 0.1285" diameter (0.0642" radius) the small difference of averages in favor of 24 S-T is of doubtful significance, so that we can regard the two materials as giving about the same result for this hole size. Since the jointed specimens also show no significant difference (at 10^7 cycles), one therefore reaches the conclusion that any additional factors introduced by the riveting or bolting together of the drilled members, such as fretting corrosion, do not introduce a difference of any considerable magnitude, at least for this set of test parameters.

The foregoing comments are based on 10^7 cycles of constant amplitude. Aircraft service conditions vary considerably. During the useful life of an airplane, some parts are subjected to a large number of stress cycles, some parts to a small number of stress cycles. The ratio of steady to alternating components of stress differs. Instead of a constant amplitude, spectrum loading is typical. In taking account of these factors, conditions may favor the use of 24 S-T in certain applications and 75 S-T in others. This is ably discussed in reference 23.

From Fig. 8 it will be noted that the sheet specimens are less notch sensitive than the round specimens. This may be due mainly to the smaller volume of material exposed to peak stress (or, say, within 1 percent of peak stress) in the sheet specimens as compared to the rounds. No attempt has been made, in the present analysis, to correlate the results on a volume basis since this would be very difficult to apply to design. The volume criterion, however, should be pursued as a matter of scientific interest. Another difference is the plane stress vs. plane strain condition. This factor, however, is more susceptible to analysis.

The top curves for bar specimens of Fig. 8 may be considered as applicable to aluminum alloy elements other than sheets, - "three dimensional" elements such as spars, fittings, etc. We note that while there is not a large difference between 24 S-T and 75 S-T, the maximum difference occurs for quite small fillet radii, in the range 1/64" to 1/32". It hardly needs to be said, at this date, that such small radii (with their high K_t values) should not be used in important joints or highly stressed regions.

Pursuant to the remarks under "Definitions" one should not conclude from the shape of the "q vs. r" curves of Fig. 4 that for constant notch depth and decreasing r that a condition of no notch effect is approached. This has been discussed in detail[27] and to

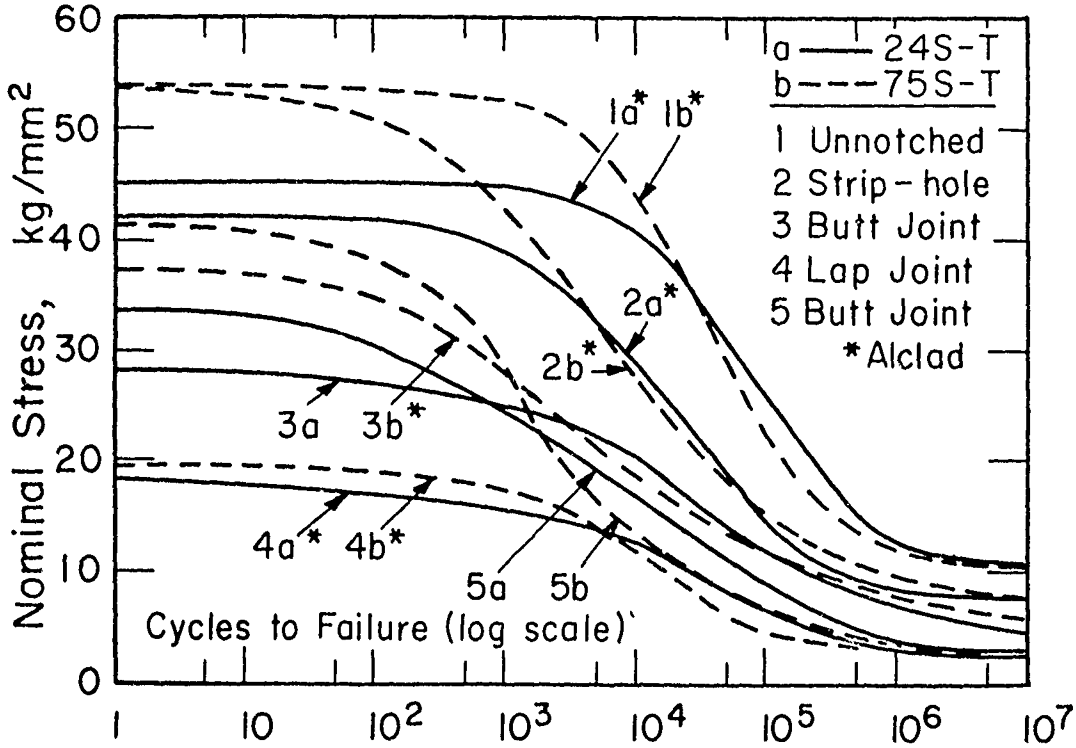

Fig.15-Comparison of 24S-T and 75S-T under conditions of pulsating stress (0 to max) (Lundberg and Wållgren)

try to prevent the above conclusion the writer has either not shown the q vs. r curve as going to zero, or has stated that the curve must not be used for ratios of notch depth to notch radius greater than 3 or 4. Actually, for constant notch depth the relation should be regarded as approaching zero as a limit. The reason for this is as follows.

The stress distribution curve for zero radius ($K_t = \infty$) is not much different from that of a deep notch in the region where δ generally falls, so that the type of analysis used herein indicates that a finite fatigue notch factor should be approached with decreasing notch radius. This is indeed what actually does happen, with values of the order of 2 to 4 being reached for notches with radii machined as sharp as possible. The actual value depends on the size of piece, depth of notch and material. So we have a situation where with decreasing notch radius, the fatigue notch factor approaches a finite value while the theoretical factor increases toward infinity, resulting in lower and lower notch sensitivity values, approaching zero.

In this connection Templin's tests are of interest. He machined as sharp a vee notch as he could in R. R. Moore specimens and for aluminum alloys he obtained a fatigue notch factor of about 2. Later he measured the notch radii and found an average of about 0.0002". Having this, one can compute the stress concentration factor, about 16, and also the notch sensitivity, less than 0.1, as shown in Fig. 4. Since the ratio of notch depth to notch radius is about 375 in Templin's tests, it would seem that the q vs. r relation need not be limited to ratios of 3 or 4.

From Table 1, it is interesting to note that the average notch sensitivity for steels comes out to be the same as for aluminum alloys tested as round specimens (bars). The steels were also tested as round specimens; in other words, if the aluminum alloy sheet tests are considered separately, then the average notch sensitivity for steels and aluminum alloys turns out to be about the same. As mentioned previously, these are averages of very wide scatter bands. The gross averages of Fig. 11 should not be used in design, since the more specific values are also listed in Table 1. No generalizations have been attempted for materials other than those given in Table 1 but references to some tests on a few other materials such as magnesium alloys, titanium, etc., have been listed.[15]

SUMMARY

It seems possible to obtain a better understanding of stress concentration effects by pursuing the concept of notch sensitivity.

The analysis also shows that it is not possible to completely

TABLE 1

AVERAGE VALUES OF "a"

24 S-T Sheet	0.05
75 S-T Sheet	0.02
Avg. Aluminum Alloy Sheet	0.035
24 S-T bar	0.008
75 S-T bar	0.003
Avg. Aluminum Alloy bar	0.005
Avg. Aluminum Alloy bar and sheet	0.02
Annealed or Normalized Steel	0.01
Quenched and Tempered Steel	0.0025
Avg. Steel	0.005

describe notch behavior in a material for a given shape of notch (given stress concentration factor) by testing only one size of specimen. If simple comparisons are to be made the analysis indicates the optimum value of notch radius to use for the comparison.

Estimates of the effect of minor defects such as scratches, tool marks, nicks, etc., turn out to be realistic and it is believed this area is now better understood.

If these concepts can be generalized then it should be possible to estimate the relative effect of certain changes in detail design.

The analysis presented herein is not regarded by the writer to have yielded results which are in "final" form in anything like the sense for example that we regard certain K_t factors. We need much more test data in regions where test specimens are difficult to make and measure. Then there is the problem of scatter, making it desirable to utilize large numbers of specimens and statistical treatment of results. The situation can be improved by a frontal attack of planned type as proposed by the opening speakers of the symposium. It is essential that not only a large amount of data become available but that the tests be made in critical regions where appreciable effects can be expected. Accumulation of such information will make it possible to rationalize this subject with a greater degree of reliability. It is hoped that in the interim the paper will be helpful as an approximate guide to engineers and designers in considering the effects of stress concentration in fatigue.

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THERMAL FATIGUE ANALYSIS

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Fatigue analysis at room temperature is based upon some method of comparison of S-N curves with gust and maneuver load spectrums, where the S-N curves depend on the mean stress level, the stress concentration factor, and the mechanical properties of the material. Elevated temperatures can affect most of these basic factors in the room temperature problem by changing the mechanical properties, by possibly changing the stress concentration factor, and by changing the mean stress level so that completely new S-N curves must be obtained. In certain cases of cyclic temperatures, a thermal load spectrum may be added to the gust and maneuver spectrums. Temperature may also introduce a new mode of failure by producing large strains in a finite life (creep and cyclic strain accumulation). The problems associated with the evaluation of these temperature effects are demonstrated by attempting to obtain results for a simple idealized skin-stringer structural element.

INTRODUCTION

The addition of temperature to the room temperature structural fatigue problem not only affects the stress cycle (S-N) curve of the material for a given mean stress and stress concentration factor by changing the material properties but also may affect the mean stress and the stress concentration factor themselves. Also a new mode of failure may arise -- a large strain failure in a finite life from

either creep or strain accumulation. To identify these various effects in the life problem of the structure, the following Table 1 has been prepared. In Table 1, various combinations of the five items of applied mean stress, applied alternating stress, thermal stress, creep, and strain accumulation have been listed for various temperature conditions. It should be noted that the problems of variation in material properties and of the stress concentration factor arise in all the combinations listed in Table 1.

Problem No. 1 in Table 1 is the conventional room temperature fatigue problem which is far from solved. Since creep at elevated temperatures is a function of the steady load or applied mean stress while fatigue is primarily a function of alternating stress, there must be some kind of interaction between creep and fatigue when both an applied mean stress and an applied alternating stress are acting.

Problem 2 in Table 1 defines a fatigue reference stress F_{aR} for no applied mean load while Problem 3 in Table 1 defines a creep reference stress F_{mR} for no applied alternating stress. Both reference stresses are at a uniform steady state temperature and will depend on the stress concentration factors for fatigue and for creep and on the material properties at the temperature. The reference creep stress may be the stress for a certain per cent strain rather than a creep rupture stress. Figure 1 shows the S-N curves for F_{aR} for two temperatures for 2024-T4 aluminum alloy (Ref. 2 and 4). It is possible to approximate F_{mR} for various temperatures and times by using a creep parameter, such as the Larson-Miller parameter. Figure 2 shows creep rupture and 1% creep strain stresses against the Larson-Miller parameter (Ref. 5) for 2024-T3 aluminum alloy.

CREEP-FATIGUE

Problem 4 in Table 1 is the combined creep-fatigue problem at uniform elevated temperatures. In Ref. 2, Padlog and Schnitt have considered this problem in some detail. By using the reference stresses F_{aR} and F_{mR} in Problems 2 and 3 they propose an interaction curve between the mean stress F_m and the alternating stress F_a in the form

$$\left(\frac{F_a}{F_{aR}}\right)^p + \left(\frac{F_m}{F_{mR}}\right)^q = 1 \qquad (1)$$

where p and q depend on temperature, frequency, time to failure, and the material. At room temperature p and q are approximately one. Test data must be used to obtain p and q at elevated temperatures. Creep-fatigue tests (Ref. 4) have been performed using different stress ratios $r = F_a/F_m$, at various elevated temperatures. The S-N curves obtained for the various values of r can be cross-plotted for a given number of cycles or a given time to give a plot of alternating stress F_a against F_m in a stress range fatigue diagram. See Fig. 3 where each curve represents a unique failure life for a given temperature. These curves are for one testing frequency; they might differ for another frequency (see Ref. 6).

If Fig. 3 is made non-dimensional by using F_{aR} and F_{mR} then curves represented by Eq. (1) are obtained. Figure 4 shows Fig. 3 in non-dimensional form. Each curve in Fig. 4 represents the allowable strength for a constant temperature level, frequency, and number of load cycles or time either to fracture or to attain a prescribed total deformation. These curves are valid for a uniform sinusoidal loading

Table 1

(Stress concentration factors in each problem)
(Material properties vary due to temperature)

Prob. No.	Temperature	Applied mean stress	Applied alternating stress	Thermal stress	Creep	Strain accum.	Title	Applicable to aircraft components
1	Room	yes	yes	no	no	no	Fatigue	All
2	Uniform steady state	no	yes	no	no	no	Fatigue reference	
3	Uniform steady state	yes	no	no	yes	no	Creep reference	
4	Uniform steady state	yes	yes	no	yes	no	Creep-fatigue	Cruise
5	Uniform cyclic	yes	yes	no	yes	no	Creep-fatigue	Repeat missions
6	Uniform cyclic	no	no	yes (restrained)	yes	no	Thermal-stress fatigue	Equipment supports
7	Nonuniform steady state	no	no	yes	yes	no	Varying creep	Cooled regions
8	Nonuniform steady state	yes	yes	yes	yes	no	Varying creep fatigue	Cooled regions, mixed materials
9	Nonuniform cyclic	no	no	yes	yes	no	Thermal-stress fatigue	Supports for cooled equipment
10	Nonuniform cyclic	yes	no	yes	yes	yes	Strain growth	Pressure vessels
11	Nonuniform cyclic	yes	yes	yes	yes	yes	All	All

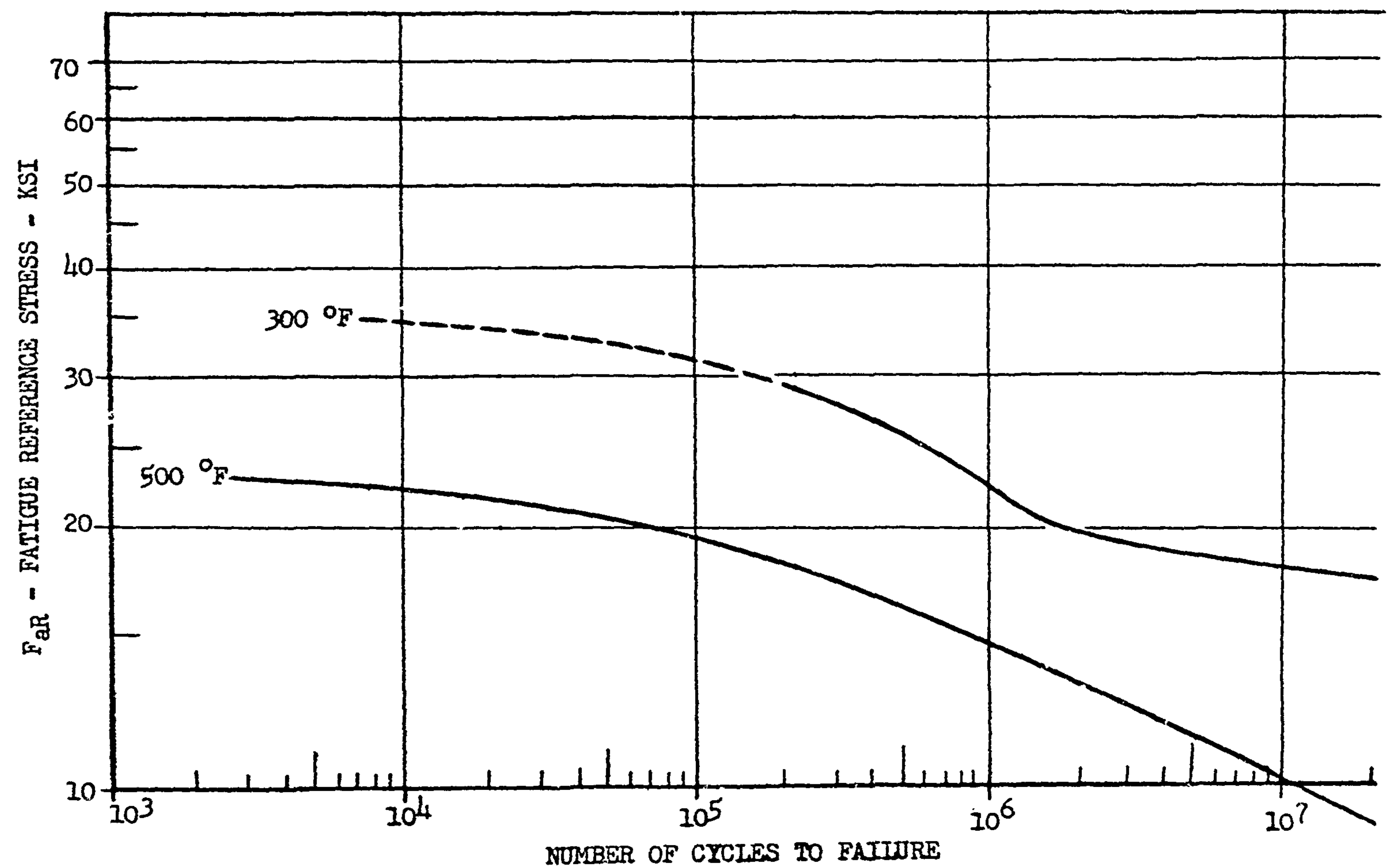


FIGURE 1. F-N FATIGUE DIAGRAM AT STRESS RATIO ($r = \infty$) FOR 2024-T4 ROLLED ALUMINUM ALLOY AT 300°F AND 500°F

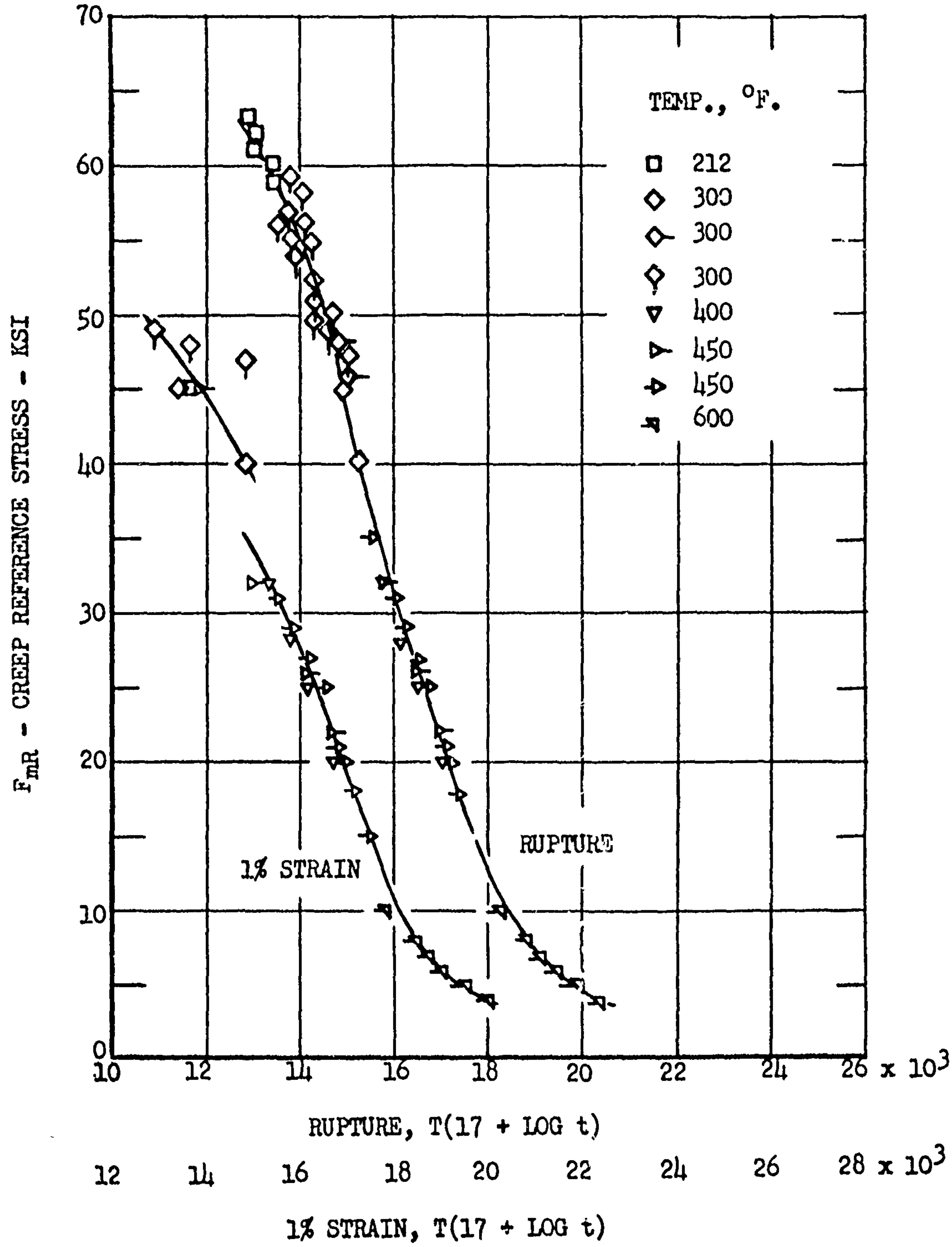


FIGURE 2. MASTER RUPTURE AND CREEP CURVES FOR 24S-T3 ALUMINUM ALLOY

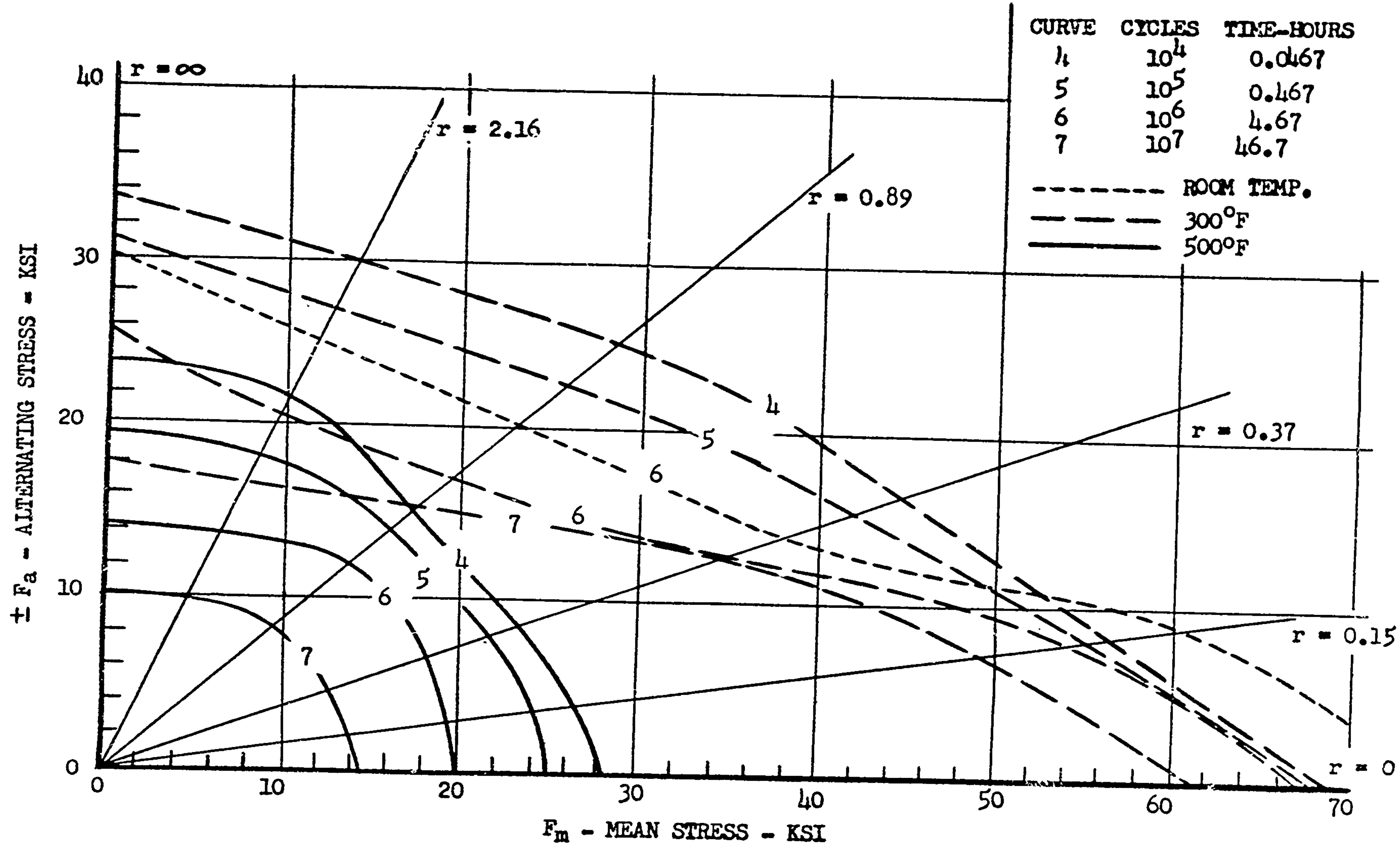
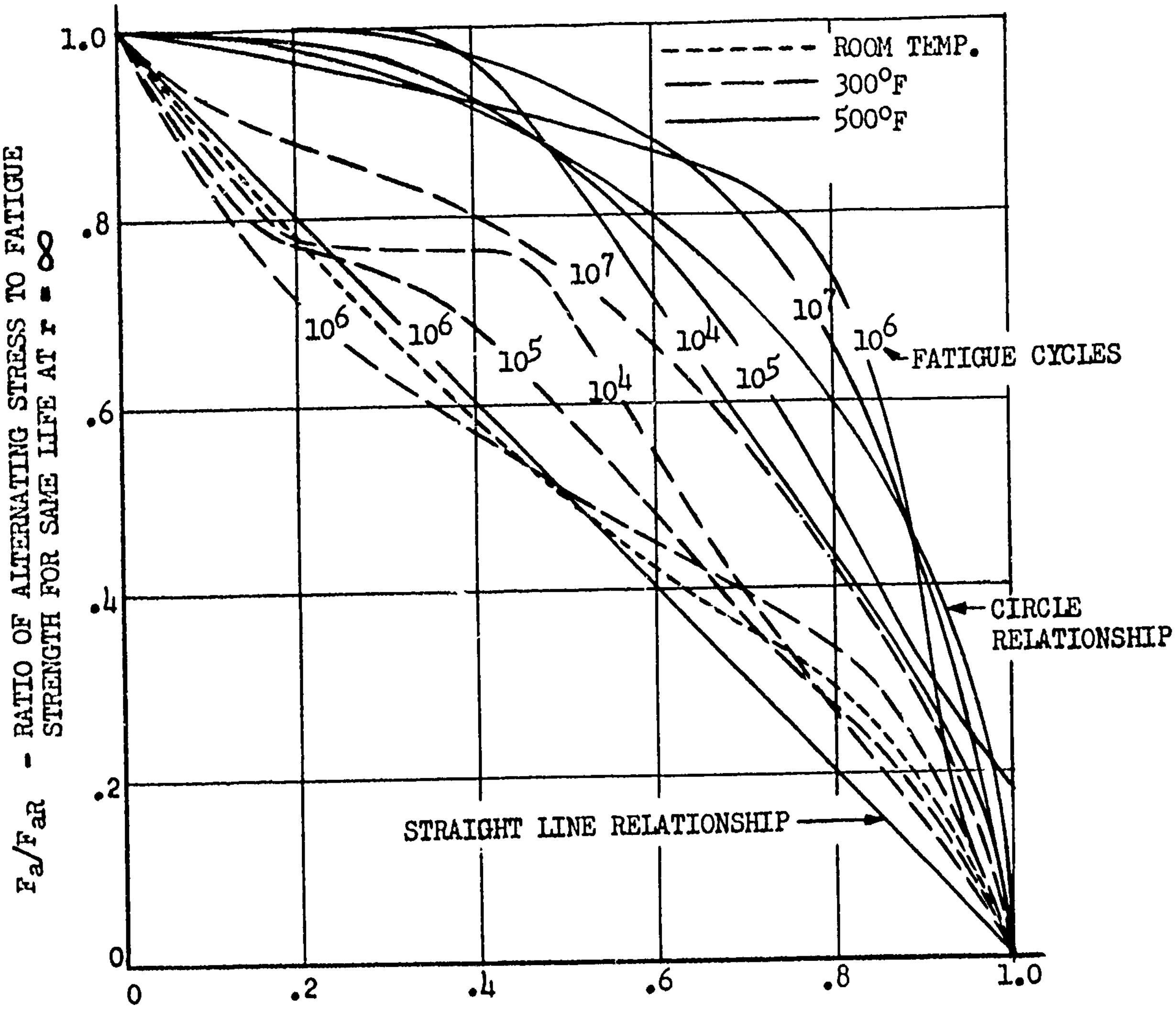


FIGURE 3. STRESS RANGE FATIGUE DIAGRAM FOR 2024-T4 ROLLED ALUMINUM ALLOY AT 300°F AND 500°F



F_m/F_{mR} RATIO OF MEAN STRESS TO RUPTURE STRESS FOR SAME LIFE AT r = 0

FIGURE 4. STRESS RANGE BANDS FOR 2024S-T4 ROLLED ALUMINUM ALLOY AT 300°F AND 500°F USING DIMENSIONLESS RATIOS

cycle. The values of F_{aR} and F_{mR} correspond to the same life and temperature. Because of the complex behavior of the fracture response of the material to combined creep and fatigue conditions, the nondimensional interaction curve in Fig. 4 and the curves for F_{aR} and F_{mR} in Figs. 1 and 2 represent a basic property of the material. As this property cannot be derived from simple material properties, it must be obtained from tests. Necessarily, many tests are needed to cover the temperature and stress ratio ranges.

In order to allow for various values of the F_a and F_m stresses it is necessary to use a cumulative damage procedure to find the life of the structure for creep-fatigue. In Ref. 2, the life fraction rule is proposed for creep in which

$$\text{Life} = \frac{L_o}{H}, \quad \text{if} \sum_{i=1}^{n} \frac{t_i}{t_{Ri}} = H \tag{2}$$

where $L_o = \sum_{i=1}^{r} t_i$, t_i = time spent at stress F_{mi} and temperature T_i and t_{Ri} = total time to creep-rupture or to attain a specified creep deformation at stress F_{mi} and temperature T_i. Miner's cumulative damage rule for fatigue is

$$\text{Life} = \frac{L_o}{D}, \quad \text{if} \sum_{i=1}^{n} \frac{N_i}{N_{Ri}} = D \tag{3}$$

where N_i = number of cycles of alternating stress F_{ai} at temperature T_i corresponding to the life L_o and N_{Ri} = number of cycles that causes fatigue failure at stress F_{ai} and temperature T_i.

In order to find the interaction between Eqs. (2) and (3) it is necessary to modify either t_{Ri} or N_{Ri} by means of Fig. 4 and Eq. (1) to account for the creep-fatigue life. If the shapes of the interaction curves in Fig. 4 are assumed to hold for various frequencies so that any combination of N cycles in t hours may be used, then Fig. 4 may be used to construct curves similar to Fig. 3 for a given problem. From those curves for given values of F_{ai}, F_{mi}, and T_i, the allowable number of cycles N_{Ri} or the allowable time T_{Ri} can be used for creep-fatigue failure. With these values either Eq. (2) or Eq. (3) can be used to calculate the life.

There are many assumptions in the above procedure for creep-fatigue life calculations that the material may not know about. The load spectrum is specified as various number of cycles of various amplitudes occurring in a random manner over a certain period of flying. Do all cycles last the same time regardless of amplitude or when they occur? Do 10^4 cycles at stresses F_{ai} and F_{mi}, and temperature T_i, isolated from each other with temperature and stress changes occurring in between, damage the material the same as if they occurred together? If a total of 10^7 cycles occur in 1000 hours of flying does this mean that the 10^4 cycles at F_{ai}, F_{mi}, and T_i last for an accumulated time of one hour? Suppose some of the mean stresses are compression, what happens? Can stress concentration factors be included without test data?

Problem 5, Table 1, is a creep fatigue problem similar to Problem 4 except the temperature is transient or cycles slowly. The calculation of the life must be based on accumulation of the temperature effects. This accumulation can be done by either Eq. (2) or Eq. (3) by obtaining T_{Ri} or N_{Ri} as described for Problem 4.

Example 1. Use Figures 1, 2, and 4 and Eqs. (2) and (3) in the above discussion to find the life of a tension element subjected to two stress and temperature levels.

Case		F_a	F_m	T	% Time
(a)	(1)	15,000 psi	15,000 psi	300°F	50
	(2)	6,250 psi	6,250 psi	500°F	50
(b)	(1)	10,000 psi	30,000 psi	300°F	50
	(2)	10,000 psi	30,000 psi	500°F	50

with equal times at each level. Assume the frequency to be 1000 cycles per hour.

From Fig. 4 assume the interaction curve for the 300°F case to be a straight line and for the 500°F case to be a circle so that from Eq. (1)

$$\frac{F_a}{F_{aR}} + \frac{F_m}{F_{mR}} = 1 \qquad (300°F)$$

$$\left(\frac{F_a}{F_{aR}}\right)^2 + \left(\frac{F_m}{F_{mR}}\right)^2 = 1 \qquad (500°F)$$

F_{aR} can be obtained from Fig. 1 for various assumed number of cycles while F_{mR} can be obtained from Fig. 2 for the corresponding time

$$t \text{ (hours)} \sim 10^3 t \text{ (cycles)}$$

Figure 5 shows the graph of the above equations for F_a against F_m for various assumed values of t. For the given pairs of values of F_a and F_m in cases (a) and (b), the time T_R can be read by interpolation from Fig. 5. Since $t_1 = t_2$, Eq. (2) gives

$$\text{Life} = \frac{L_o}{H} = \frac{t_1 + t_2}{\frac{t_1}{t_{R1}} + \frac{t_2}{t_{R2}}} = \frac{2t_1}{t_1\left(\frac{1}{t_{R1}} + \frac{1}{t_{R2}}\right)} = \frac{2t_{R1}\, t_{R2}}{t_{R1} + t_{R2}}$$

For case (a), t_{R1} = 1000 hours and t_{R2} = 1000 hours so that the life is 1000 hours.
For case (b), t_{R1} = 300 hours and t_{R2} = 0.4 hours so that the life is 0.8 hours.

It is evident that Fig. 5 can be used for other stress levels and the life obtained in Eq. (1) as long as the frequency is 1000 cycles per hour and the temperature is either 300°F or 500°F. All that is needed is the per cent of time at each stress and temperature level.

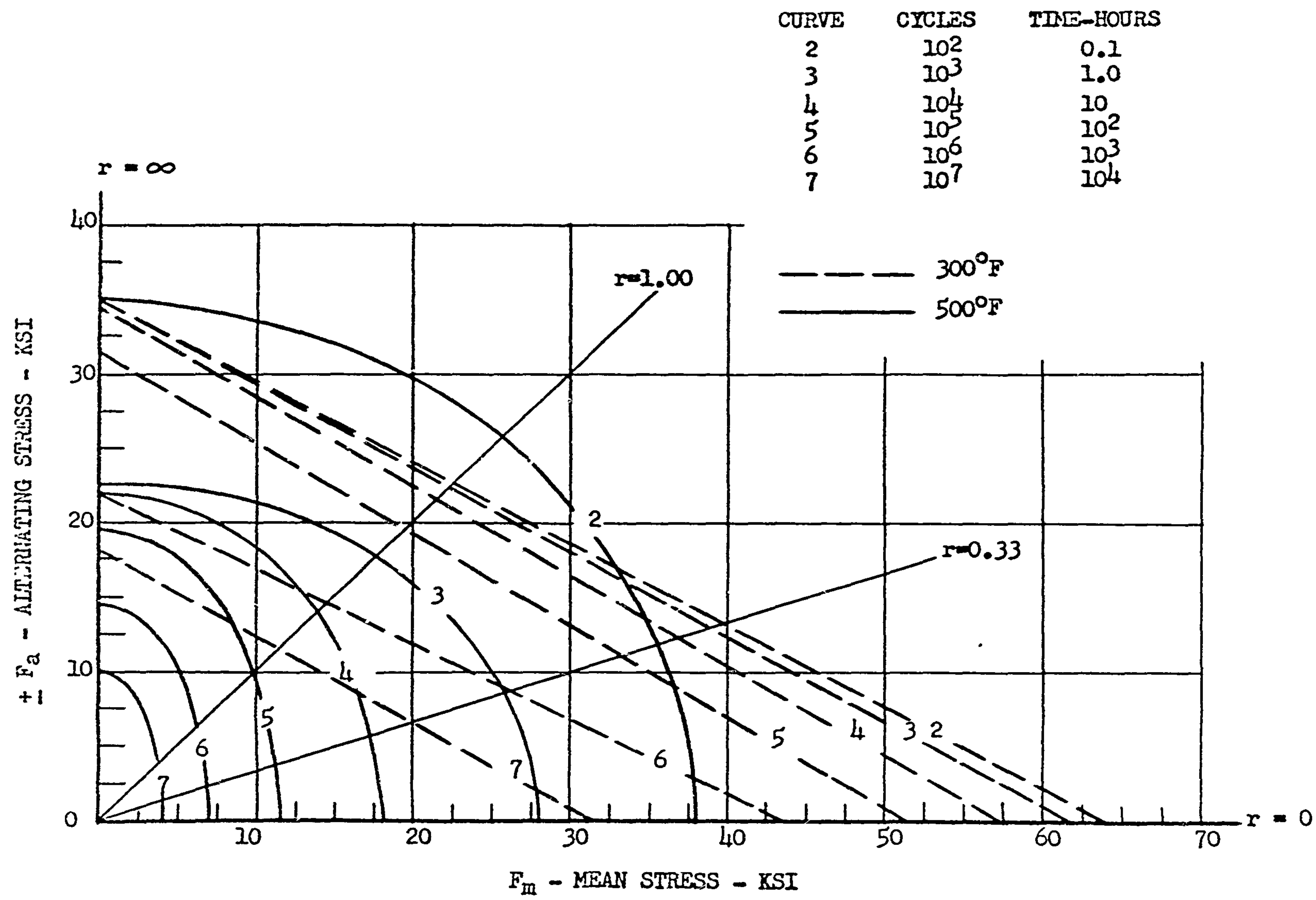


FIGURE 5. STRESS RANGE FATIGUE DIAGRAM FOR 2024-T3 ALUMINUM ALLOY FOR EXAMPLE 1

In Problem 6 the temperature is uniform on the cross-section but cycles in time -- heating up and cooling off. If the structure is restrained then thermal stresses are produced

$$f = - K\alpha ET(t) \tag{4}$$

where $T(t)$ is change in temperature above datum as a function of time and K is restraint coefficient. If the stress-strain curve of the material does not change with temperature, then the thermal stress cycles for various maximum values of $K\alpha T(t)$ are shown in Fig. 6. The cycle may be elastic, shakedown to an elastic case, or have a hysteresis loop. It should be noted that due to the Bauschinger effect (loading beyond yield stress in one direction decreases yield stress in the other direction, Ref. 7) the hysteresis loop may have yield stresses of the order of ten per cent smaller than the original values. The stress alternations occur about some mean stress, although the mean stress is not steady as it is just a point on the cycle. As indicated in Ref. 2, the prediction of thermal stress fatigue from conventional fatigue data is difficult. Thermal stress fatigue is a strain range cycle while conventional fatigue testing is a stress range cycle, and the two ranges cannot be simply correlated in the inelastic range. Also, there is variable temperature in the thermal stress case while the temperature is constant in the stress cycling. Some discussion of this problem with the use of total accumulated plastic strain is given in Ref. 2.

Figures 7 and 8 show the thermal stress cycles for aluminum alloy for which the stress-strain curves change on the first cycle. It is assumed that recovery takes place back to the 500°F curve in subsequent cycles. Creep effects are shown in Fig. 8. These creep effects depend on the time of the temperature cycle. In both Fig. 7 and 8 it is evident that a hysteresis loop arises unless the restraint coefficient is very small.

In Problem 7 the thermal stresses arise because of the nonuniform temperature. The total thermal load is zero so that the creep effects may be positive on some elements and negative on some elements of the cross-section, while some elements may not creep. The creep should approach zero over a period of time. By using a set of isochronous stress-strain curves (if available) it is possible to obtain an idea of the thermal stresses and strain under creep conditions.

<u>Example 2</u>. Consider the creep of a two-element structure (skin-stringer element) with no bowing or buckling with element 1 at 200°F and element 2 at 500°F. Use aluminum alloy with equal areas in each element.

The thermal stress equations for two elements are

$$e_2 = -\frac{(\alpha_2 T_2 - \alpha_1 T_1)}{1 + \frac{A_2 E_2}{A_1 E_1}} \tag{5}$$

$$e_1 = e_2 + (\alpha_2 T_2 - \alpha_1 T_1) \tag{6}$$

$$f_1 A_1 + f_2 A_2 = 0 \tag{7}$$

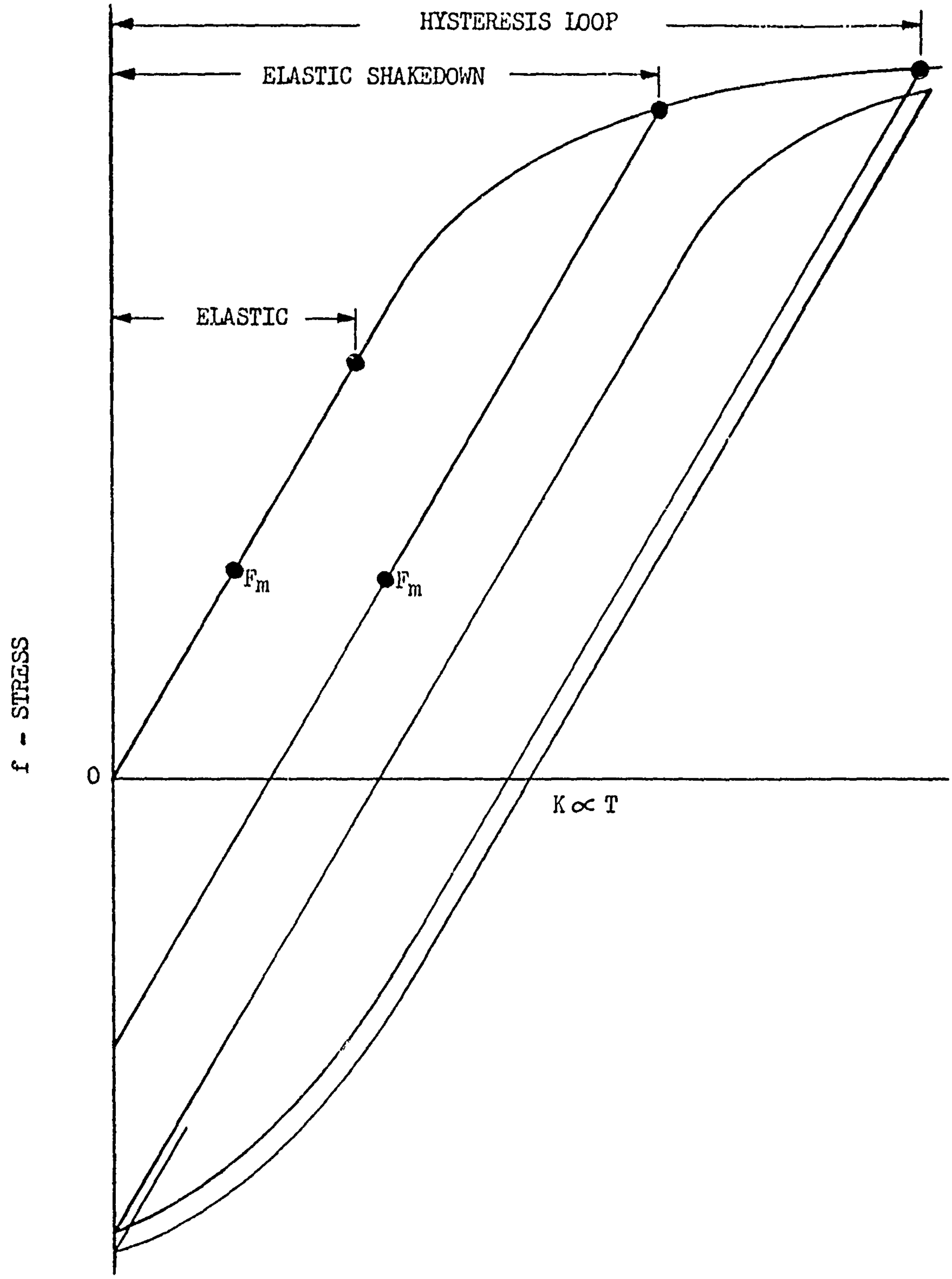


FIGURE 6. THERMAL STRESS FATIGUE

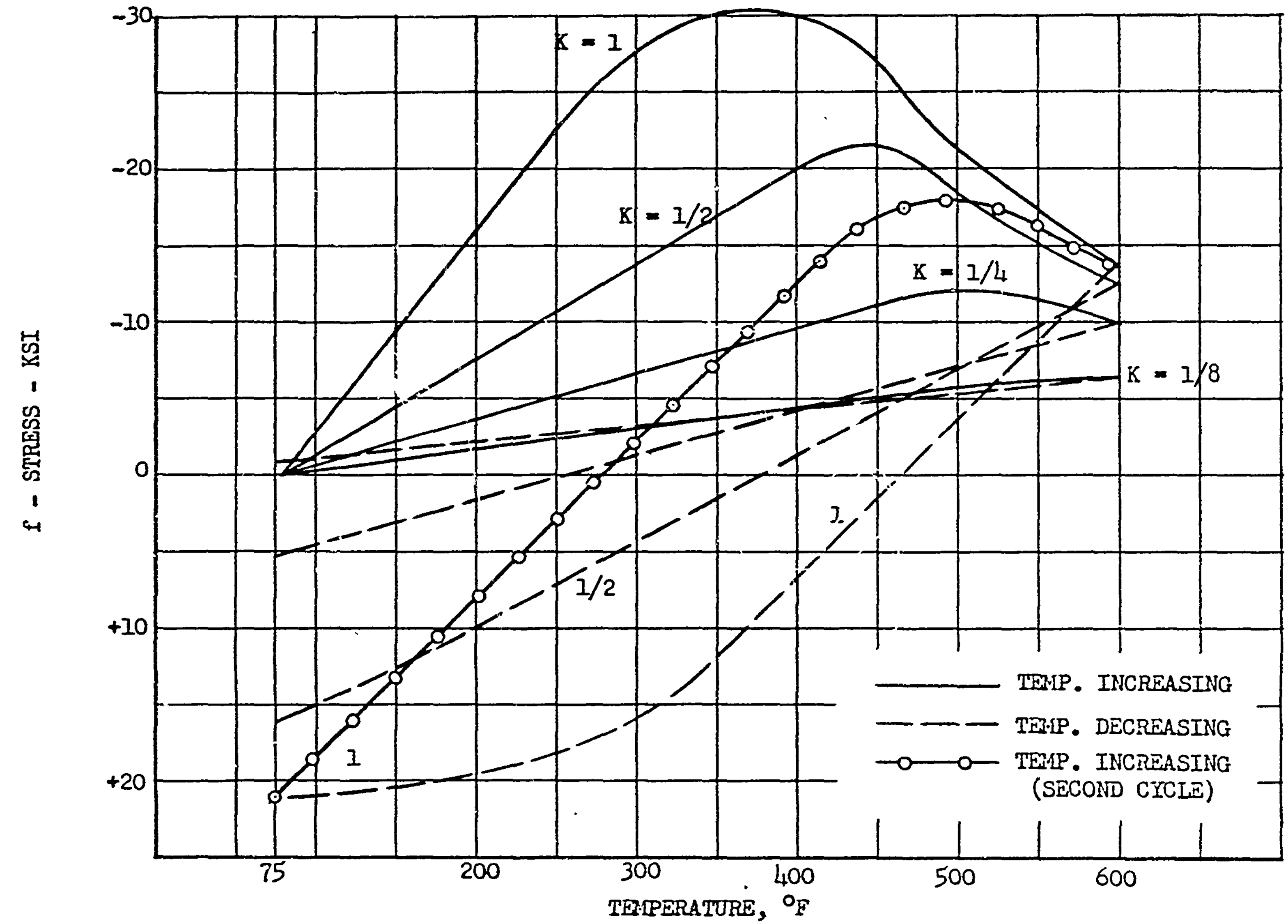


FIG. 7. THERMAL STRESSES IN RESTRAINED BAR FOR TEMPERATURE CYCLE TO 600°F

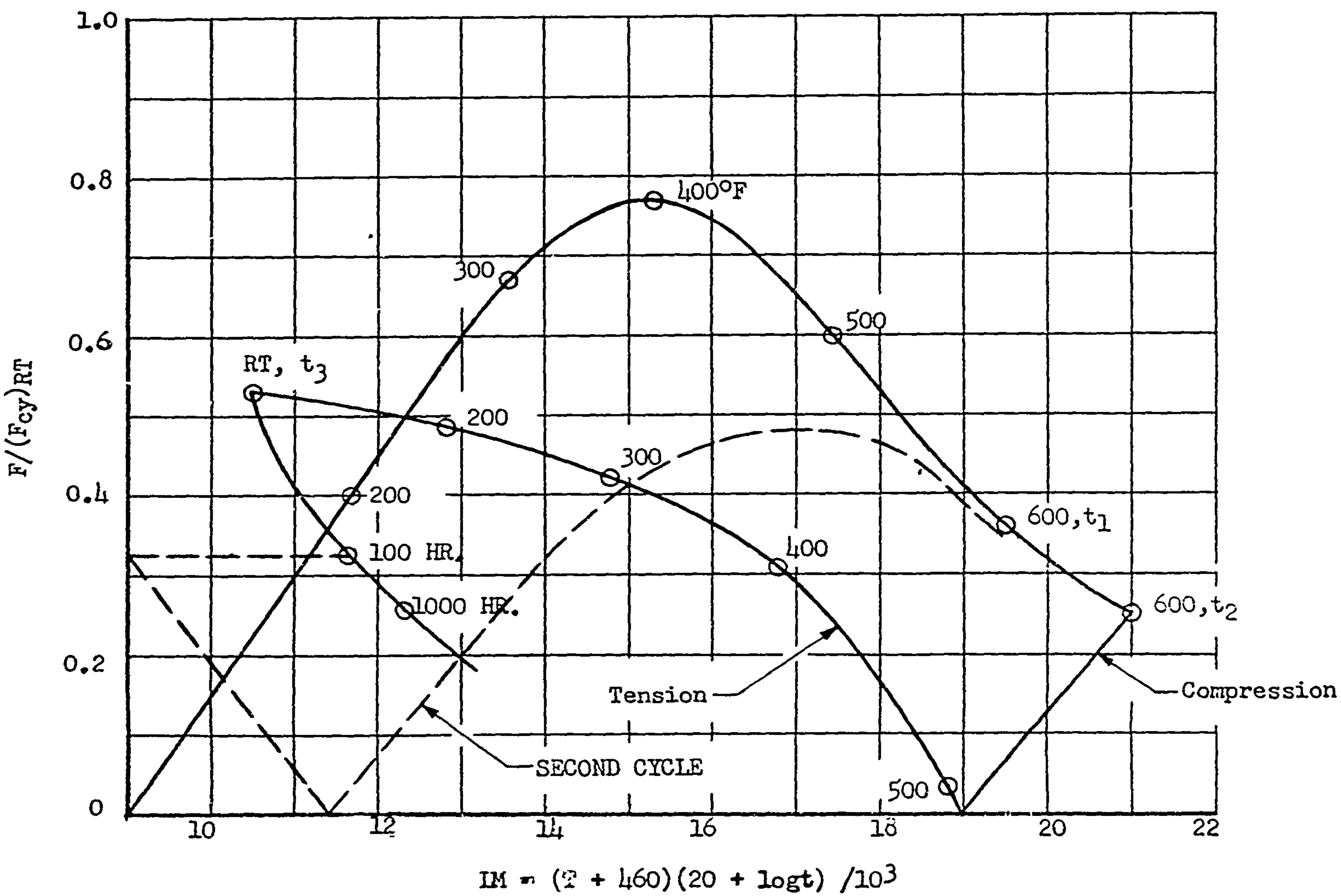


FIG. 8 INELASTIC THERMAL STRESSES WITH CREEP.

where E_1 and E_2 are the secant modulus values. Using $\alpha_1 = \alpha_2 = 13(10^{-6})$ and using the curves in Fig. 8, the strains and stresses are obtained by trial and error from Eqs. (5) - (7). The results are plotted in Fig. 9 for various creep times, where element 1 is in tension and element 2 in compression. Element 1 is assumed not to creep.

NONUNIFORM STEADY STATE TEMPERATURE

In Problem 8 when thermal stresses are present the mean stress on each element of the cross-section will be different and the magnitude of the alternating stress may be different. See Fig. 10. On some elements the mean stress may be quite large while on others it may be small. Thus some elements may creep or relax more than others so that the mean stress on the elements will change, decreasing on the creeping elements and increasing on the non-creeping elements. In Fig. 10, element 1 is assumed to creep more than element 2 with a resultant decrease in F_{m1} and increase in F_{m2}. Note also that the alternating stress also changes and may affect the strain by going further into the inelastic region on some elements. It appears from Fig. 10 that several possibilities could occur -- the creep might stop after the creeping elements shifted sufficient load to the other elements (if some elements did not creep and did not overload inelastically, the creep would always stop) or the creep might continue with all elements finally creeping together.

If the creep ceases then the problem becomes a fatigue problem with each element having its own mean stress and alternating stresses, which will differ from the other elements. If the creep continues, then each element must be checked by some procedure such as that discussed for Problem 4, with each element having its own creep-fatigue interaction. Would failure occur on an internal element before it would on a surface element? Would load shifting continue to occur in order to give a maximum life with the mixed creep-fatigue effects?

Problem 9 is similar to Problem 6 for uniform temperature in that thermal stress fatigue arises. For two elements, Fig. 11 shows that one element may be elastic while the other may be elastic, or shakedown to elastic state, or have a hysteresis loop. There may be some creep effects depending on time of the temperature cycle.

STRAIN ACCUMULATION

In Problem 10, an applied mean load is maintained while the temperature is cycled, producing thermal stresses. If some elements are inelastic on the heating cycle while other elements are inelastic on the cooling cycle, then it is possible for the total deformation of the cross-section to grow on each cycle. This strain accumulation may cease if elastic shakedown occurs (see Fig. 12) or it may continue to grow until failure occurs (see Fig. 13). If elastic shakedown occurs, then the problem becomes a thermal stress fatigue problem with a certain mean stress and alternating stress for each element of the cross-section. Creep may also occur depending on time of the cycle, although it may cease (see discussion of creep in Problem 8). If the strain accumulation diverges, then failure may occur before either fatigue or creep enters into the problem. What are the criteria for convergence and divergence of the strain growth? Is strain divergence permissible if the structure experiences only a few temperature cycles? Can the design be made such that strain divergence is avoided?

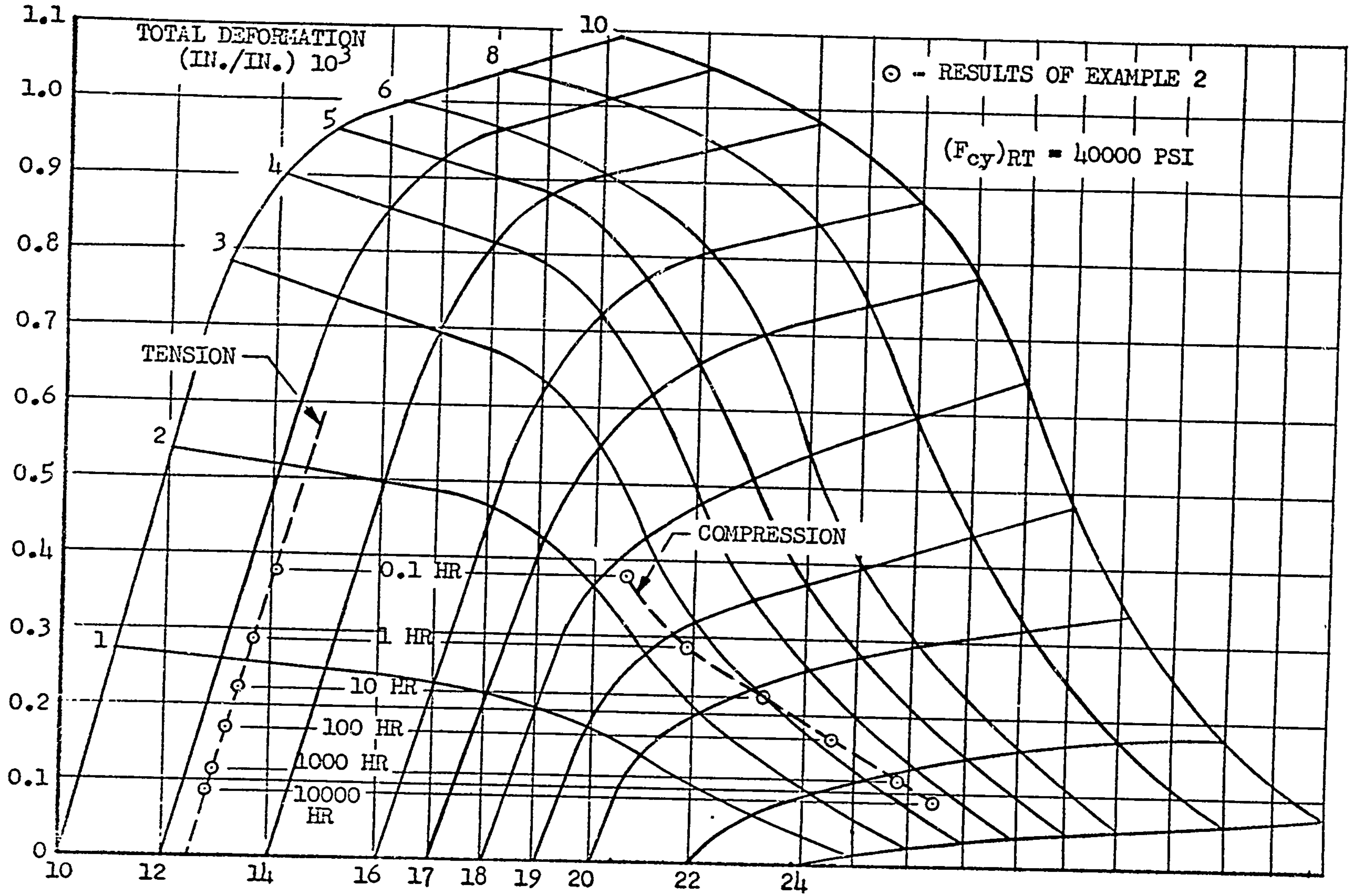


FIGURE 9. MASTER CREEP CURVES FOR 2024-T3 CLAD ALUMINUM ALLOY SHEET

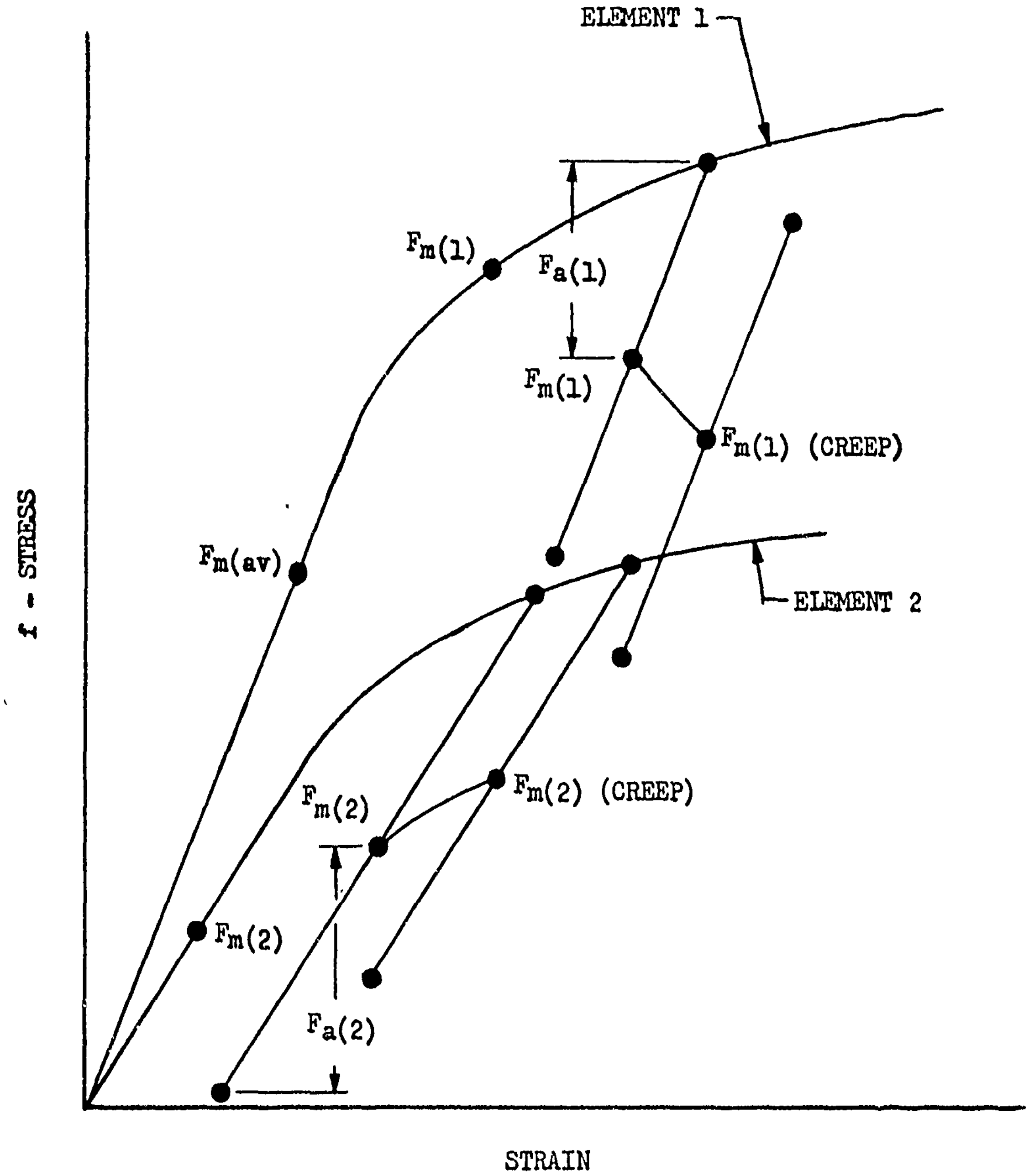


FIGURE 10. STRESSES FOR NON-UNIFORM STEADY-STATE TEMPERATURE

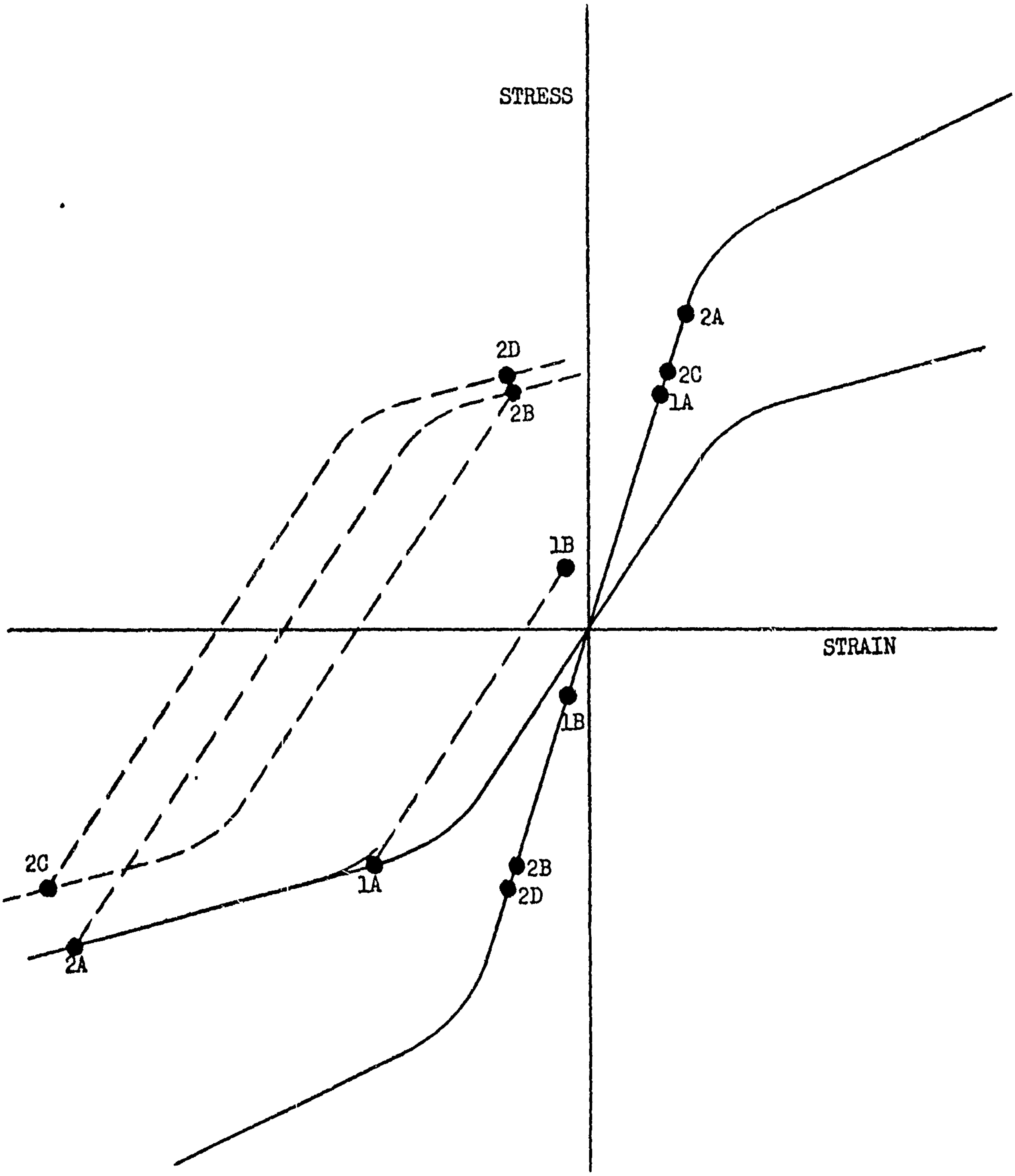


FIGURE 11. THERMAL STRESS FATIGUE AT NON-UNIFORM TEMPERATURE

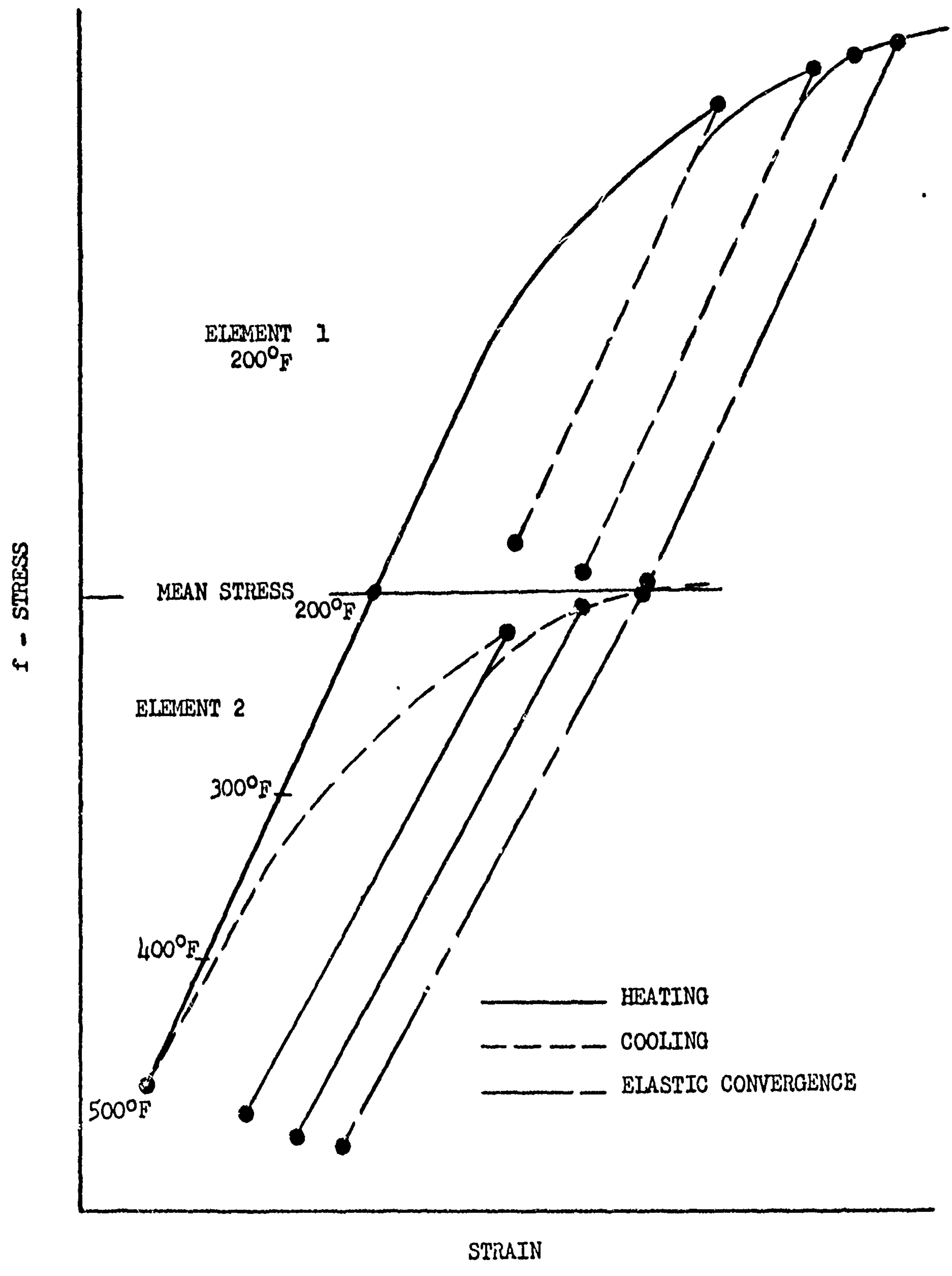


FIGURE 12. THERMAL CYCLING (CONVERGING)

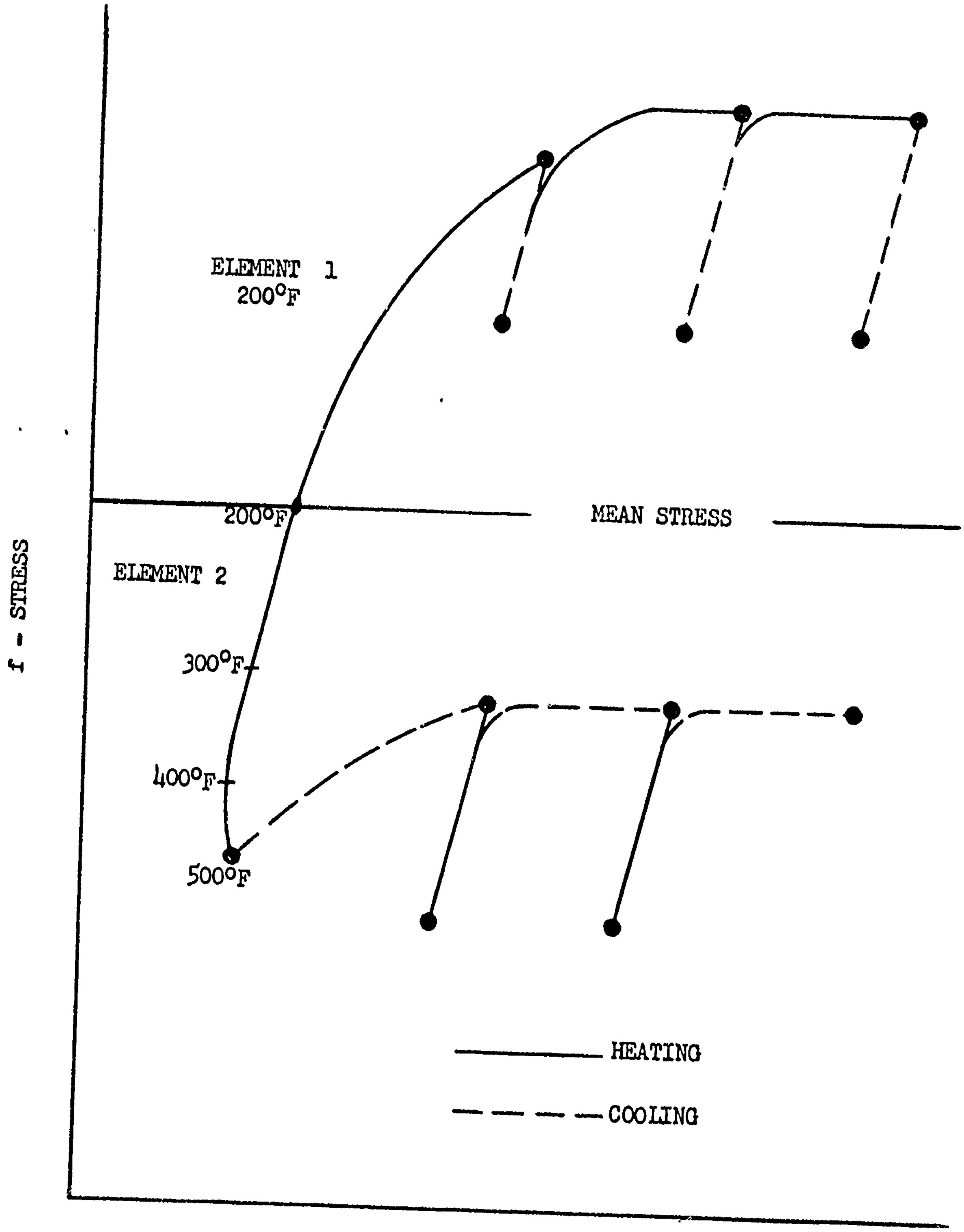


FIGURE 13. THERMAL CYCLING (DIVERGING)

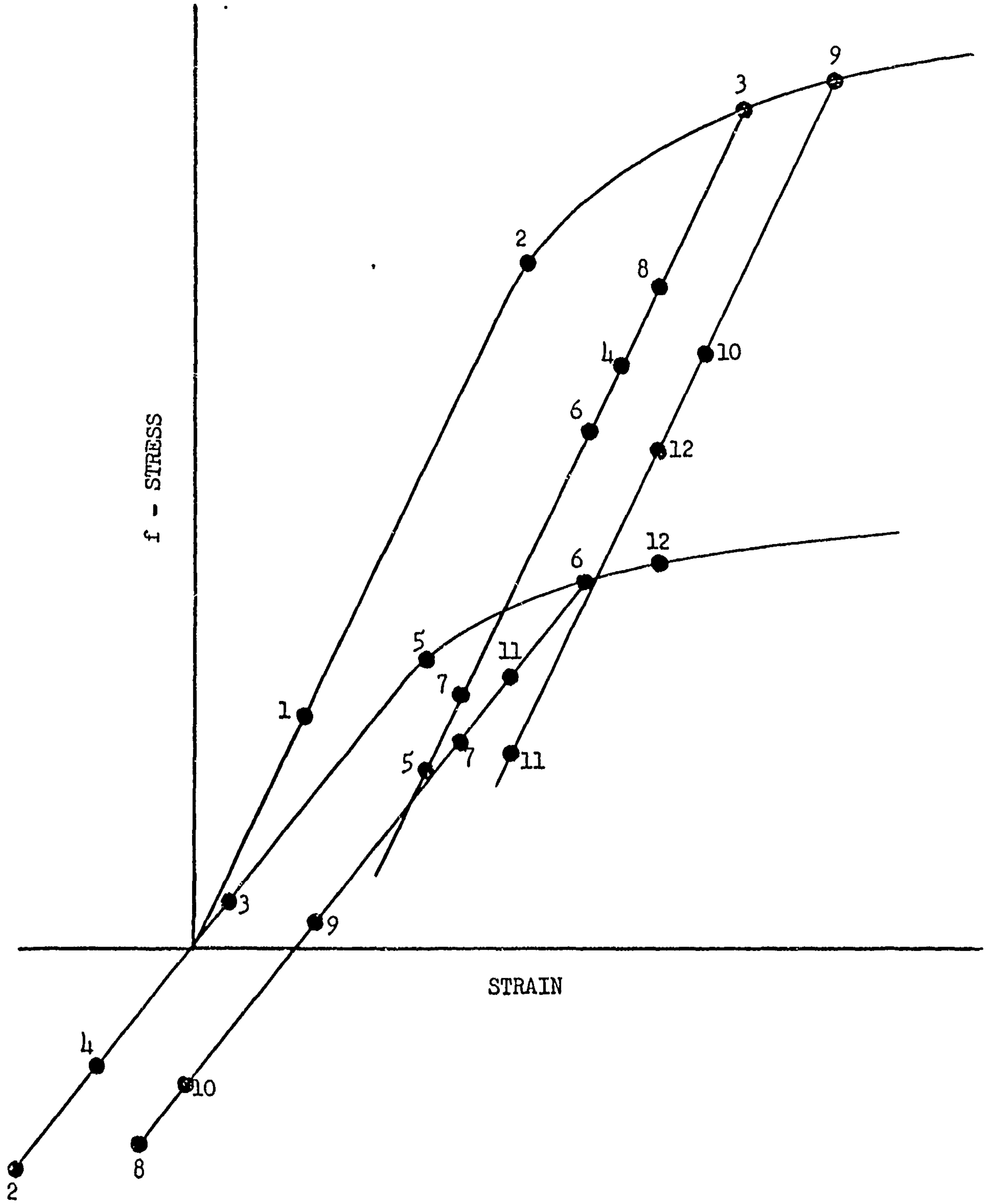


FIGURE 14. MEAN, THERMAL, AND ALTERNATING STRESSES

In Problem 11 the applied alternating stresses are added to Problem 10. These applied alternating stresses may be superimposed on the thermal stress cycle at any time so that each element will have a varying mean stress. The strain accumulation will be affected if the alternating stress produces any inelastic effects. Creep-fatigue is present so that all the effects of creep, fatigue, and strain growth may be acting simultaneously. It would appear that to prevent strain divergence elastic shakedown would have to be obtained for the combined mean stress and maximum alternating stress. How can this be done? It may be a serious limitation in some cases. Figure 14 shows how the strain accumulates for mean stress (1) plus thermal stress (2) plus alternating stress (3), minus alternating stress (4) minus thermal stress (5) plus alternating stress (6) minus alternating stress (7) plus thermal stress (8) plus alternating stress (9) minus alternating stress (10) minus thermal stress (11) plus alternating stress (12), etc.

CONCLUSIONS

Thermal fatigue analysis is involved with a group of problems affected in various ways by fatigue, creep. thermal stress, material property change, stress concentration, and strain accumulation. Some approximate calculations can be made for some of the problems, but little can be done with the more complicated combinations. In some cases it may be desirable to attempt to design around the problems of creep and strain accumulation so that the fatigue problem is somewhat like the room temperature case, except for possible thermal stress effects and material property variations.

ACKNOWLEDGEMENT

I wish to express my appreciation to the Structures Development Group, North American Aviation, Columbus Division, for preparing the examples and figures in this paper.

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SONIC FATIGUE DESIGN ANALYSIS

By

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ABSTRACT

The importance of detailed design is emphasized in a review of the general philosophies of industry towards structural design for sonic fatigue. The sonic load and response problem is described and illustrated with examples of measured data obtained from the RB-66 aircraft. Load and structural response theories are summarized and the difficulties in applying these theories to design are discussed. A comparison is shown between measured stress response of typical aircraft structure and that obtained using Miles' single degree of freedom analysis. The need for laboratory fatigue tests is determined by discussion of such topics as cumulative damage theories, differences in stress amplitude probability produced by different techniques of data reduction, the influence of non-zero mean stress, and the effects of stress concentration.

INTRODUCTION

With the introduction of high-powered propulsion systems, and paralleling their continued development, there has been an accompanying increase in sonic fatigue problems. The degree of damage, dependent primarily on the quality of the detailed design as well as the sound intensity, has ranged from failed hydraulic and electrical line clamps to the almost complete destruction of wing trailing edge and control surface structure. Often when failure occurs late in the development cycle of an aircraft, costly redesign of major sub-assemblies is required.

To achieve a solution to this problem much engineering and scientific effort has been expended. Theoretical work and laboratory testing accomplished both at the universities and by industry[1] has yielded much insight into the physical nature of the forces and responses involved; however, a formidable engineering problem exists in applying these theories to the design of complex aircraft and missile components. The task is additionally complicated by the fact that rather sizeable errors are common in measuring many of the important parameters of the problem. As in any mechanical design problem, the three phases in the solution are to define the loads, to predict the response, and to determine whether the response will yield the required level of reliability. Because of the inherent complexities in both analytical and testing procedures, a well coordinated analytical and experimental program is required to achieve these goals. In fact, the final judgment of the reliability of a design subjected to severe sonic loading must be based upon a sonic proof test[2] of a full-scale article.

To provide a systematic procedure for the development of sonic fatigue resistant structure, the Aircraft Laboratory (WADC) has prepared "Detail Requirements for Structural Fatigue Certification Programs[3]." The Douglas Aircraft Company has comcompleted all phases of the program in conjunction with the development of the RB-66[4,5]. In addition, sonic fatigue resistant structure has been developed and tested for the DC-8 in a program[6] closely paralleling that proposed by WADC. During these development programs, a constant effort has been maintained to monitor test data in order to establish analytical or empirical design techniques.

The purpose of this paper, therefore, is to describe the sonic fatigue phenomena, to discuss the analytical phase of sonic fatigue, to illustrate its present limitations based upon available test data, and to indicate where the emphasis should be placed to remove these limitations.

DESCRIPTION OF LOAD SPECTRUM

Sonic loads result from the fluctuating pressure caused by the turbulent mixing of a high velocity jet with surrounding air. These pressure fluctuations, having randomly varying intensity and covering a wide range of frequencies, are defined at a particular point in the sound field by both their power spectral density and spatial correlation.

Power spectral density is of primary importance to the structural engineer since it is a measure of the mean square pressure available at each frequency to excite resonant dynamic response in the adjacent structure. Ideally, power spectral density at a given frequency would be obtained by passing the random voltage output of a microphone or accelerometer through a narrow band electrical filter and measuring the power of the filtered signal with a wattmeter. In reality, commercially available power spectral density analyzers are used.

As illustrated in Figure 1, the shape of the power spectral density curve changes radically with geographical location on the aircraft. Near the engine exhaust plane the noise is relatively white (constant power spectrum over all frequencies), while downstream near the aft fuselage and empennage there is a noticeable increase in power spectral density in the frequency range corresponding to most fundamental structural resonances (200 to 500 cps). A general awareness of this phenomena should indicate to the designer the importance of maintaining high resonant frequencies in the aft structure. Similar considerations should be given to areas such as the elevator lower surface at the root where reflected sound from three surfaces is compounded to cause a buildup in the low frequency range.

MEASURED POWER SPECTRAL DENSITY (ϕ_N) – PSF^2/CPS

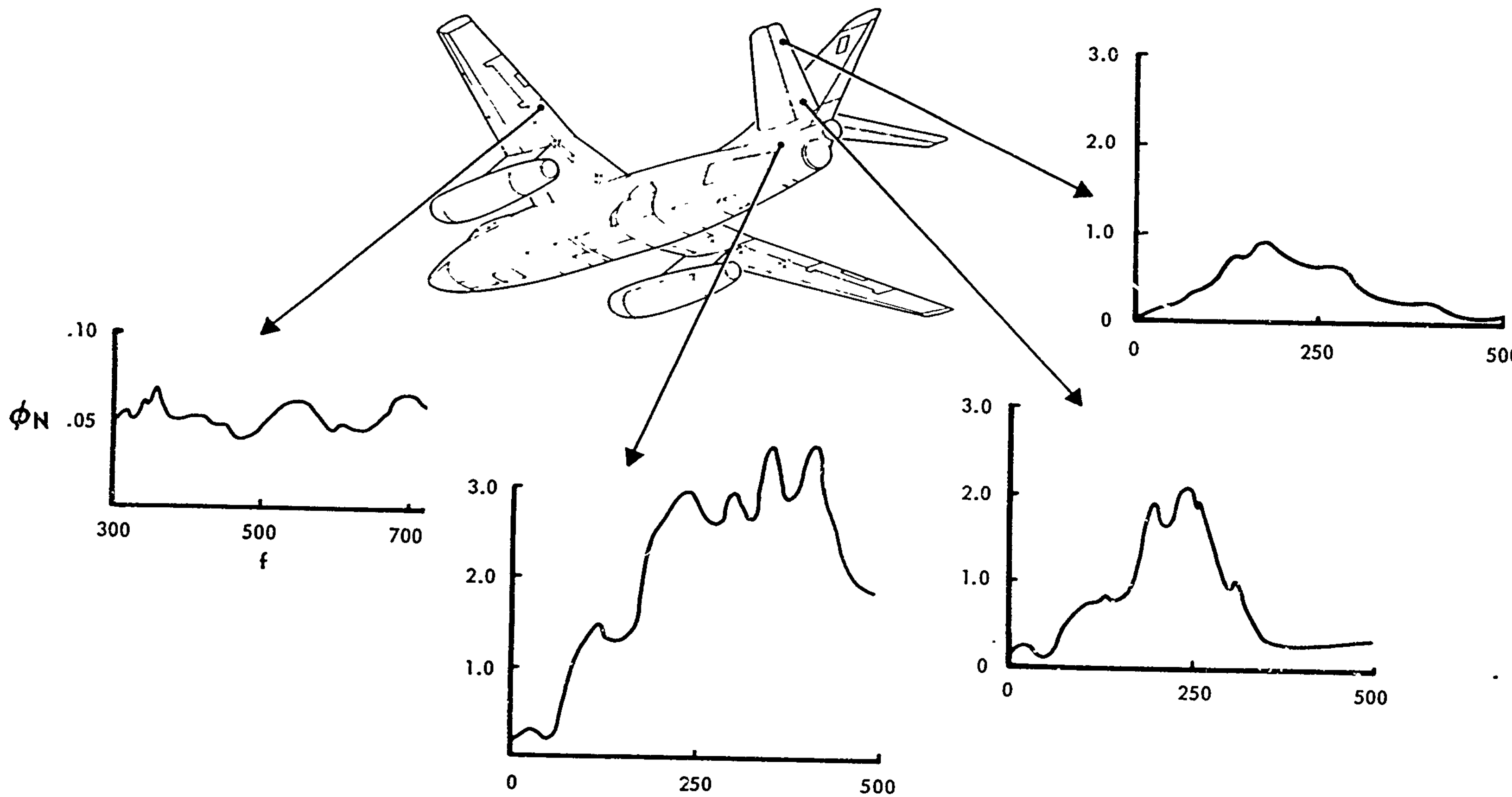


Figure 1

Spatial correlation defines the phase relationship between pressures at different points on the structure. For practical design purposes the pressures can be considered in phase or well correlated over distances short in comparison to a wave length. In general, this means that although spatial correlation is an important consideration in the overall dynamic response of a fuselage or missile shell, it has less importance in the design of individual skin panels between stringers. A discussion of spatial correlation is beyond the scope of this paper, however the dynamic response of a hypothetical missile, including the effects of spatial correlation, has been presented by Dyer[7].

ANALYTICAL PREDICTION OF LOAD SPECTRUM

The present "state-of-the-art" does not include a reliable technique for calculating power spectra; however, Powell[8] has shown that measured sound spectra can be extrapolated to include variations in jet velocity and nozzle area. This technique is ordinarily employed to obtain preliminary design estimates, but sound reflections from the ground and adjacent structure sufficiently affect the predicted spectra that measurements on a prototype article are necessary. To determine the rough magnitude of error involved in the prediction of the acoustic loading, comparisons were made between measured and extrapolated data from two aircraft equipped with engines of the 10,000 pound thrust class operating over a wide range of thrust conditions. Table 1 illustrates the results of this comparison in terms of the possible errors in acoustical loads. The need for a sound survey of a prototype aircraft is clearly indicated in order to reduce the margin for error in the sonic fatigue design analysis as early as possible in the development cycle.

When it is significant in the design problem, pressure correlation information must be determined experimentally; however, some experimental evidence[9] indicates that small model jets may be used to obtain correlation data.

DESIGN CRITERIA

Little design criteria has thus far been established for sonic fatigue; however it is certainly an important consideration in the overall load prediction problem. Several items significant to design criteria are summarized in the following discussion.

The acoustic loading is a maximum during ground run and takeoff roll and hence is not generally coupled with other structural loads. Acoustical loads decrease rapidly with increasing forward airspeed as the relative velocity between the jet exhaust and the surrounding air is reduced. A further reduction occurs as a result of the decreased air density at altitude, the absence of ground reflected sound, and the fact that reduced power settings are used for cruise.

Consideration of engine growth potential is also essential in the early design stages of an aircraft. For example, higher thrust engines were installed on the RB-66 aircraft late in the production program and the resulting 40 percent increase in acoustical pressure loads required an extensive structural modification program. Similar redesigns have been required with other jet aircraft.

Another important detail that should be considered concerns the design criteria for the lightly loaded side of a thin shell structure of small depth. To aid in the solution of this problem, experiments were conducted[5] on skin and rib type control surfaces having depths of four to six inches. The structural response measurements showed that the effective sound attenuation through the structure was on the order of 10 per cent and that when both sides of a symmetrical control surface were exposed

ERRORS IN PREDICTION OF NEAR FIELD ACOUSTICAL LOAD

TABLE I

METHOD OF PREDICTION OR MEASUREMENT	ERROR IN DECIBELS	ERROR IN PERCENT
EXTRAPOLATION OF FREE FIELD DATA FROM 10,000-LB THRUST ENGINE TO STRUCTURAL LOADS FOR 20,000-LB ENGINE	±7	+ 124% − 55%
EXTRAPOLATION OF FREE FIELD DATA FROM 10,000-LB THRUST ENGINE TO STRUCTURAL LOADS FOR 15,000-LB ENGINE	±5	+ 78% − 44%
EXTRAPOLATION OF MEASURED ACOUSTICAL LOADS FOR THRUST VARIATIONS IN THE ORDER OF 2000 LB	±3	+ 41% − 29%
INTERPOLATION OF MEASURED ACOUSTICAL LOADS BETWEEN THRUST LEVELS OF 10,000 LB AND 12,000 LB	±2	+ 26% − 21%
MEASUREMENT AT A POINT ON THE AIRCRAFT STRUCTURE	±1	+ 12% − 11%

Table 1

to the same sound pressure level, the resulting structural response was 40 per cent greater than with only one side exposed. This factor would be especially significant in extrapolating siren test data to the design of a rudder loaded on both right and left sides.

DESCRIPTION OF STRUCTURAL RESPONSE

Since typical aircraft structure is very lightly damped, its response to a wide band of exciting frequencies is essentially a sharply tuned resonant phenomena. The peak stress occurring at any given time cannot be predicted; however, the probability of occurrence of a particular peak stress is well described by the Rayleigh probability distribution which is in turn related to the overall RMS stress. This quantity is easily measured, as well as being the object of both simple and complex response theories.

Examples of several types of structural response are shown in Figure 2, together with corresponding sound pressure spectra. As a general rule skin and stringer panels respond in a single mode and multiple mode response is confined to skin supported on ribs.

Several studies were conducted at the Douglas Company to determine the origin of the observed multi-mode response as well as to improve the technique for predicting panel frequencies. The results indicated that for simple beams having various end conditions, a negligible response should be encountered in modes higher than the fundamental.[10] The beams were theoretically subjected to a plane pressure wave characterized by a power spectrum with peak power at 200 cycles per second and decreasing power at higher frequencies. Unpublished data concerning the response of a three-span beam with fixed ends indicated a similar result. Both equal and unequal spans were considered and in all cases higher modes of response were negligible compared with the lowest symmetrical mode. In addition, phase checks made during tests of full-scale structural components have consistently shown that adjacent skin panels vibrate symmetrically and in phase.

The previously mentioned results suggested that the higher modes of response must be caused by the dynamic coupling of the plating and sub-structure. Although no calculations have been made to predict the higher modes, the application of energy techniques to the simpler skin and stringer problem yielded the fundamental coupled panel frequencies for six test specimens within a scatter band of only ±15 cycles per second. Since these specimens included a wide variation in skin and stringer stiffnesses, it is suggested that the crux of the modal response problem lies in developing design charts for simple panels with flexible supporting structure.

ANALYTICAL PREDICTION OF STRUCTURAL RESPONSE

During the last few years a number of theoretical papers have been written concerning the response of structure to jet noise loading. The complexity of these papers varies from the simple results of Miles[11] to the more complex results of Powell[12], Thomson and Barton[13], and Eringren[14]. Essentially the more complex papers attempt to compensate for many degrees of freedom and the spatial quality of the sound waves.

For preliminary design requirements, Miles' linear single degree of freedom approach given by equation (1) yields results compatible with the overall accuracy of the problem. However, for optimizing a design, the analytical procedure is

MEASURED STRESS POWER SPECTRAL DENSITY OF TYPICAL RB-66 STRUCTURE

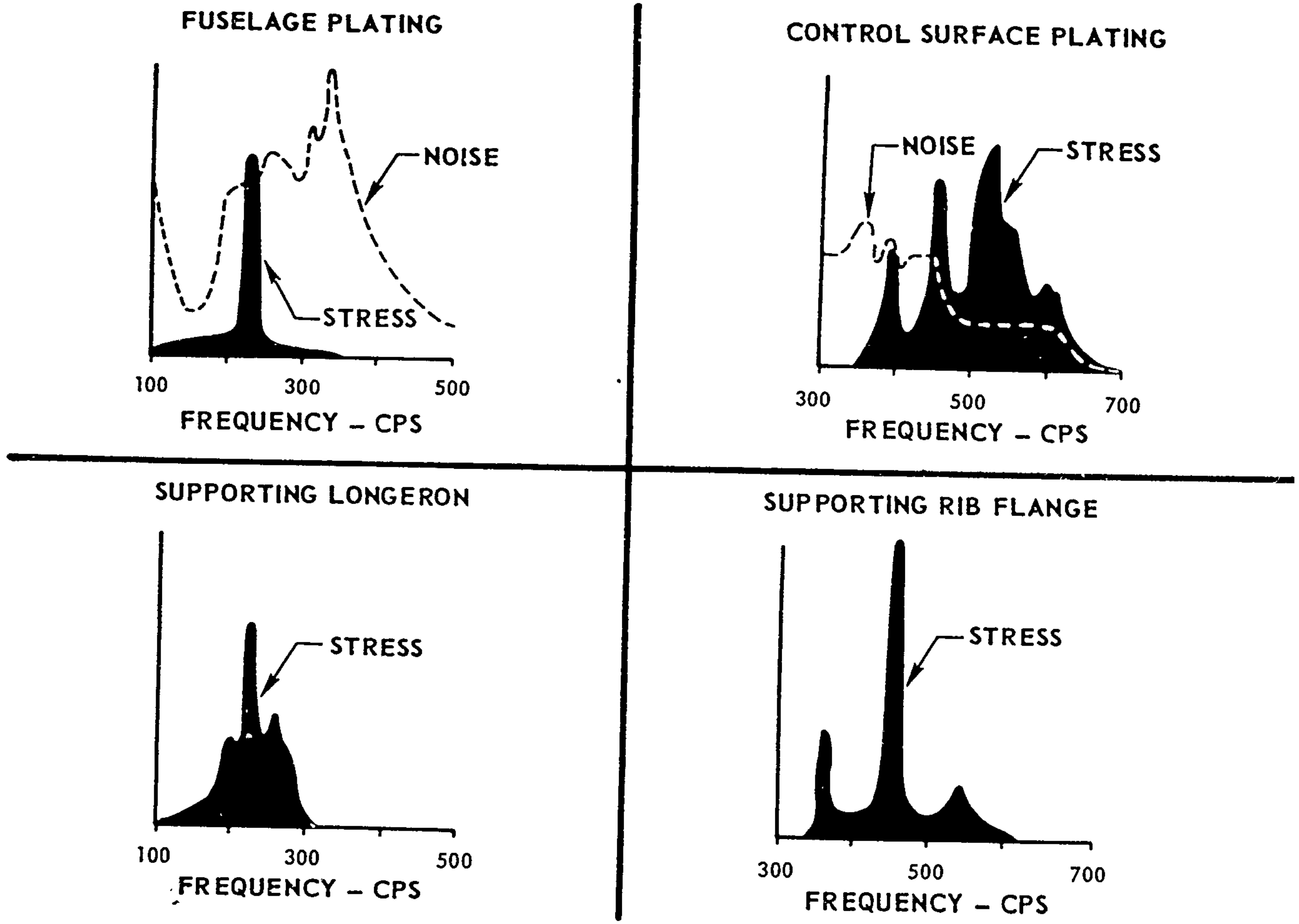


Figure 2

inadequate and a combined analytical and experimental program is recommended such as that used for the DC-8[6].

$$\bar{\sigma}^2 = \frac{\pi}{4\delta} f_o \, \phi_N(f_o) \, \sigma_o^2 \tag{1}$$

where

$\bar{\sigma}$ = Root mean square stress response ~ psi

δ = System damping expressed in terms of ratio to critical damping

f_o = Fundamental resonant frequency ~ cps

$\phi_N(f_o)$ = Power spectral density of the noise ~ $(psf)^2$/cps

σ_o = Static stress response to a unit load ~ psi/psf

Excellent agreement between measured stresses and those predicted by equation (1) has been shown by Lassiter and Hess[15]. Their results, obtained using simple 11" x 13" flat and curved clamped edge panels, demonstrate the reliability of Miles' approach for cases where the important parameters are known.

Figure 3 shows a comparison between measured stresses on the RB-66 fuselage and those predicted by equation (1). In this case all terms in the equation were known except the damping. Static pressure boxes were placed against the side of the fuselage to obtain σ_o. A narrow band analysis of the response data yielded f_o which, for the data analyzed, corresponded to the only observed mode of response. No attempt was made to measure the damping because of the lengthy data reduction time required and also because it was desirable to check the validity of the damping ratios normally assumed for preliminary design. Since the damping ratio δ is affected by construction techniques, proximity of joints and splices, sealants, as well as the stress level, it must either be estimated based on tests of similar structure or measured. In this case δ = .01 was assumed for the skin panels and δ = .02 for the longerons. The resulting scatter of data shown in Figure 3 could easily be caused by variations in δ from the assumed values. Since these variations are a part of the overall design problem, however, it can be concluded that the errors shown represent the minimum attainable errors with a strictly analytical approach to the problem. Since the analysis would be subject to additional error in the calculation of f_o and σ_o, it would be entirely inadequate for the purpose of final design.

To compensate for these inadequacies in the ability to determine f_o, σ_o and δ for complex structure, a method was developed for the DC-8 design whereby the parameters f_o and δ were determined from the shape of the measured frequency response curves. A conversion was established between the sinusoidal structural response to siren loading and the random response to jet noise loading. Since σ_o is common to both problems it cancels in the conversion equation and is thus eliminated as a potential error. Using an extension of the linear single degree of freedom approach, this technique also permits consideration of higher modes of response.

FATIGUE ANALYSIS

Fatigue analysis as applied to the sonic fatigue problem requires a statistical description and interpretation of the randomly varying stress peaks, a cumulative damage rule, a basic S-N curve extending into the high cycle range, and a knowledge

COMPARISON BETWEEN MEASURED AND THEORETICAL STRESSES

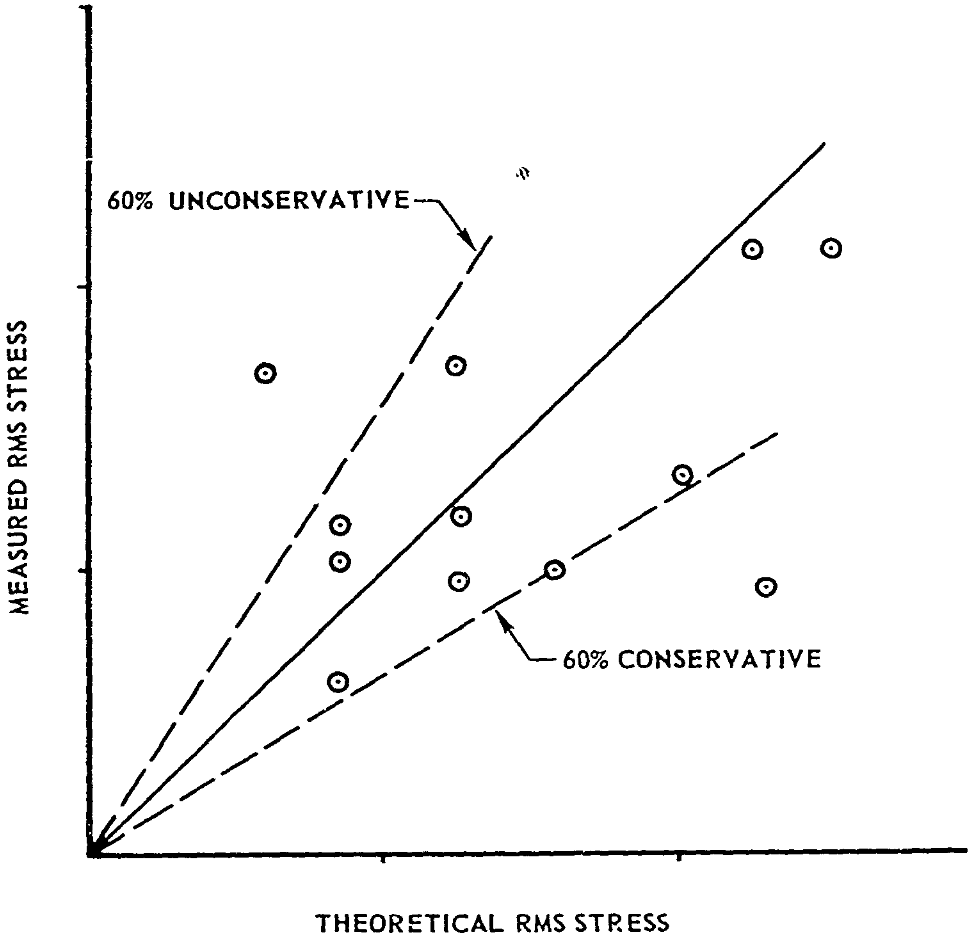


Figure 3

of the effective stress concentration factors. With these basic tools it would be possible to analytically construct random fatigue curves where the cycles to failure are related to the RMS stress rather than the peak stress. However, as in the other phases of sonic fatigue design, the state-of-the-art has not progressed to the point where much dependency can be placed on analysis. On the other hand, random fatigue curves constructed as described can be used to extrapolate existing test data as well as to set up rules for accelerating random fatigue tests. The construction of two random fatigue curves is described in the following discussion to illustrate the technique as well as to show differences in result based upon different interpretations of the basic data.

For a given RMS stress the probable number of cycles occurring at other stress levels is well defined by the Rayleigh probability distribution according to equation (2).[5,16]

$$P\left(\frac{\sigma}{\bar{\sigma}}\right) = \frac{\sigma}{\bar{\sigma}} e^{-\left(\frac{\sigma}{\bar{\sigma}}\right)^2} \qquad (2)$$

where σ is the peak stress response and $\bar{\sigma}$ is the RMS stress. If then a cumulative damage theory is assumed, a random fatigue curve can be generated based on RMS stress. For the purpose of this example consider Miner's Hypothesis of Cumulative Damage[17], expressed in equation (3), as valid.

$$\text{Damage at failure} = 1 = \int_0^{\sigma_{max}} \frac{dn(\sigma)}{N(\sigma)} \qquad (3)$$

where $n(\sigma)$ is the number of cycles n at stress σ, and $N(\sigma)$ is the allowable number of cycles at stress σ.

Fatigue curves currently in use give the relationship $N(\sigma)$ in the form of S-N diagrams, where the fluctuating stress σ is plotted against the number of cycles to failure N. A similar curve may be constructed in terms of the RMS stress ($\bar{\sigma}$) by combining equations (2) and (3) into equation (4) and solving for the total number of cycles to failure $N(\bar{\sigma})$.

$$N(\bar{\sigma}) = \frac{1}{\displaystyle\int_0^{\left(\frac{\sigma}{\bar{\sigma}}\right)_{max}} \frac{P\left(\frac{\sigma}{\bar{\sigma}}\right) d\left(\frac{\sigma}{\bar{\sigma}}\right)}{N(\sigma)}} \qquad (4)$$

This technique tacitly assumes that following each positive stress peak there will be an equal negative stress peak. Although the Rayleigh distribution adequately describes the distribution of both positive and negative peaks, it does not necessarily imply that equal positive and negative peaks occur in the same cycle. In fact, examination of oscillograph records clearly shows that adjacent positive and negative stress peaks are often unequal in magnitude.

As an alternative method for describing the fluctuating stress, the peaks can be considered as a variable alternating stress superimposed on a variable mean stress

as illustrated in Figure 4. When test results were analyzed statistically, the peak stresses were found to obey the Rayleigh probability distribution; however, both the mean and alternating stresses were more nearly represented by normal probability distributions. Although no definite relationship between the RMS stress and the normal curve could be established, reasonable correlation was obtained by considering the RMS stress equal to the mean alternating stress. A similar integration to that described in equation (4) using the experimentally derived normal distribution of alternating stress yields the random fatigue curve shown in Figure 5. In order to evaluate the effect of mean stress, the joint probability of occurrence of a given σ_A and σ_M were considered. The resulting random fatigue curve compared with Figure 5 showed that the effect of mean stress was negligible. Thus by merely changing the interpretation of the stress spectra, a significantly different random fatigue curve has been generated. A more detailed discussion of this subject is currently being prepared by Dr. H. C. Schjelderup for presentation in the Journal of Aeronautical Sciences.

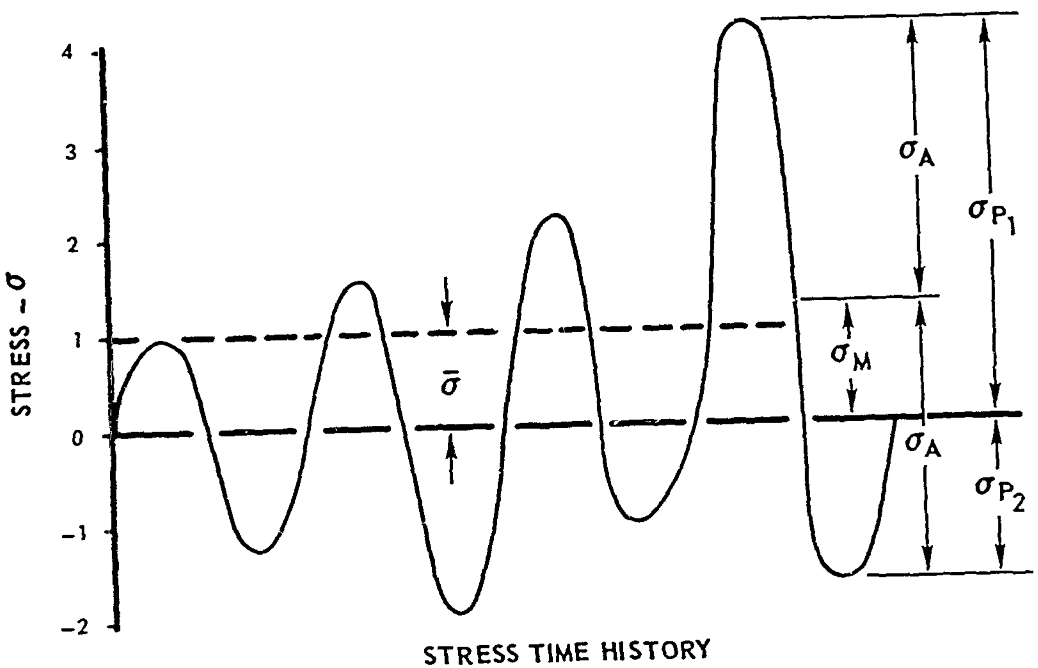
Additional variations are introduced by considering other than Miner's linear damage theory. A summary of current damage theories related to the sonic fatigue problem[18] indicates that a more reasonable approach would be to concentrate on experimentally determining the random fatigue curve. Rotating beam random fatigue experiments by Freudenthal and Heller[19] have indicated that Miner's damage theory is unconservative and that a safe life would be one tenth that indicated by the linear theory. The lack of significance of endurance limit was also shown experimentally for both aluminum and steel.

The effect of stress concentration factors offers an additional hurdle to the designer. Since the sonic loading is generally in a plane rotated 90 degrees from the normal load carrying paths in an aircraft or missile, there has been little test data accumulated regarding stress concentration factors. This lack of data is most unfortunate since nearly all sonic fatigue failures occur at stress concentration points. Beside concentrations caused by rivet and drain holes, slight amounts of pre-stressing during assembly operations have resulted in many premature failures. A series of skin failures on the outboard end of a control surface was attributed to a slight skin canning which was well within normal production tolerances. No failures occured in the inboard section of the control surface where the loading and stress level were 1.5 times as great and the skins were smooth.

DETAIL DESIGN REQUIREMENTS

Much can be done to prevent sonic fatigue by eliminating or reducing areas of stress concentration. This has the added effect of shifting the maximum stresses out into the smooth flat area of the structure where they can be measured and computed with greater accuracy, and where the fatigue life can be estimated using existing unnotched data. An example of this concept is shown in Figure 6 which illustrates features of the control surface design of the DC-8. The use of a bonded scalloped doubler between the skin and rib flange improved the fatigue life of the structure by reinforcing the critical area and did so for less weight penalty than would result from increasing the skin thickness. In areas subjected to higher acoustic loading, an extruded tee section was used as a rib cap. The advantage of the tee section is three fold: it allows for a more symmetrical load path, it reduces the skin panel size, and it reduces the stress concentration in the flange bend radius. The back to back rib flanges shown in Figure 6 accomplish the same thing but to a lesser extent.

PEAK STRESS SPECTRUM AND PROBABILITY DENSITY



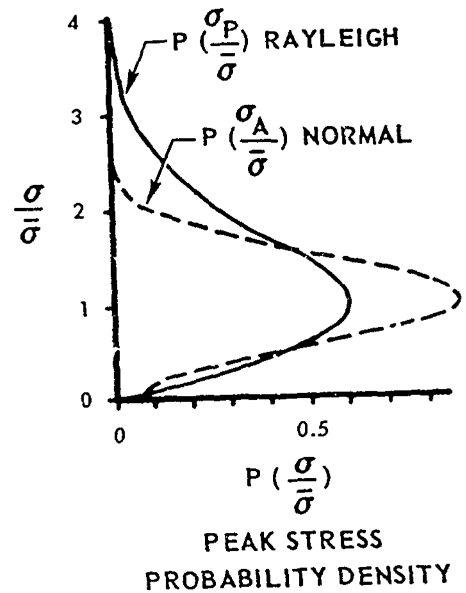


Figure 4

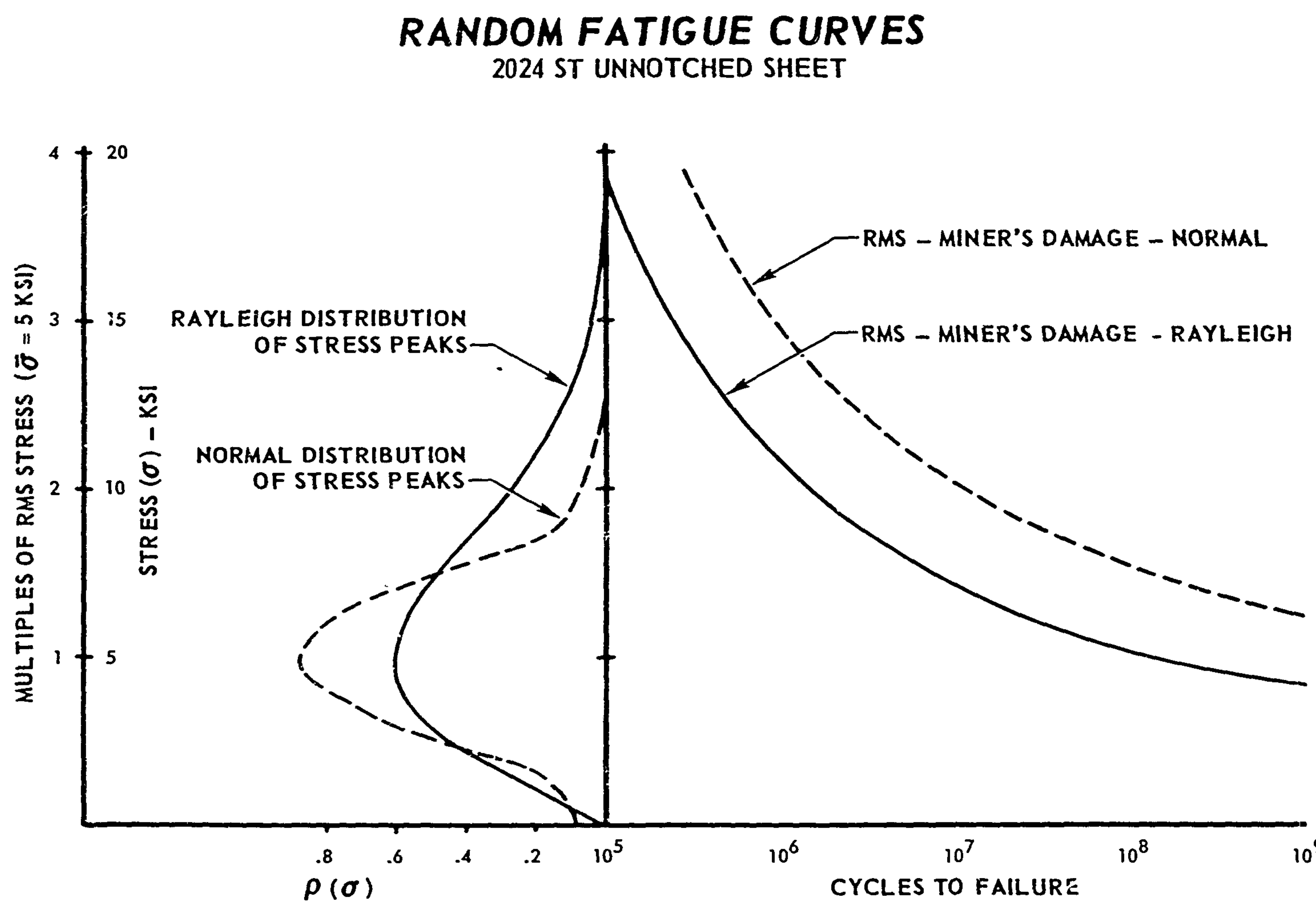

RANDOM FATIGUE CURVES
2024 ST UNNOTCHED SHEET
RAYLEIGH DISTRIBUTION OF STRESS PEAKS
NORMAL DISTRIBUTION OF STRESS PEAKS
RMS – MINER'S DAMAGE – NORMAL
RMS – MINER'S DAMAGE – RAYLEIGH
MULTIPLES OF RMS STRESS ($\bar{\sigma}$ = 5 KSI)
STRESS (σ) – KSI
4
3
2
1
20
15
10
5
.8
.6
.4
.2
10^5
10^6
10^7
10^8
10^9
$\rho(\sigma)$
CYCLES TO FAILURE

Figure 5

DETAIL DC-8 DESIGN OF SKIN TO RIB ATTACHMENT

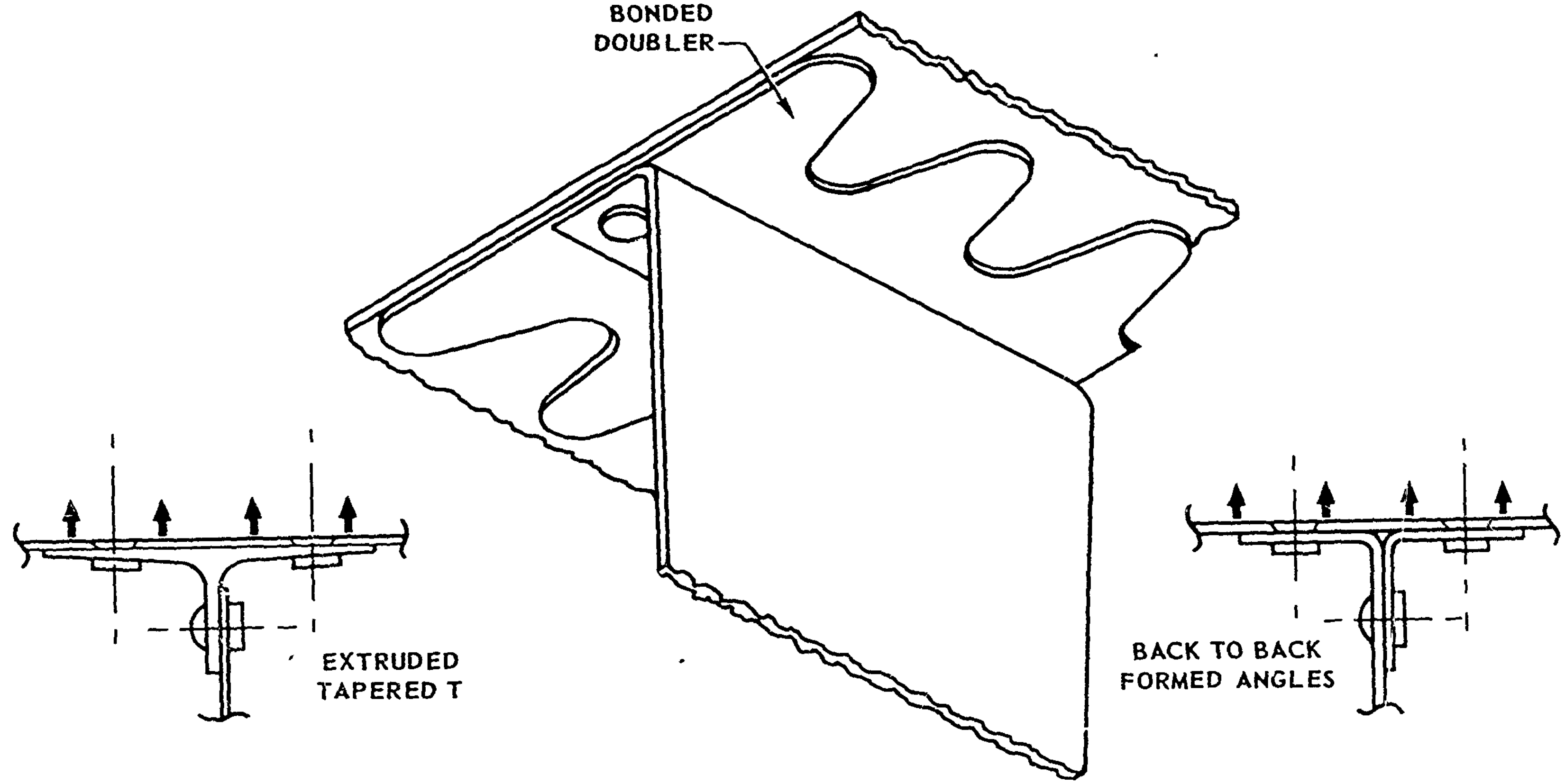


Figure 6

Another important design detail concerns properly transmitting shear loads from vibrating stringers to the supporting ribs or frames. Many failures have occurred in existing aircraft where the shear load is transmitted as a tension load which results in bending of both the stringer and rib flange. Attempts to reinforce the attachment by adding backup angles on the rib flange have generally resulted in failure. In some cases the addition of intermediate frames only resulted in additional failures in the new frames. In this regard, a simple application of equation (1) to the calculation of the shear load at the end of a stringer shows clearly that the load is not a function of the stringer span when the power spectrum is relatively flat. Hence, the proper way to improve the design is to strengthen the attachment by transmitting the shear load through a shear attachment. Figure 7 shows the cruciform clip which was developed for the stringer to rib attachment on DC-8 control surfaces. The lack of eccentricity in the clip permits a smooth direct load transmission path.

Other common design features that should be avoided even at moderate sound levels (overall sound pressure levels of 145 to 150 decibels) are bend relief notches and unsupported web edges, as well as clips and tabs that might be prestressed during assembly.

CONCLUSIONS

The tone of this paper has been pessimistic but at the same time it is believed to be realistic. The underlying principle has been that although the state-of-the-art does not permit reliable analytical results on an absolute basis, the available theories provide tools with which the experienced designer can ratio and extrapolate from the known to the unknown. A rapidly converging combined analytical and experimental design program can thus be accomplished. Small panel tests using siren noise are an important step in the process, but the final design can only be proven with a sonic fatigue proof test of a full scale structure using a jet engine as the noise source. In the final analysis, the quality of the detailed design will mean the difference between success and failure in a sonic environment.

DETAIL DC-8 DESIGN OF STRINGER TO RIB ATTACHMENT

SHEAR LOADS THRU SHEAR CONNECTIONS

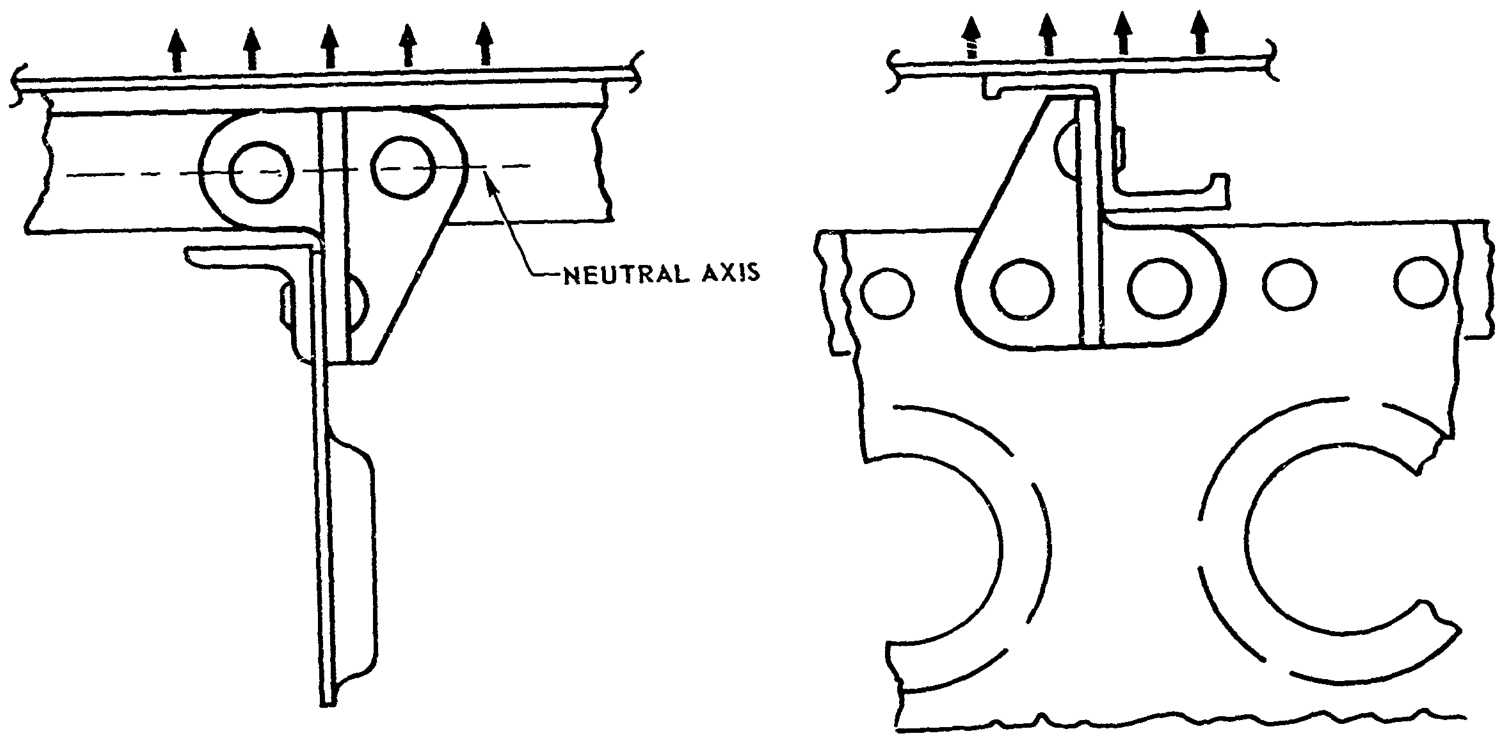


Figure 7

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AN ANALYTICAL METHOD FOR PREDICTING AIRCRAFT FATIGUE LIFE

by

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SUMMARY

The paper contains load information necessary for a fatigue analysis. Repeated loads considered are: gust, maneuver, landing impact, taxi loads and ground-air-ground cycle. A new method of determining a single magnitude of load, which will best represent all load cycles within a given load increment is presented. This load is compared to a S-N curve to obtain fatigue damage. With the exception of a S-N curve for the structure under consideration, this analysis requires data which can be determined during preliminary design such as typical flight mission, wing area, landing gear data, dynamic response and stress analysis data. Since aircraft companies have a tendency to use similar structural design on later designed aircraft, laboratory test of previously designed joints can be used during the preliminary design stages of analysis. Because the mean load level during taxi is different than the mean load level during flight, a new method of establishing a family of S-N curves from one S-N curve and simple specimen data is also presented. An actual case history where fatigue cracking occurred during operation is compared to the analytically calculated life.

I. FATIGUE LOADS

As in any structural problem, a stress analysis cannot be completed without the loads. It is desirable that the calculated values of the repeated load histories be at least as accurate as the present methods of calculating ultimate loads.

1. GUST LOADS

The gust load information given here is a summary of SRG Report 31 "Fatigue Loads Due to Gusts". (1)[a] Numerous articles have been written on gust loads and the effect thereon of various parameters such as speed, altitude, size, etc. The articles generally contain a limited amount of flight data to determine the effect of the particular parameter under investigation. This report contains all available unclassified gust data obtained from VGH records irrespective of parameter investigated and is presented here according to altitude. This method of

a. Numbers in parentheses refer to references at the end of the paper.

presentation gives a much larger sample of flight data and thus reduces the probability of a mistaken conclusion being drawn on the effect of any given parameter.

Gust loads have been used for a number of years to calculate the ultimate design gust factors. The formula generally used is as follows:

$$\text{Eq. 1} \quad \Delta g_{max} = \frac{m \; S \; V_e \; U_e \; K \; \rho_o}{2 \; W}$$

Where: $\Delta g max$ = airplane maximum normal - acceleration increment
m = wing lift - curve slope, per radian
ρ_o = air density at sea level, slugs per cu. ft.
ρ = air density at given altitude, slugs per cu. ft.
S = wing area, sq. ft.
V_e = equivalent airspeed, ft. per sec. $V\sigma^{\frac{1}{2}}$
σ = density ratio = ρ / ρ_o
U_e = "effective" gust velocity, ft. per sec.
K = alleviation factor
W = airplane weight, lb.

The alleviation factor "K" was modified by the various regulating agencies and resulted in some confusion. These agencies agreed to standardize on a new "gust factor, K_g". The new formula will be used in this paper and is as follows:

$$\text{Eq. 2} \quad \Delta g max = \frac{m \; S \; V_e \; U_{de} \; K_g \; \rho_o}{2 \; W} \qquad \text{Ref. (2)}$$

Where: U_{de} = "derived" gust velocity of a single (1-cosine) gust of 25 wing mean chord lengths, ft. per sec. $U\sigma^{\frac{1}{2}}$

U = gust velocity, ft. per sec.
K_g = dimensionless "gust factor" which accounts for the alleviation motion of the airplane and the time lag of the build up of aerodynamic lift.

For speeds below the critical mach number (subsonic):

$$\text{Eq. 3} \quad K_g = \frac{.88 \mu g}{5.3 + \mu g} \qquad \text{Ref. (3)}$$

For speeds above mach 1.4 (supersonic):

$$\text{Eq. 4} \quad K_g = \frac{\mu g^{1.03}}{6.95 + \mu g^{1.03}} \qquad \text{Ref. (3)}$$

and: μg = airplane mass ratio = $\dfrac{2 \; W}{\rho \; m \; g \; c \; S}$

g = acceleration due to gravity, ft. per sec.^{2}

c = mean geometric wing chord, ft.

The supersonic formula, Eq. 4, can be used with caution in the transonic region, but the values of Δg will be lower than actual.

Until a few years ago most of the gust load data came from NASA V-G recorders located near the center of gravity of the airplane. This data was analyzed and plotted as maximum effective gust velocity vs. flight miles. The accuracy of this type of data is questionable because the recorder was not designed for such measurements. The V-G recorder was designed to record an envelope of airplane speed vs. load factor and is used to determine maximum or ultimate loads. A method to obtain frequency from this data was derived and has been used in the past. However, NASA realized that a more accurate method was desirable and subsequently, data was obtained using a VGH recorder. The VGH recorder records time history of airspeed, acceleration, and altitude of the airplane. This data with the characteristics of the airplane, W/S, m, c, determines the gust velocity.

During the early stages of development many factors were considered to effect the number of gusts of a given magnitude, encountered in a given distance. Among these were the wing chord, wing span, wing flexibility, airplane speed and altitude. There are published reports that show all of these variables effect the gust loads. However, in most cases, due to the small sampling size, these results are misleading. From Figure 1 and Table 1, it can be shown that, although the airplanes vary in size from a single-engine fighter type to a four-engine transport type, there is no size effect either wing chord or wing span.

TABLE 1

GUST DATA FOR ALTITUDES OF 5,000 FEET OR LESS

I Single-Engine Jet Fighter of the F-80 type
- ⊙ Based on 904.1 miles of flight at 200 M.P.H.
- ⊘ Based on 522.6 miles of flight at 300 M.P.H.
- ◑ Based on 522.9 miles of flight at 450 M.P.H.
- ● Based on 371.5 miles of flight at 500 M.P.H.

II Single-Engine Swept-Wing Jet Fighter of the F-86 type
- ⊡ Based on 517.4 miles of flight at 300 M.P.H.
- ⊠ Based on 523.7 miles of flight at 450 M.P.H.
- ■ Based on 258 miles of flight at 600 M.P.H.

III Twin-Engine Transport Airplane
Design gross wt. 39,900#; wing span 93'; design speed 256 M.P.H.
- ▽ Based on 77,600 miles at all typical flight speeds.

TABLE 1 (CONTD)

GUST DATA FOR ALTITUDES OF 5,000 FEET OR LESS

IV Twin-Engine Transport Airplane
Design gross wt. 40,500#; wing span 92'; design speed 280 M.P.H.
△ Based on 59,800 miles at all typical flight speeds.

V Four-Engine Transport Airplane
Design gross wt. 107,000#; wing span 123'; design speed 271 M.P.H.
◿ Based on 48,000 miles at all typical flight speeds.

VI Four-Engine Transport Airplane
Design gross wt. 91,550#; wing span 117.5'; design speed 300 M.P.H.
◺ Based on 51,000 miles at all typical flight speeds.

Also, from Figure I, it can be seen that both fighter gust load data points are practically on top of each other, even though one has straight-wing while the other has a swept wing. True there is some scatter in the data, but this is less than by including the so called "corrections" for size and speed. In addition, this simplifies the calculations. It is the opinion of the author that altitude is the only parameter which is not weighed properly in Eqs. 2, 3, and 4, so that the gust load history could be described by a single curve.

The flight data were plotted according to altitude, and curves were drawn through the mean values. These curves are shown on Figure 2 as solid lines. The long dash lines which extend from the solid lines are extrapolated values. The short dash lines are from reference (4), which is now considered by the author to be obsolete because the data in this reference were not direct measurements, but were largely derived from indirect measurements (such as the airspeed fluctuations and turbulence telemeter data) as described on pages 3, 4, 5, and 7 of reference (4).

2. MANEUVER LOADS

The maneuvering load investigations are not as far advanced as are the investigations of gust loads. NASA TN 3086 gives maneuvering load data from five types (two two-engine, three four-engine) commercial transports recorded by means of VGH recorders. A comparison of the VGH records showed that maneuvers performed during airplane or pilot check flights were quite different in both magnitude and frequency of occurrence from maneuvers performed during routine operational flights. Figure 3 is a comparison of acceleration increments caused by operational maneuvers. Figure 4 is a similar comparison of check-flight maneuvers. As shown in TN 3086 positive and negative distributions caused by operational maneuvers are essentially symmetrical, while the check-flight maneuvers cause a greater number of positive accelerations than negative. The scatter in check-flight data is caused to a large extent by the difference between air-line and pilot practice rather than any aircraft parameter. Hence, it would be wise to

average the check-flight data unless it was known which airline would be using the aircraft. To simplify the calculations, the positive and negative number of accelerations have been averaged for Figures 3 and 4. Hence, these can be considered as the number of load reversals about 1 g load. Figure 4 is plotted using total flight miles rather than flight miles spent in check flight operation. Thus for a given number of flight miles the loading due to check flight can be read direct from Figure 4.

3. GROUND LOADS

a. LANDING IMPACT

Figure 5 shows the probability of equaling or exceeding a given value of vertical velocity caused by landing impact. (6) The generalized curves in NACA Report 1154 provide an analytical method for converting the contact vertical velocity to maximum center of gravity acceleration of the upper-mass. A brief summary of that method is as follows:

Eq. 5 $$U_o' = V_v \left(\frac{A^2 g}{W_1 a}\right)^{\frac{1}{2}}$$

Where: $$A = \frac{A_h{}^3 \rho}{2 (C_d A_n)^2 \cos \zeta}$$

V_v = Vertical velocity
ρ = Mass density of hydraulic fluid
A_h = Hydraulic area
C_d = Orifice discharge coefficient
A_n = Net Orifice area
ζ = Angle between shock-strut axis and vertical
g = Gravity
W_1 = Weight of upper mass on one strut
a = Slope of linear approximation to tire force-deflection characteristics
U_o' = Contact dimensionless velocity

Knowing the landing gear parameters, use Eq. 5 to determine the dimensionless velocity at contact with the ground by obtaining vertical velocity from Figure 5. With the dimensionless velocity thus obtained, determine maximum dimensionless upper-mass acceleration, U_1'' by use of Figure 6. Substituting U_1'' into Eq. 6 the vertical acceleration, Va, is obtained.

Eq. 6 $$V_a = U_1''\left(\frac{a}{A}\right)$$

To get V_a in terms of Δg, divide by g.

Eq. 7 $\Delta g = \frac{V_a}{g} \quad U''_1 \left(\frac{a}{Ag}\right)$

b. TAXI LOADS

Figure 7 gives the taxi load probability distributions for 1000 flights. (6) This curve is based on a detailed analysis of all the loads experienced in taxiing for four airplane types and the scatter in the load level at a given probability was small. Thus it may be assumed that the combined distribution can be used to represent all taxi operations.

c. GROUND-AIR-GROUND CYCLE

This loading consists of transfer of airplane weight from the landing gear to the wing and back again once a flight. Because the load is of large magnitude it can be significant.

d. GROUND HANDLING

Braking, turning, towing, etc., are important from the viewpoint of fatigue of the landing gear; however, these conditions have little effect on the aircraft structure as a whole, and is therefore not discussed further in this paper. NASA TM 1422 is good reference on fatigue failures in landing gears.

II FATIGUE ANALYSIS EXAMPLE

The following example is an actual case where fatigue cracking occurred during flight operation of the aircraft. This actual case was used so that a comparison of the actual to the analytically calculated life could be made.

1. REQUIRED DATA FOR ANALYSIS

a. The laboratory fatigue test S-N curve for the wing joint considered, Figure 8.

b. Typical flight mission plot of airspeed, weight, and altitude vs. distance, Figure 9.

c. Wing Data: Wing area = 1463 $ft.^2$; wing chord = 13.64 ft.; slope of lift curve = 4.699 units/rad.

d. Landing Gear Data: Density of hydraulic fluid = 54.1 lbs./$ft.^3$; hydraulic area = .1704 $ft.^2$; orifice discharge coefficient = .96; angle between shock-strut axis and vertical at time of initial ground contact for a typical landing = 0; net orifice area = .00278 $ft.^2$; slope of linear approximation to tire force-deflection curve = 219,000 lbs./ft.

TABLE 2

1 Alt.	2 Wt.	3 Vel.	4 Miles	5 V_v	6 μ_g	7 K_g	8 U_{de}=.2g	9 U_{de}=.3g	10 U_{de}=.4g	11 U_{de}=.5g	12 U_{de}=.6g	13 U_{de}=.7g
0-5	94.0	160	35	260	28.6	.742	11.9	17.9	23.8	29.8	35.8	41.7
5-10	93.1	175	35	264	32.9	.756	11.2	16.9	22.5	28.2	33.8	39.4
10-15	92.2	183	35	255	38.2	.772	11.5	17.2	22.9	28.7	34.4	40.2
15-20	89.8	309	386	396	43.7	.785	7.1	10.6	14.1	17.7	21.2	24.7
10-15	88.2	324	23	451	36.6	.769	6.2	9.35	12.5	15.6	18.7	21.8
5-10	88.1	290	23	437	31.2	.753	6.6	9.84	13.1	16.4	19.7	22.9
0-5	88.0	258	23	419	26.7	.735	7.0	10.5	14.0	17.5	21.0	24.5

Δg	0-5 Alt.			5-10 Alt.			10-15 Alt.			15-20 Alt.		
.2	4.2	8.3333		5.2	6.7508		170	.2059		80	4.8250	
			7.9649			6.6179			.1657			3.9273
.3	95	.3684		310	.1129		870	.0402		430	.8977	
			.3312			.0860			.0285			.6008
.4	940	.0372		1,300	.0269		3,000	.0117		1,300	.2969	
			.0284			.0199			.0082			.1763
.5	4,000	.0088		5,000	.0070		10,000	.0035		3,200	.1206	
			.0061			.0047			.0023			.0655
.6	13,000	.0027		15,000	.0023		30,000	.0012		7,000	.0551	
			.0017			.0016			.008			.0310
.7	36,500	.0010		50,000	.0007		85,000	.0004		16,000	.0241	
			.0010			.0007			.0004			.0241

Δg	10-15 Alt.			5-10 Alt.			0-5 Alt.			Total Cycles
.2	17	1.3529		.70	3.2857		.45	51.1111		
			1.0791			2.6092			42.7475	65.1116
.3	84	.2738		34	.6765		2.75	8.3636		
			.1853			.4465			6.8303	8.5086
.4	260	.0885		100	.2300		15	1.5333		
			.0502			.1448			1.2493	1.6771
.5	600	.0383		270	.0852		81	.2840		
			.0199			.0487			.2201	.3673
.6	1,250	.0184		630	.0365		360	.0639		
			.0092			.0201			.0439	.1083
.7	2,500	.0092		1,400	.0164		1,150	.0200		
			.0092			.0164			.0200	.0718

e. Stress Analysis Data: 1g stress for joint = 13,000 p.s.i. Stress in joint with aircraft fully loaded on ground = -5,500 p.s.i. Ultimate allowable stress = 57,000 p.s.i.

f. Dynamic response approximately equal to 1.0 and can be omitted.

2. CALCULATION OF REPEATED LOADS

a. GUST LOADS

Eq. 2 $$\Delta g_{max} = \frac{(4.699)\ (.002377)\ (1463)\ V_e\ U_{de}\ K_g}{2\ W}$$

Eq. 3 $$K_g \quad \frac{.88\mu_g}{5.3+\mu_g} \qquad \mu_g = \frac{2\ W}{(4.699)\ (32.17)\ (13.64)\ (1463)\ \rho}$$

$$\mu_g = \frac{W}{\rho}(.672 \times 10^{-6})$$

From 0-5000 ft. altitude, use 2,500 ft. $\rho = .00221, \sigma^{\frac{1}{2}} = .9638$

From Figure 9 W = 94,000 lbs.; V = 160 knots

$V_e = 160\ (1.15)\ (1.465)\sigma^{\frac{1}{2}} = 260$ ft./sec.

$$\mu_g = \frac{94,000}{.00221}(.672 \times 10^{-6}) = 28.6 \qquad K_g \ \frac{.88\ (28.6)}{33.9} = .742$$

$$\Delta g_{max} = \frac{8.17\ V_e\ U_{de}\ K_g}{W} \quad \frac{(8.17)\ (260)\ U_{de}\ (.742)}{94,000} = .0168\ U_{de}$$

For $\Delta g_{max} = .3\ U_{de}$ $\quad .3/.0168 = 17.9$

In a similar manner, the rest of columns 2 through 13 are determined. Using Figure 2 and U_{de} obtained in columns 8 through 13, record the miles required to encounter gust velocities greater than U_{de}. Thus to determine an acceleration increment of $.3 < \Delta g < .4$ at altitudes between 0 and 5,000 feet during take-off U_{de} equals 17.9 and 23.8. Miles required to exceed these values of U_{de} are 95 and 940 from Figure 2. This means that for every 95 miles of flight an acceleration increment of greater than .3g is expected. From Column 4, the flight distance for this altitude range is 35 miles. Hence 35/95 .3684 cycles greater than .3g every 35 miles. Also 35/940 .0372 cycles greater than .4g. The number cycles between .3g and .4g is .3684 -.0372. Finally, the cycles for all altitude ranges for the same acceleration increment are summed.

b. MANEUVER LOADS

Maneuvering load cycles per mile exceeding a given value of Δg are obtained from Figure 3 and entered in Column 2 of Table 3. To determine Column 3, start with the largest value of Δg and subtract the load cycles per mile exceeding g from the next largest Δg. Column 3 now gives in this particular case the load cycles per mile between .3g to .4g; .4g to .5g and greater than .5g. In a

similar manner, Column 4 and 5 are determined with the use of Figure 4. To obtain the load cycles per flight, multiply Column 3 and 5 by the miles per flight which is 560. These values are entered in Columns 3 and 4 of Table 6.

TABLE 3

MANEUVERING LOADS

	Routine Maneuvers		Check Flight Maneuvers	
1	2	3	4	5
Δg	Cycles/Mile Exceeding Δg	Cycles/Mile Increment Δg	Cycles/Mile Exceeding Δg	Cycles/Mile Increment Δg
.2	.00550	.00488	.00100	.000319
.3	.000620	.000532	.000681	.000257
.4	.0000877	.0000682	.000424	.000186
.5	.0000195	.0000195	.000238	.000100
.6			.000138	.0000622
.7			.0000755	.0000356
.8			.0000399	.0000267
.9			.0000132	.0000090
1.0			.00000420	.00000309
1.1			.00000111	.00000111

c. LANDING LOADS

Solving for $A = \dfrac{(54.1)(.1703)}{2(.96)(.002377)^2\ (1)} = 1.88 \times 10^4$

From Figure 9 landing gross weight = 88,000 lbs. Thus for two main landing gears $W_1 = 44,000$ lbs.

Eq. 5 $U_o' = V_v \dfrac{(1.88 \times 10^4)^2\ (32.17)}{(414 \times 10^4)\ (2.19 \times 10^5)} = 1.19\ V_v$

Eq. 7 $\Delta g = U_1'' \dfrac{2.19 \times 10^5}{(1.88 \times 10^4)\ (32.17)} = .362\ U_1''$

For $\Delta g > .3$ $U_1'' = .3/.362 = .83$

From Figure 6 for $U_1'' = .83$ $U_o' = 1.65$

$V_v = 1.65/1.19 = 1.39$

From Figure 5 for $V_v = 1.39$ $n = .74$

This method is used to determine the other values of Table 4.

TABLE 4

LANDING LOADS
(Per Landing)

1	2	3	4	5	6
Δg	U_1''	U_o'	V_v	n (cycles) of Load > Δg	n for Δg Increments
.3	.83	1.65	1.39	.74	.11
.4	1.10	2.00	1.68	.63	.14
.5	1.38	2.43	2.04	.49	.13
.6	1.66	2.80	2.35	.36	.13
.7	1.93	3.18	2.67	.23	.090
.8	2.21	3.55	2.98	.14	.072
.9	2.49	3.93	3.30	.068	.035
1.0	2.76	4.25	3.47	.033	.021
1.1	3.04	4.63	3.89	.012	.0076
1.2	3.31	4.95	4.16	.0044	.0033
1.3	3.59	5.30	4.45	.0011	.0011

d. TAXI LOADS

Taxi loads are calculated in Table 5 using Figure 7 and the same procedure for maneuvering loads.

TABLE 5

TAXI LOADS
(Per Landing)

1	2	3
Δg	Cycles of Load > Δg	Cycles for Δg Increments
.3	3.30	3.115
.4	.185	.1765
.5	.0085	.0085

e. GROUND-AIR-GROUND CYCLE

Stress range for ground-air-ground cycles 13,000 - (-5,500)=18,500 p.s.i. Alternating Stress = 18,500/2 = 9,250 p.s.i. Mean Stress = (13,000 - 5,500)/2 = 3,250 p.s.i.

3. $\overline{\Delta g}$ FOR THE AVERAGE NUMBER OF CYCLES

In the previous section the number of cycles within a given load increment have been determined. To calculate the fatigue damage caused by these load

cycles it is necessary to determine a single value of $\overline{\Delta g}$ which will best represent all load cycles within a given load increment. Consider the load increment between .4 and .5 Δg for cycles caused by gust, maneuver and check flight. The average load, .45 Δg should not be used because more cycles are applied within the lower half of this load increment than the higher half. What should be used is the value of Δg which will divide the cycles in half so that as many cycles occur at load levels below $\overline{\Delta g}$ as occur above. To calculate this value of $\overline{\Delta g}$ it is necessary to know the variation of the frequency of cycles within the load increment. As an approximation to this variation the number of cycles within the load increments below, above and the increment in question will be used. The Lagrange Interpolation Formula will be used to pass a curve through these three points; 8.9452, 1.8195, .4342.

$$y = C_1\,(8.9452)\,(x-1)\,(x-2) + C_2\,(1.8195)\,(x-0)\,(x-2) + C_3\,(.4342)(x-0)(x-1)$$

To evaluate C_1, let $x = x_1$ (i.e, o)

$$y_1 = 8.9452 = C_1\,(8.9452)\,(-1)\,(-2) \quad \therefore \quad C_1 = .50$$

$$y_2 = 1.8195 = C_2\,(1.8195)\,(1)\,(-1) \quad \therefore \quad C_2 = -1$$

$$y_3 = .4342 = C_3\,(.4342)\,(2)\,(1) \quad \therefore \quad C_3 = .50$$

$$y = 4.4726(x^2 - 3x+2) - 1.8195\,(x^2 - 2x) + .2171\,(x^2 - x)$$

Integrating and evaluating constant gives:

$$Y = 4.4726\left(\frac{x^3}{3} - \frac{3x^2}{2} + 2x\right) - 1.8195\left(\frac{x^3}{3} - x^2\right) + .2171\left(\frac{x^3}{3} - \frac{x^2}{2}\right) + 8.9452$$

Using the middle one third of the curve which is from .5 to 1.5 and solving for $\overline{x}$ which will divide the area in half.

$$\left[4.4726\left(\frac{x^3}{3} - \frac{3x^2}{2} + 2x\right) - 1.8195\left(\frac{x^3}{3} - x^2\right) + .2171\left(\frac{x^3}{3} - \frac{x^2}{2}\right) + 8.9452\right]_{x=.5}^{x=\overline{x}} =$$

$$\left[4.4726\left(\frac{x^3}{3} - \frac{3x^2}{2} + 2x\right) - 1.8195\left(\frac{x^3}{3} - x^2\right) + .2171\left(\frac{x^3}{3} - \frac{x^2}{2}\right) + 8.9452\right]_{x=\overline{x}}^{x=1.5}$$

Solving for $\overline{x}$ gives .77. Since $\overline{x} = .5$ corresponds to .4 Δg and $\overline{x} = 1.5$ corresponds to .5 Δg thus for $\overline{x} = .77$, $\overline{\Delta g} = .427$.

4. EFFECT OF MEAN STRESS ON ALTERNATING STRESS

Because of the expense and time involved in constructing and fatigue testing wing joints, usually only one S-N curve of a given mean stress is obtained. Since gust, maneuver and check flight loads all fluctuate about a one G load condition, this is the mean load or mean stress generally used. The problem is to extrapolate this single S-N curve to a family of S-N curves with mean stresses corresponding to the mean stresses shown in Figure 10 for taxi, landing impact and ground-air-ground loading conditions.

The tension surface of the wing joint is .064 inch thick clad 7075-T6 so that the fatigue properties of notched specimen made from sheet 7075-T6 can be used to estimate the fatigue life of the wing joint. To take into consideration such factors as secondary bending stresses, which are difficult to calculate in complex built up structure, mean stress and alternating stress will be plotted as a per cent of ultimate tensile stress. There are four variables which are necessary for the extrapolation: mean stress, alternating stress, fatigue life, and stress concentration factor. To be able to plot this data will require a three dimensional plot or holding one of the variables constant. With fatigue life held constant a family of curves as shown in Figure 11 can be plotted. The solid lines are from S-N curves of notched 7075-T6 sheet specimens and the single data points are from the S-N curve of the wing joint. The dashed line is an interpolation between the stress concentration factors and is used to establish allowable fatigue cycle curves shown in Figure 12. Note that for all fatigue lifes the wing joint has the same stress concentration factor. This indicates that, for one G mean load at least, the S-N curve of the wing joint has the same shape as the S-N curve of simple specimen with the same stress concentration factor.

5. FATIGUE DAMAGE

To calculate fatigue damage, the value of $\overline{\Delta g}$ is used with Figure 12 to obtain an allowable number of cycles, N. Miner's cumulative damage theory is then employed to calculate the life of the wing joint. The cumulative damage theory was checked by N.A.S.A with fatigue tests on C-46 aircraft wings. The results of the constant amplitude tests are given in N.A.S.A. TN's 2920, 3190 and 3847. These results were used to establish the S-N curve for the C-46 wing. Tests were then conducted using a spectrum of loads derived from gust-frequency. The results are given in N.A.S.A TN 4132 and the following information is from that TN. The first cracks to appear were predicted reasonably well by the Linear-Cumulative-Damage theory. (Actual value of $\Sigma n/N = 1.45$) Since the example considered is life to cracking the fatigue life is also calculated using the 1.45 factor.

TABLE 6

$\overline{\Delta g}$	Gust n	Man n.	Check n	Σ n	N	$(n/N)10^{-3}$
.220	65.1116	2.7328	.1786	68.0220	∞	0
.321	8.5086	.2927	.1439	8.9452	80,000	.11182
.427	1.6771	.0382	.1042	1.8195	34,000	.05352
.530	.3673	.0109	.0560	.4342	19,000	.02285
.636	.1083		.0348	.1431	11,000	.01301
.720	.0718		.0199	.0917	7,600	.01207
.824			.0150	.0159	5,000	.00300
.922			.00186	.00186	3,650	.00051
1.046			.00186	.00186	2,450	.00076
1.150			.00062	.00062	1,750	.00056
					$\Sigma n/N$ =	.21810

$\overline{\Delta g}$	Land n	N	$(n/N)10^{-3}$
.250	.270	∞	0
.341	.220	300,000	.00073
.439	.107	110,000	.00097
.531	.0286	66,000	.00043
.650	.0044	42,000	.00010
		$\Sigma n/N$ =	.00223

$\overline{\Delta g}$	Taxi n	N	$(n/N)10^{-3}$
.314	3.115	∞	0
.413	.1765	∞	0
.550	.0085	33,000	.00026

Δg	G-A-G n	N	$(n/N)x10^{-3}$
.712	1	35,000	.02857

Gust, maneuvering and damage check flight $= .21810 \times 10^{-3}$

Ground-air-ground cycle $= .02857 \times 10^{-3}$

Landing impact $= .00223 \times 10^{-3}$

Taxi $= .00026 \times 10^{-3}$

Total fatigue damage per flight $= .24916 \times 10^{-3}$

$$\frac{\text{Total allowable fatigue damage}}{\text{Total fatigue damage per flight}} = \frac{1.0 \times 10^{-3}}{.24916} = 4,013 \text{ Flights}$$

(Flights) (Hours/flight) = 4,013 (2.1) = 8,427 hours calculated fatigue life
N.A.S.A. factor 1.45 (8,427) = 12,219 hours

The summary of actual service difficulties by the airlines report fatigue cracking for the joint as occurring between 10,000 and 12,200 hours.

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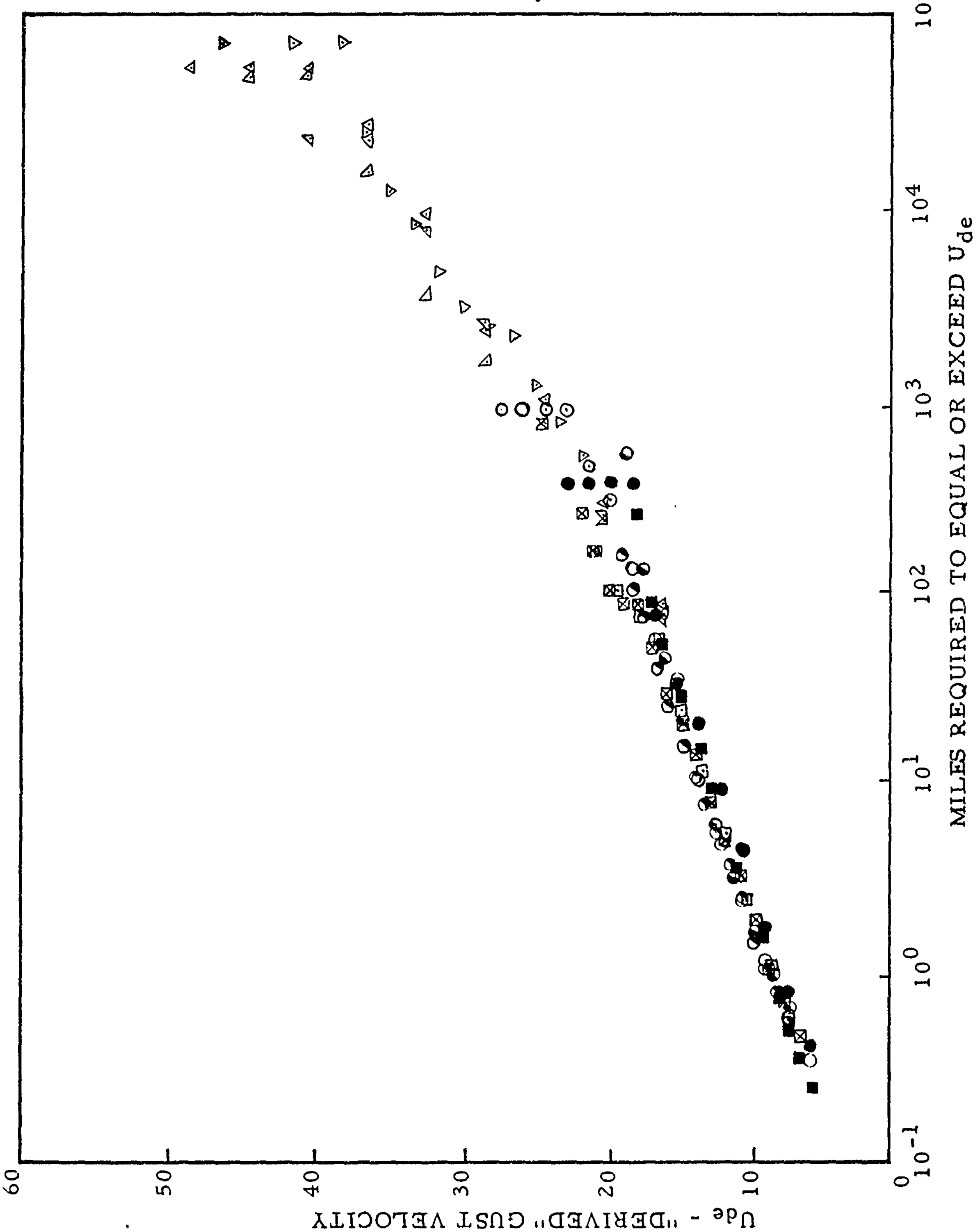
FIG. 1
GUST LOADS
ALTITUDE 0-5,000 FEET



GUST LOADS ALTITUDE 0- 55,000 FEET

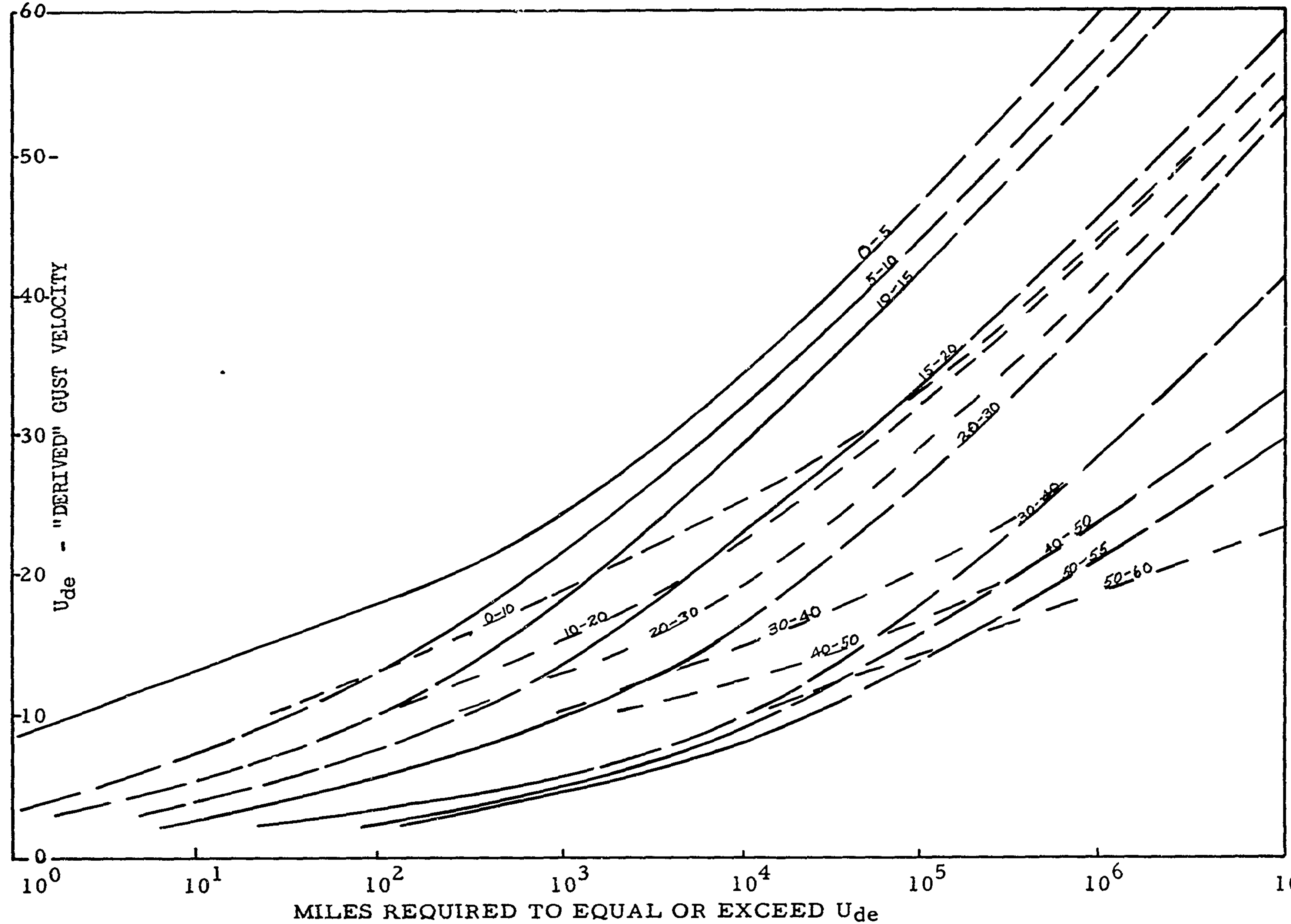


FIG. 2

FIG. 3

ROUTINE MANEUVER LOADS

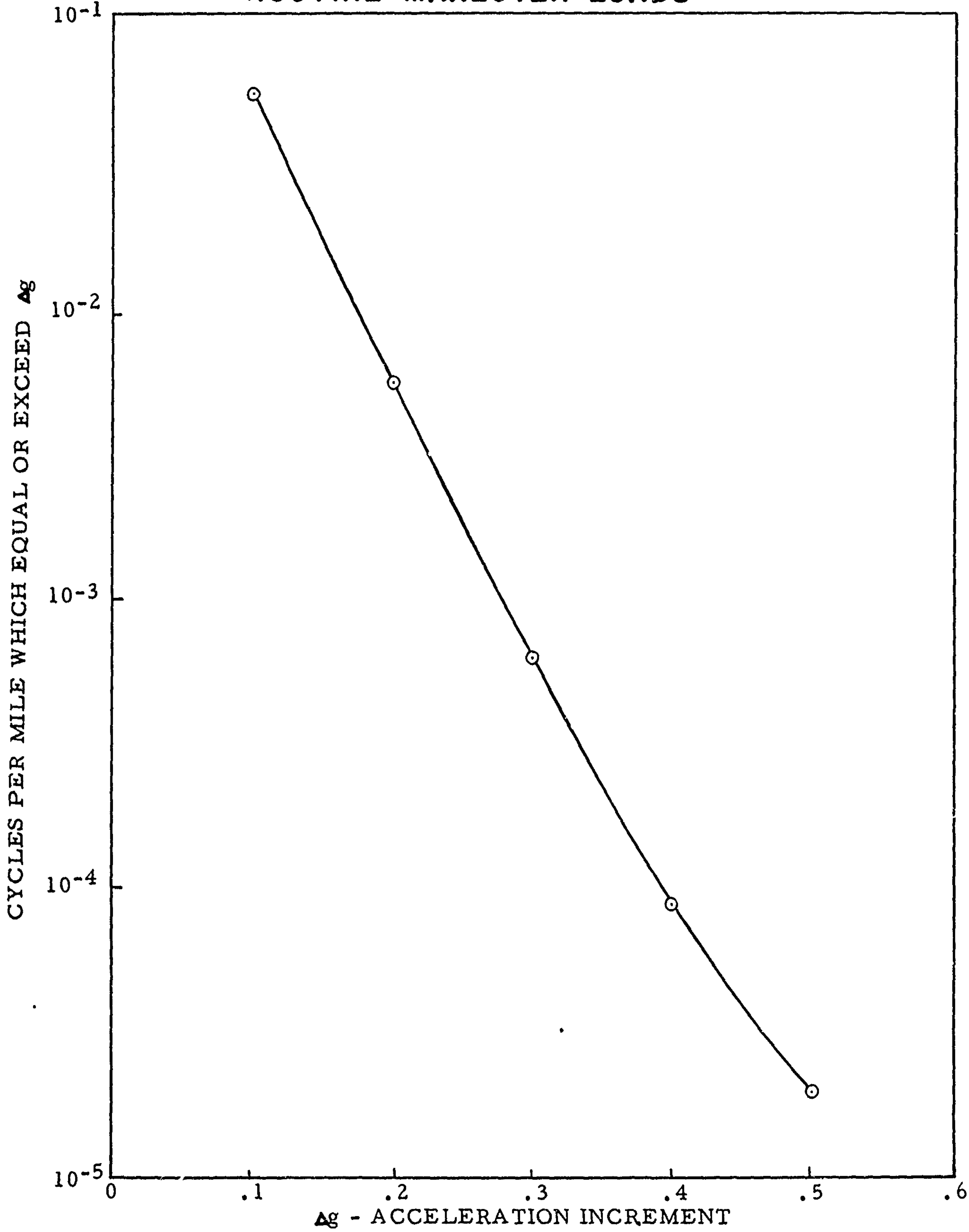


FIG. 4
CHECK FLIGHT LOADS

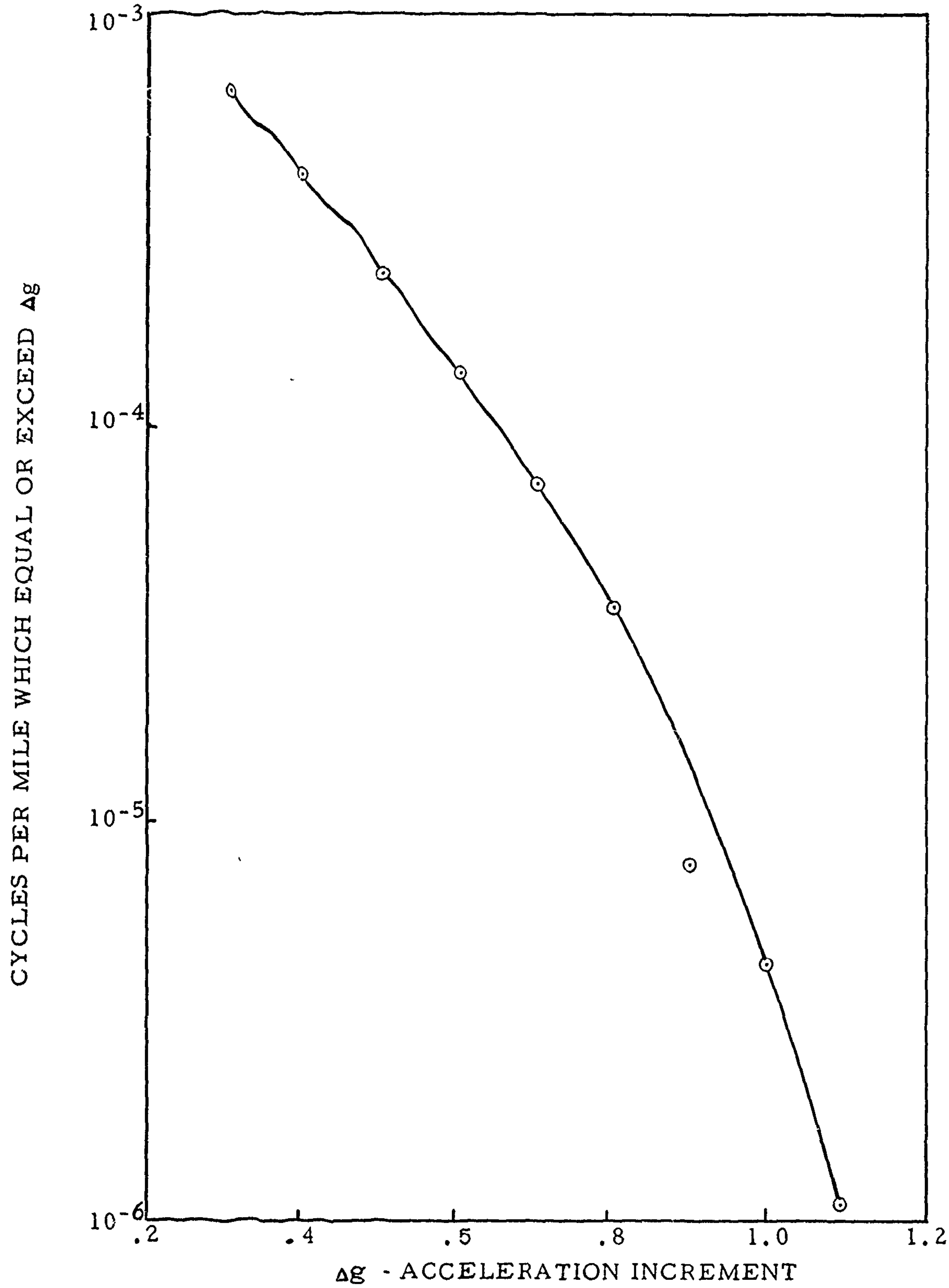


FIG. 5

LANDING VERTICAL VELOCITY

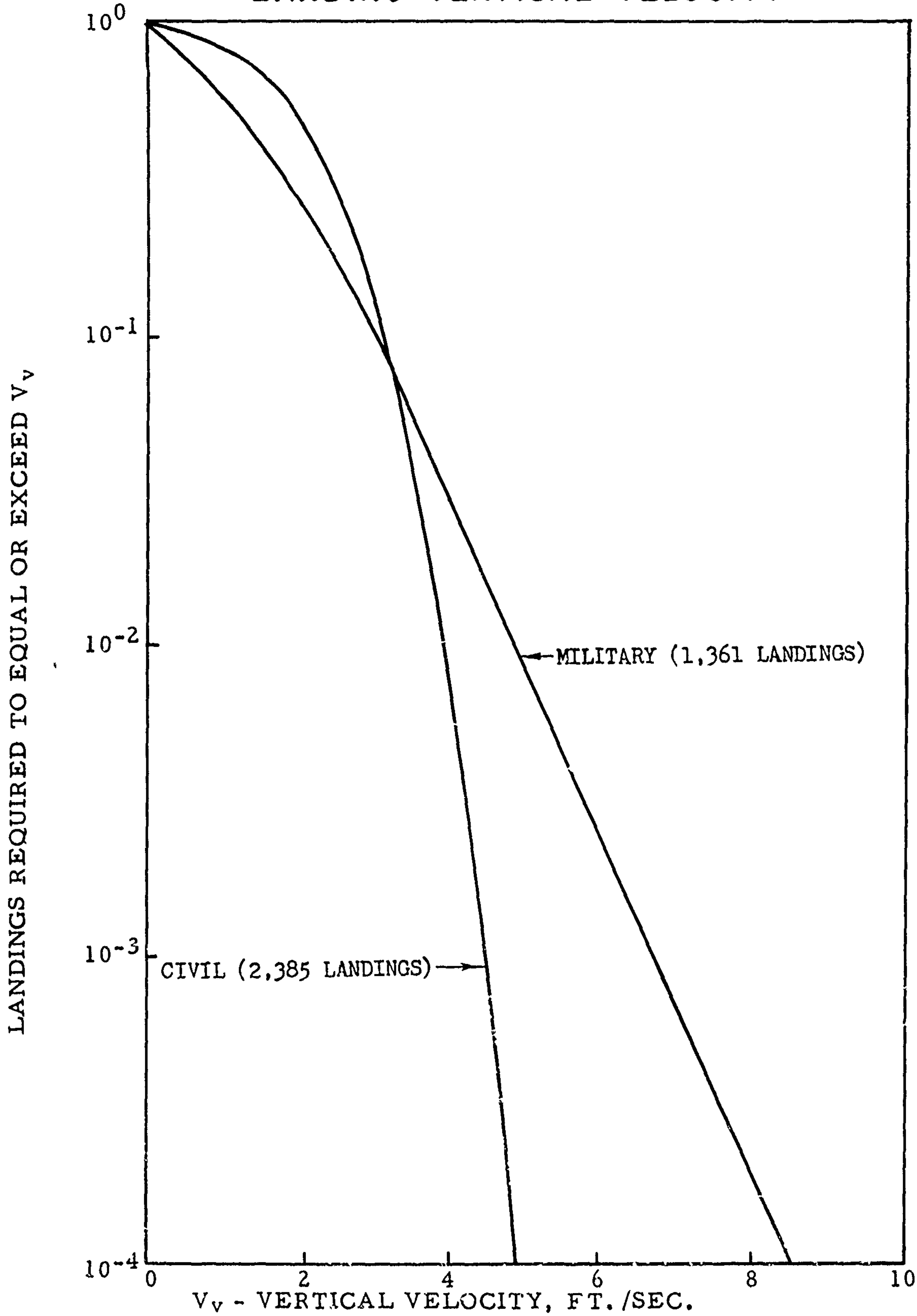


FIG. 6
LANDING ACCELERATION

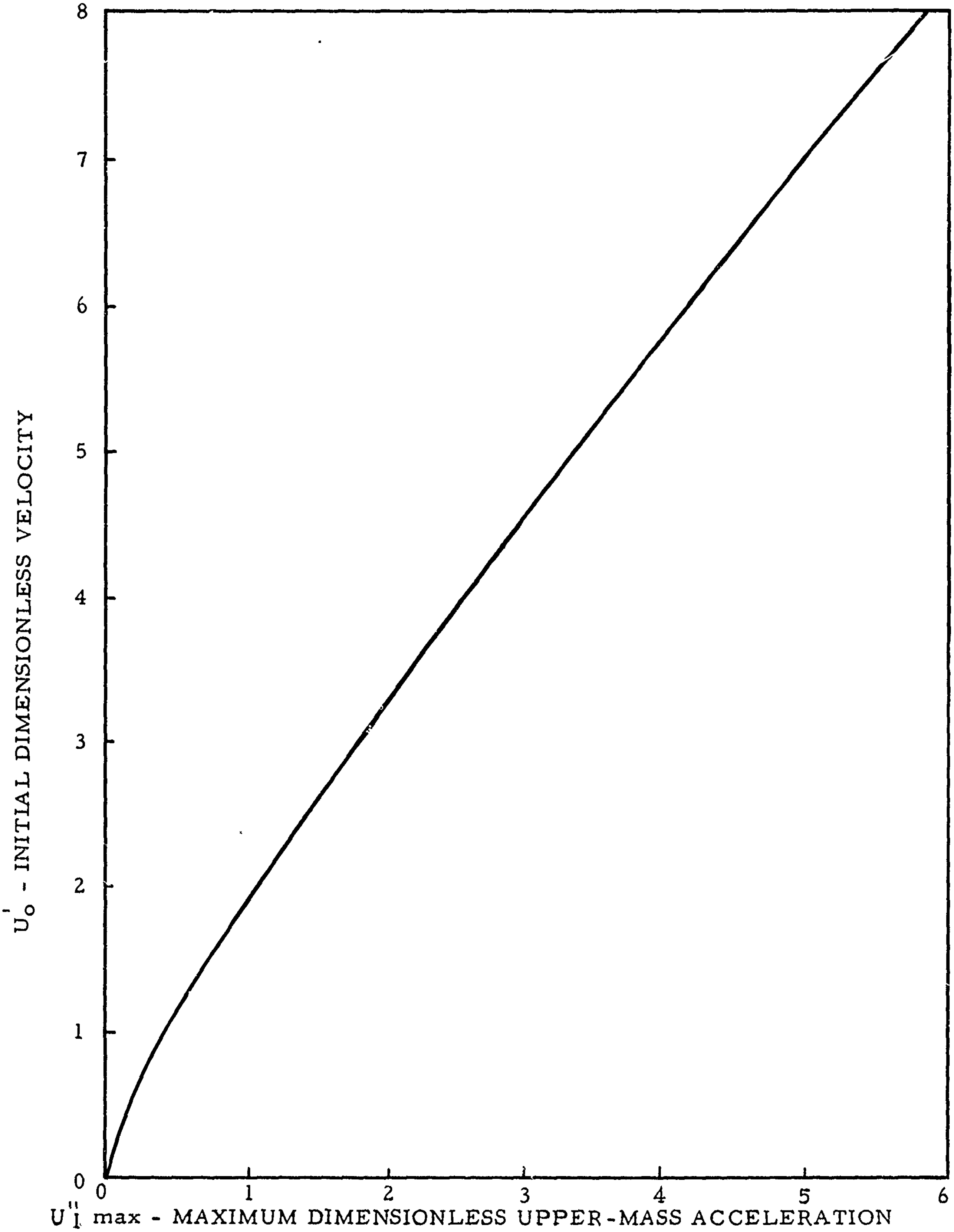


FIG. 7

TAXI LOADS PER 1000 FLIGHTS

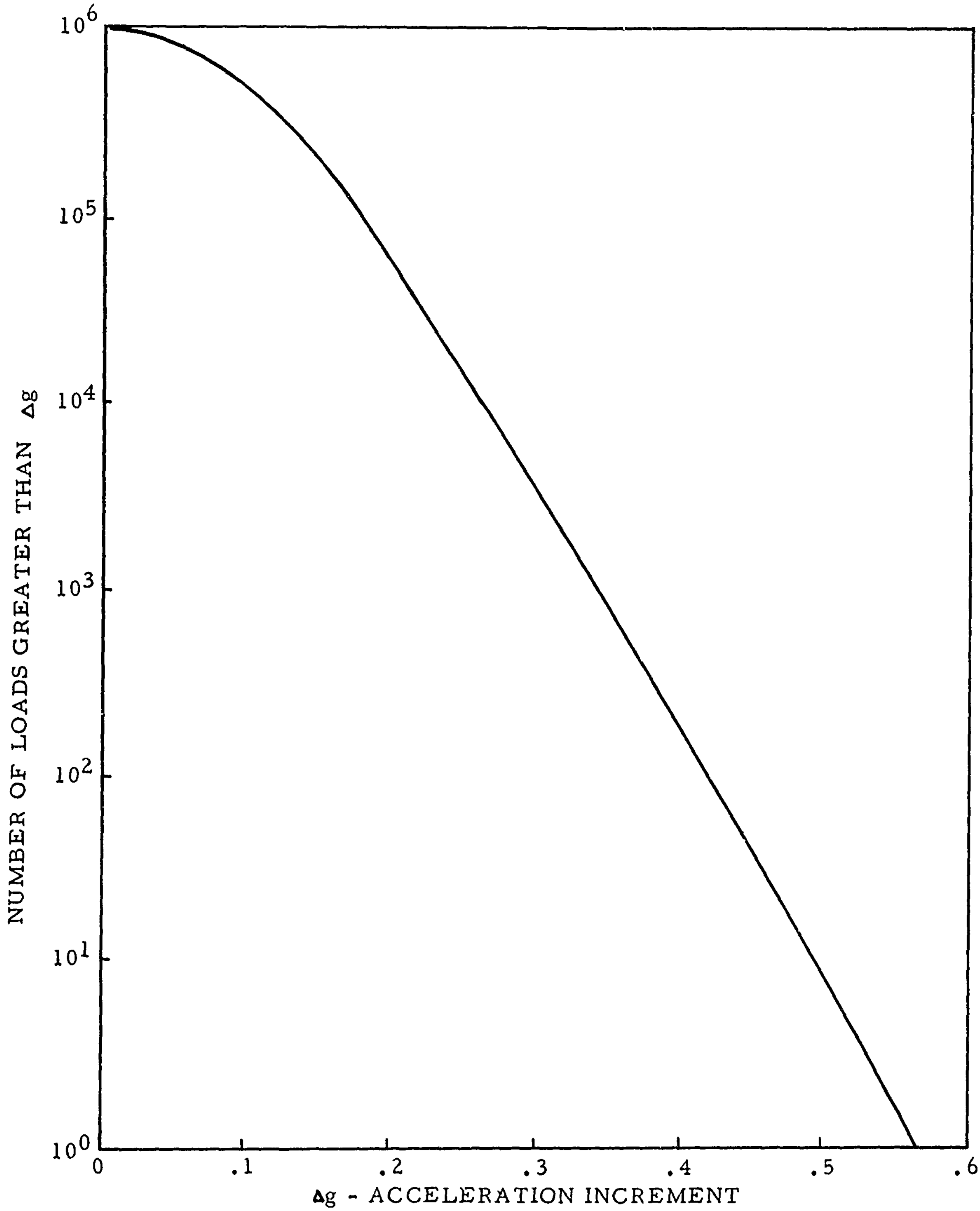


FIG. 8
LABORATORY S-N CURVE

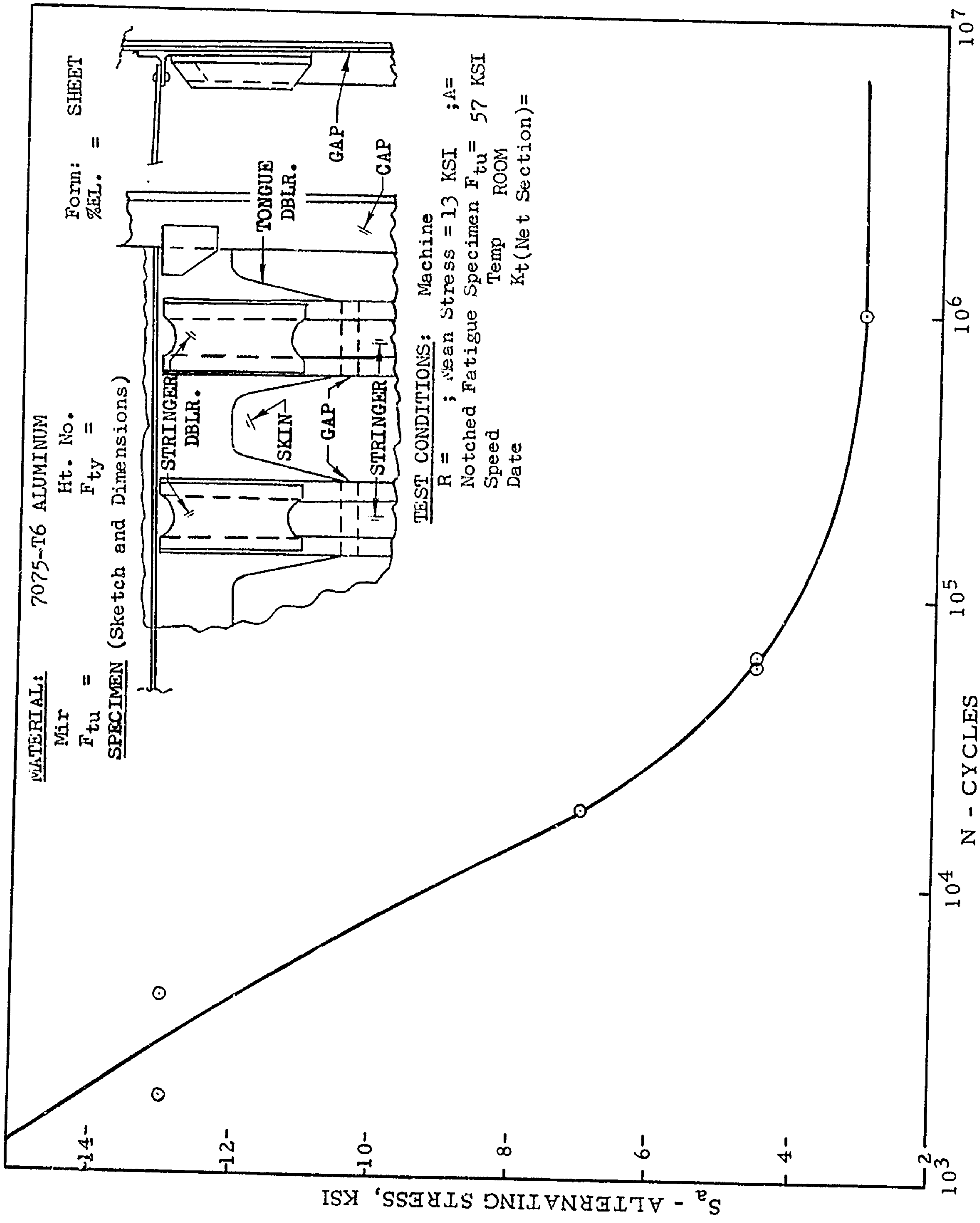


FIG. 9
TYPICAL FLIGHT MISSION

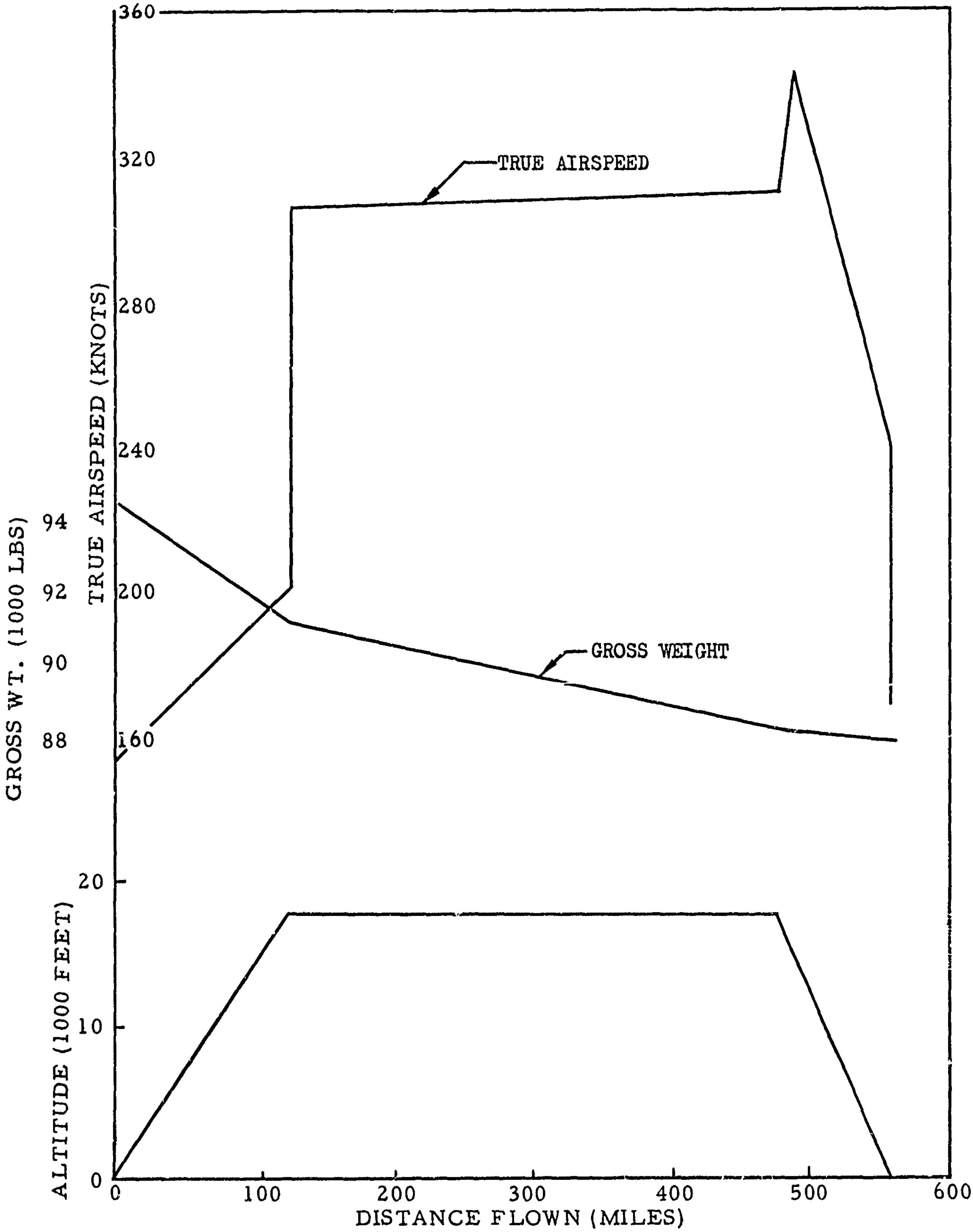
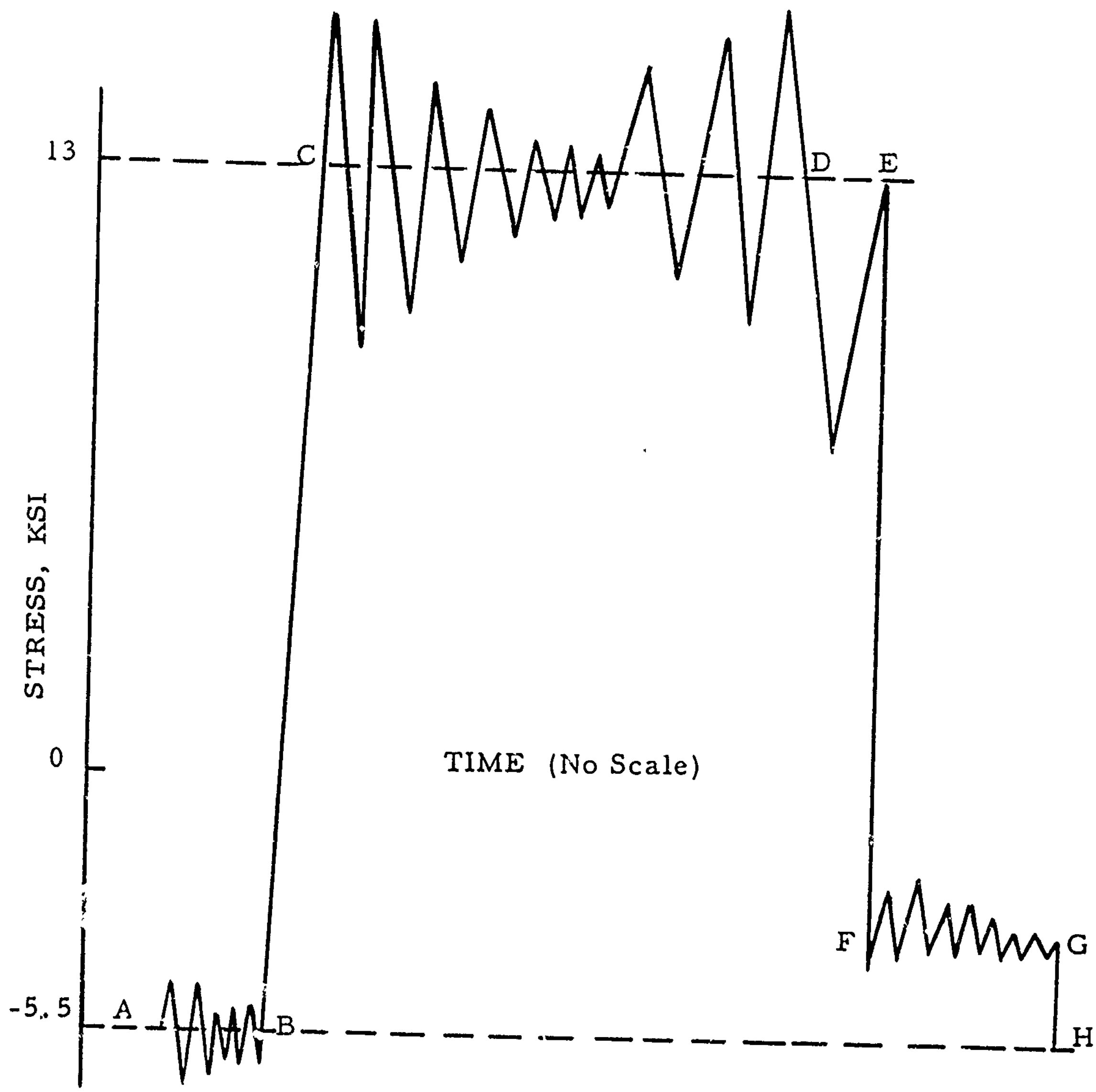


FIG. 10

LOAD CYCLE DIVISION



A - B & F - G Taxi Loads
B - C, E - F & G - H Ground-Air-Ground Cycle
C - D Gust, Maneuver & Check Flight Loads
D - E Landing Impact

FIG. II

INTERPOLATION CURVES

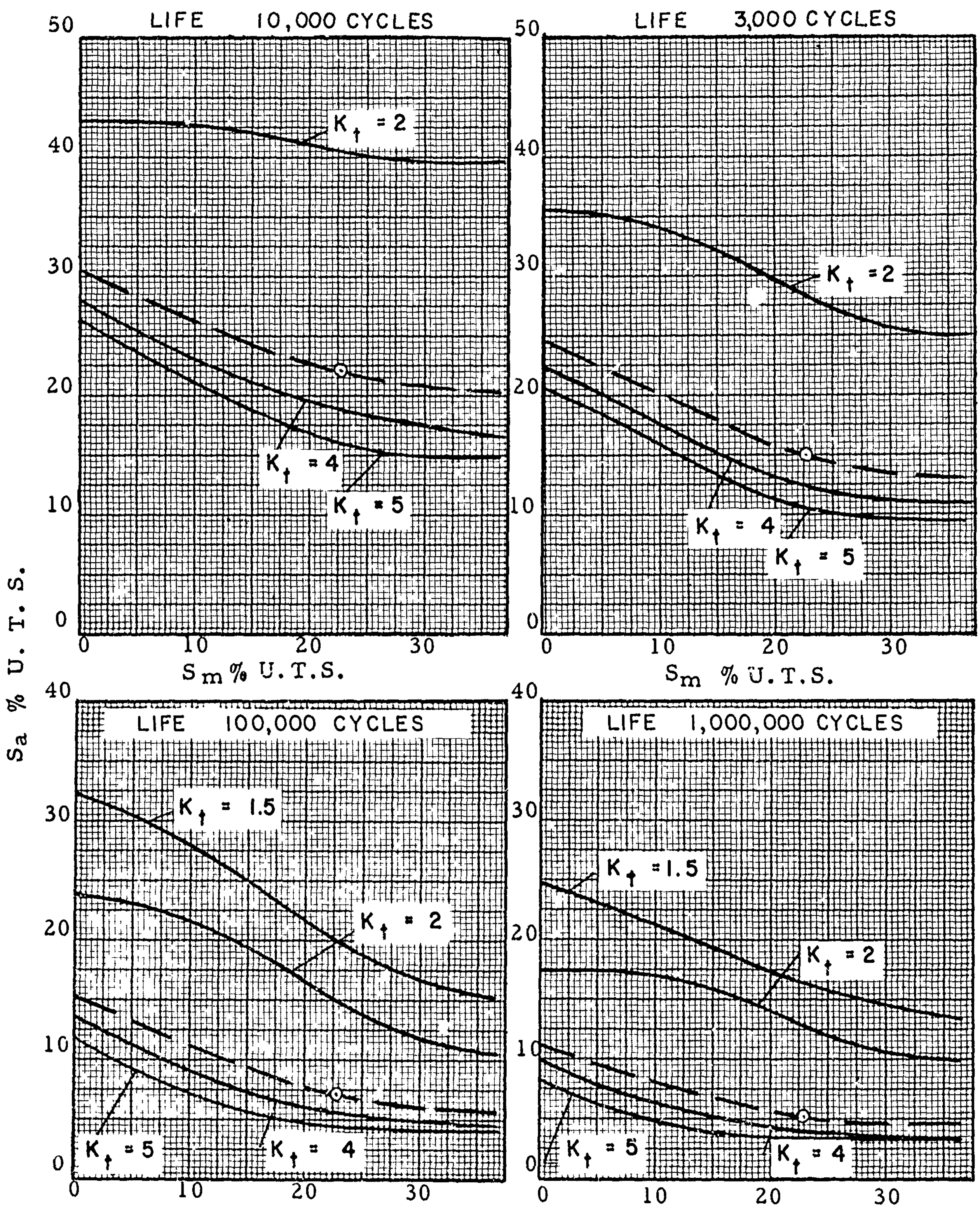
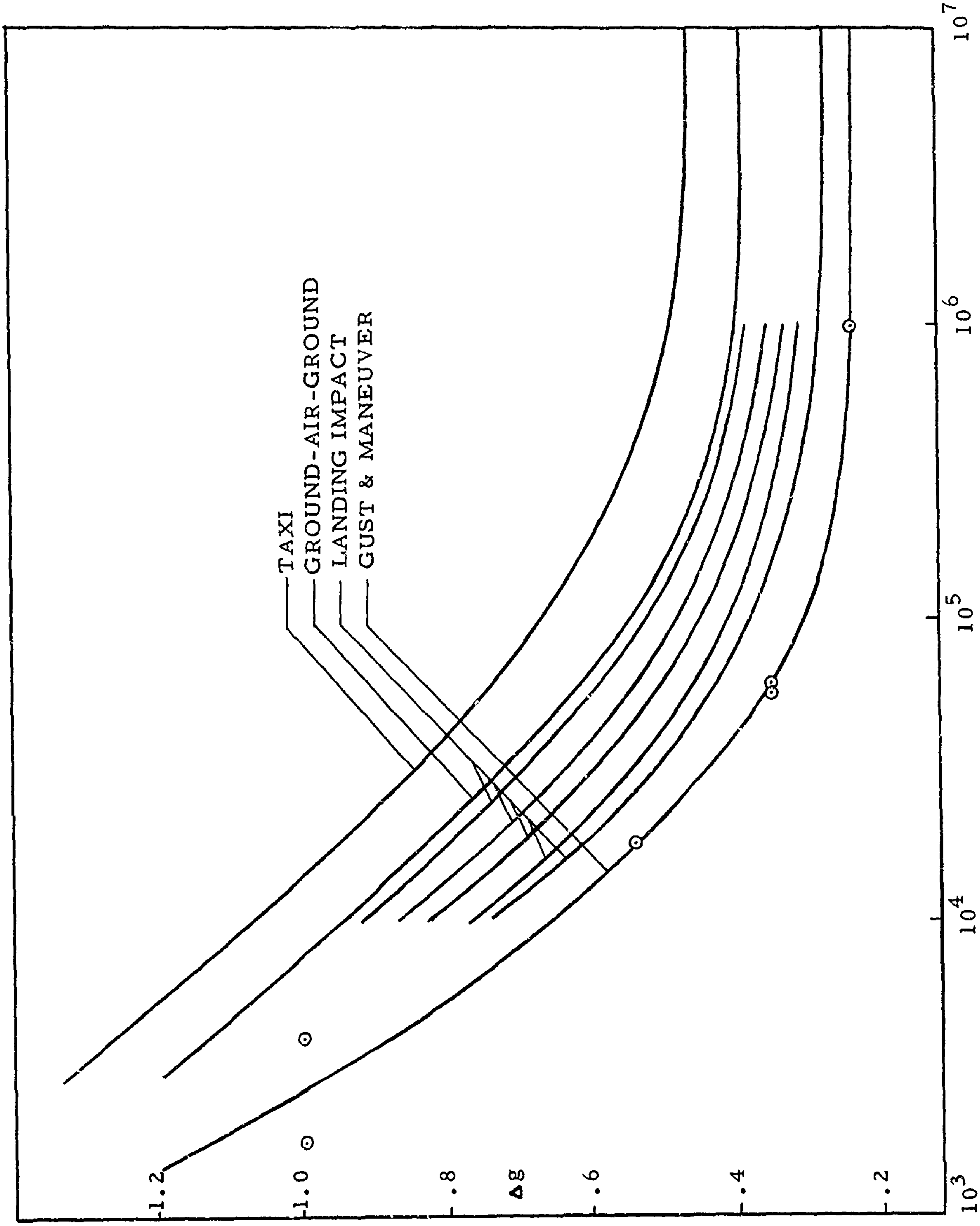


FIG. 12
ALLOWABLE FATIGUE CYCLES



Session Chairman:
Professor A. M. Freudenthal, Columbia University

Panel Members:
Mr. E. R. Shanley, University of California and Rand Corp.
Dr. Horace Grover, Battelle Memorial Inst.
Mr. Joseph P. Butler, Boeing Airplane Co.
Mr. R. E. Peterson, Westinghouse Research Lab.
Dr. B. E. Gatewood, Air Force Institute of Technology
Mr. James C. McClymonds, Douglas Aircraft Co.
* Mr. James E. Hayes, Oklahoma City Air Materiel Area

Editorial Note: Attention is directed to the editorial policies presented in the Preface which were followed in editing the discussions of the Forum.

CHAIRMAN, PROFESSOR FREUDENTHAL:

The first question is directed to Professor Shanley, by Mr. Schuette of Dow Metal Products. It says: "You indicate fatigue may be a function of plastic strain amplitude alone. This seems unreasonable if part or all of the strain is compressive. Would you please comment on this?"

MR. SHANLEY:

We have fatigue failures in compression and the results - the type of failure that we see is usually a little wedge which indicates that the crack did form along a diagonal plane and is no exception to the general idea that the fatigue failure is caused by shear stresses — shear stresses cause plastic strain. Perhaps the point that is bothering you is that it does require much longer time to produce failure if you have a mean compressive stress and as you know, they have produced failures in compression when the entire stress range stayed on the compressive side and this has puzzled some people. But what I didn't have time to say in the paper is that in developing the basic theory, I not only assumed that the fundamental factor in causing the crack is the cyclic plastic strain but the next hypothesis or assumption after that is that the way the crack grows is also affected by the normal stress acting across this plane. In fact it is through that hypothesis that I get the correction factor which moves the SN curves up and down on the SN chart and gives you the different curves for the mean stress. So, unless I misunderstand the question, the answer is that the theory does include compressive stresses and this is done by adding one more factor which may be linear or may be some other function but at any rate, it is a function of the normal stress.

CHAIRMAN:

Which brings us immediately into the next question! There are two questions. The two questions are rather similar. One is by Mr. Schleicher of North American who asks: "Please discuss briefly fatigue strength (or life) involving combined stress i.e. biaxial or even triaxial stresses".

*Mr. Hayes presented a summary of his paper at the start of the forum. Technical recording difficulties prevented the inclusion of his presentation in the forum session.

And then there is the question by Mr. Ades of Bendix which says: "What is ϵ (epsilon) under conditions of combined stress?"

MR. SHANLEY:

Again I had to skip that portion of my paper. There were no slides on it either but it seems to me that a very simple concept will go a long way towards solving the combined stress problem of which the mean stress problem is just one factor. This concept which I had in my early paper is simply this: during the largest part of the fatigue life, the crack is growing within probably a single crystal or at least it should be if the design is correct. Now if we think of this crystal (of course there may be thousands of these that are trying to develop cracks, but let's pick any one of them) this crystal is completely surrounded except for the surface by others and it is not free to move in any direction or slip merely as a result of the stress on it itself. You might say that the little crystal is a specimen in a larger testing machine and the testing machine is the fatigue specimen itself. Now this idea means that when we encounter combined stresses what we should do is apply the best knowledge we have on the theory of plasticity to calculate the plastic strain in the region of this crystal. If the crystal is part of the overall mass of crystals, it must accommodate itself to this strain whether it be weak or strong. That furnishes us with a tool by which we can apply all that we know about the theory of plasticity, whether you want to use octahedral stress or what-not. Actually, all I did in the paper was to take the same equations, replace - instead of saying that the delta epsilon sub-t is say a power function of the axial stress or whatever you want to call it, you merely say it is a function of the shear stress - whatever shear stress you wish to use.

CHAIRMAN:

Thank you, Professor Shanley. Let's change the subject for a moment so we get around and hand a question to Mr. Butler - again two questions - I'll read the overlapping part first. "What is industry's definition of 'first fatigue crack' and how is it determined?" which is a question by Mr. Frankel of the Bureau of Standards and the same question is put by Mr. Stone of WADC, who says "In general, what is the definition of 'first crack'?"

MR. BUTLER:

The definition of "first crack" is of particular application to structural component tests or multiple member tests. In other words, it's an attempt to define the appearance of the first crack in either a stiffener or in the skin member of a panel that consists of several stiffeners or several skin panels or combinations of it. Its detection is either by inspection at repeated intervals or by the use of crack detection wires. This latter scheme is merely the cementing of a very fine insulated wire to the structural member in the vicinity of those areas which are suspected of being critical areas in the panel. This is not always successful, I'm sorry to say, and that perhaps the panel too will be devoted to finding the critical locations but by use of this wire which is cemented adjacent to a rivet or a bolt location, the growth of fatigue crack from that particular area will pass under the wire and cause its subsequent fracture or cause the fatigue test machine to stop, or lights to flash or bells to ring or whatever you prefer. This does not provide detection of the crack in microscopic form but rather it is a fatigue crack which can by careful or reasonably careful inspection be seen. It is something that you would expect a maintenance man of an airline or a service to be able to detect during inspection periods on the aircraft structure.

CHAIRMAN:

Thank you. The second part of Mr. Frankel's question is as follows: "With reference to the factors causing variability, such as environment, manufacturing processes, etc. can you state how much is contributed to the total by each of the components?"

MR. BUTLER:

I can answer that real quick-like! I can't. It is possible to examine data in the literature to find out the influence of these individual processes and I think that each one must be considered in the light of the particular design or the particular fatigue problem which faces the individual.

CHAIRMAN:

There is another question to Mr. Butler from Mr. Bernard Simon of Aerojet which reads: "Have the specimens which fell on the high end of the scatter been investigated to determine how these specimens differed from those which fell near the mean or normal scatter points and if so, in general what was found?"

MR. BUTLER:

In many of the specimens which fell on the high side or rather I should say for most of the specimens that failed on the high side this data is obtained through literature search and examination of the particular material, its condition and any of the many associated details important to fatigue performance were not detectable by our study, hence I cannot add any information to that. Generally though, those in the low scatter band were those of typical joints or bolted or riveted type specimens.

CHAIRMAN:

Thank you. There is a question to Mr. McClymonds from Mr. Galof, Radioplane Division of Northrop: "Have you investigated the results of white noise applied to a structure which has significant response at two or more natural frequencies?"

MR. McCLYMONDS:

The answer is that we haven't investigated this. I wish I could. I wish I had a wave-noise generator that would put out the type of acoustical output that I require. If this were available, I think it would be one of the finest tools that we could have in this problem but to my knowledge it is not available.

CHAIRMAN:

Thank you. To Dr. Grover from Mr. Berks, Reaction Motors Division, Thiokol Chemical Corporation: "What is the effect of cycling rate on fatigue damage? An S-N diagram, for example, implies that the cycles to failure is independent of cycling rate; is this true?"

DR. GROVER:

First, I would like to say that the statement that the S-N diagram is independent of cycling rate certainly has some limitations. This is not true for situations where other independent processes - it is not true for example in elevated temperatures where creep may exist and be important. It is not true in the presence of corrosion which has different time dependents. So these limitations, of course, apply to damage also. So far as the damage being independent of cycling rate in circumstances where the S-N curve appears not to be, I don't know of any real evidence that is statistically significant. I suspect that the damage is not solidly dependent on cycling rate except under situations where the S-N curve would be so likewise. Perhaps someone else on the panel has a better answer than this.

CHAIRMAN:

Thank you, Dr. Grover. There is a question by Mr. Marble of General Electric to Dr. Grover which I'll read: "Isn't the important thing the reduction in endurance or long-life stress to failure rather than life? The crack or incipient fracture which will propagate is the factor of danger." I hope Dr. Grover understands the question.

DR. GROVER:

I'm not sure I understood that one either. There are several ways of looking at calculations. One may look at the calculation of reduction in strength or reduction in life with some advantage in one way or the other as in arithmetic or alegbra. I am not quite sure, perhaps, Mr. Marble could explain more accurately what the question is, what he had in mind.

MR. MARBLE:

Well, essentially, shouldn't we have a damage criteria on stress rather than life? Shouldn't we calculate it all at constant life basis? What we want to know is an aircraft that we all know will last about the same length of time, they're all going to be obsolete at about the same time, what we want to know is what does it go through that causes it to lose some load-carrying ability and it is this measure of loss of load-carrying ability that we really want to know - not that we want to change the life?

DR. GROVER:

This one would be a good thing if I catch your point. We might be able, for example, to use an aircraft longer if we knew something more about the reduction in its stress-carrying ability and could use it longer under lighter loading conditions but I don't think that I know enough about the details of design to answer that question really. Perhaps Dr. Shanley would care to make a remark on this question?

MR. SHANLEY:

I'd like to make a remark on a closely related subject which I think should always be brought up about this time and that is, that in all the discussions of scatter and variability, we seem to deal with life which is natural because it is the only thing you can measure in the test. But many years ago, in fact, when

the very first NACA test for variable amplitude loading was published (I have a reference in my paper - at least among my area papers) it occurred to me to translate this into the stress required to produce a given life rather than the life which would be obtained with a given stress, and I was very much surprised to discover two things. First, that the very wide scatter and stress in life became what I would consider a perfectly nominal scatter in stress and secondly, that the different methods of simulated damage tended to give just about the same results. I had developed a method at that time which I called the 2x method and it was because of the publication of the NACA results that I developed the method. Then when I found that really the stress - the designer is interested in the stress, that's all he can control. He cannot control the life only through the stress, so the real question and the one which affects the weight of the airplane and which finally gives us the payoff on this thing which I mentioned before, is not really the life. We should say what life we want and put a good big factor. You saw factors of four or five I wouldn't mind saying ten. After all you're only moving one notch over on the logarithmic curve to get a factor of ten and why not do it? Do you think that this is going to ruin the capability of the airplane? I don't think so. Then having picked this long life, now go back and take everything that has been said on scatter and translate it into stress and if you'll do that, you will be very much surprised. I would be willing to bet money that the scatter is always going to be less than that that we know we have for the buckling of the thin wall shell which we use without any question. This I think must always be said that the structure's design and stress man is interested in stress and much of the pessimism that we've heard about scatter will become optimism if we do that.

CHAIRMAN:

I hope this implies the realization that you accept the scatter band of one to ten as something normal in life.

MR. SHANLEY:

Now imagine that you designed a specimen which is just barely below the endurance limit. It will have infinite life. Maybe you slipped up a little and the stress is one-thousand psi higher, maybe only five hundred or the diameter is just a little smaller, this specimen will now fall over on the diagonal part of the curve. This scatter is infinite. That's all I have to say on that.

CHAIRMAN:

If you'll keep the microphone on, we'll have another question. The question is from Mr. Grosskreutz of Midwest Research Institute: "Do you know of any data supporting the linear log delta epsilon versus log N plot for the values of delta epsilon smaller than one-thousandth?"

MR. SHANLEY:

There are some in the references that were given there which as I said show that the curves start to branch off to the right and now you do see a large difference in what might be called the endurance limit and this is to be expected because the main difference between the materials is that some materials start to slip earlier than others. After all, that's all we're doing in the heat-treated materials. We're raising the stress at which it starts to slip, so that if you can remember this curve rather than this single line -- if you now heat-treat this material and raise

the yield stress or the proportional limit by ten thousand psi, you're going to change the point at which you begin to get fatigue and this gets tied up with the question of whether you're plotting plastic strain or just plain cyclic strain. These details I didn't want to go into. I didn't make it clear on that diagram because some of the papers report plastic cycling strain and others report cyclic strain. Unfortunately when you make the test, you have to control the strain period - not the plastic strain, so this is a little detail that needs to be worked out. I hope that answers the question.

CHAIRMAN:

Yes, it answers this question but Mr. Yorgiadis has one about the same type, so that you can't get out of it. It says: "What is the 'plastic cyclic strain' ordinate of Fig. 1? What is the bandwidth beyond one million cycles and up to one hundred million cycles? Why was plot limited to one-half a million cycles?" This brings you right back.

MR. SHANLEY:

I merely re-plotted Figure 1 beyond the paper I referred to - the ordinant I can't very well tell you without putting the slide back on but one thing I will say is that at say ten to the three cycles to failure, one thousand cycles, the plastic strain is about two per cent. Now if you remember the equation which has the power of two in the denominator, that means that all you have to remember is that one point of the power of two and you can draw a diagram. As to the second part going out into the high cycle range, I think I answered that in the first question. I hope.

CHAIRMAN:

Thank you. But keep the microphone because there is a question by Mr. Frankel of the Bureau of Standards: "I believe that you stated that plastic flow at the root of a crack causes an advance in crack length. Plastic flow implies dislocation or vacancy movement, however recent work by Rosenberg in England at liquid helium temperatures, seems to preclude any vacancy motion. Do you think that another mechanism operates at these low temperatures?"

MR. SHANLEY:

I don't know whether to say I don't know enough about it or whether we should meet outside but in any case I think we had better not try to go into that. I'd like to discuss it later though.

CHAIRMAN:

Thank you, it is a rather complicated question. I just wondered how you could get out of it!

The next question is to Mr. Butler by Mr. Goodling, Olmsted AFB: "Have many fatigue life tests been made on shot-peened specimens? If so, what scatter characteristics were indicated as compared with tests on plain specimens?"

MR. BUTLER:

From personal direct knowledge I am not in a position to discuss this question, I think. I have observed data performed by other people and shot-peening in itself does provide scatter in test results. There is data existing which I would like to call to your attention concerning something like shot fatigue, that it is possible in the long life cycles to get tremendous increases in fatigue performance by shot-peening. However, if the test or test stress levels are increased to the finite levels often experienced in aircraft structures, the advantage of shot-peening is negligible. It dwindles out. There is also other work that says that repeated load cycles on materials that have residual stresses such as you would get from shot-peening can disappear with time.

I would just like to make a switch here - about a question a moment ago and indicate that I was perhaps out of step with a good many people in the auditorium here. This concerns the significance of stress and fatigue life. I'm afraid that if you use typical structures and use stress as a means to accomplish increased life - that is the reduction of stress - that there are finite weight penalties associated with this type of endeavor and I suggest that any one interested in exploring this factor do try this approach with developed fatigue curves and if I may be so bold to suggest the minor scheme of the damage and note the problems in which one attains or reaches by attaining fatigue life, by reduction of stress, it is more than a few per cent that Mr. Shanley here has indicated but again fatigue is a many-sided subject and it is all from the point of view. Thank you.

CHAIRMAN:

We have a few minutes - not too many and since there have been no questions on stress concentration and Mr. Peterson is sitting at complete ease, I will put one question to him: "I would like to know Mr. Peterson's view as to whether through stress concentration there is a possibility of correlating specimen tests with structural parts or with whole structures - how would you look at this subject?"

MR. PETERSON:

I think I indicated that you have a long ways to go from just a very simple test to complete airplanes. There are so many factors. I think Mr. Rhode brought this out quite well yesterday. You have to consider spectrum loading. You have to consider the relation of in-stress to alternating stress. You have to consider all these statistical variations, so that it is very, very difficult. I regard this as simply the first step in the whole process.

CHAIRMAN:

There have been proposals from some quarters saying that an airplane, for instance, or a structure can be defined in terms of its critical stress concentration factor and then correlated. These tests were on specimens with the same concentration of stress.

MR. PETERSON:

I doubt whether that is possible.

CHAIRMAN:

We may have one or two more questions from the floor. Will you identify yourself?

MR. BERKS:

My name is Berks from Reaction Motors Division of Thiokol Chemical. I have a question for Mr. Peterson. What is the effect of stress concentrations on a single load application which is applied at a very rapid rate? I know we ordinarily disregard stress concentration for a single load application but does the rate of load have any effect?

MR. PETERSON:

Well, it certainly must because you're speaking about an impact test and if we take a charpie test - of course this is just what you do, you put a "V" notch into the piece and this of course has a very pronounced effect. But whatever method you use for analyzing that would be quite different from fatigue.

MR. BERKS:

I wasn't considering an impact test as such but considering a rocket motor case that is suddenly pressurized. With the pressurization we may have built in stress concentration in the case somewhere.

MR. PETERSON:

Ordinarily we disregard stress concentration and disregard the rapid loading rate.

MR. BERKS:

I would like your opinion on whether this is a reasonable assumption to make or whether we should take into account this stress concentration under this loading?

MR. PETERSON:

What kind of material is this?

MR. BERKS:

Let's say a high strength steel.

MR. PETERSON:

I would think it would be very questionable to ignore this, although I don't have any information on this. I would think you would have an effect.

MR. BERKS:

How about in a ductile material, such as the aluminums?

MR. PETERSON:

Again it would depend upon the rate of loading. I don't think you can make a specific answer to your question unless you put down all of the conditions - the precise rate of loading and so on.

MR. BERKS:

Well, what can we compare the loading rate to? Is it to the speed of sound in materials, for example? Or is there any other criteria which we can use? What is to decide whether our loading rate is "rapid" or not?

MR. PETERSON:

I don't think I can answer that particular question. I haven't worked on that particular one myself. Perhaps someone else on the panel has.

CHAIRMAN:

I could try just to make one short remark. It appears obvious that any reduction in stress concentration effect depends on the appearance of plastic formation and if the loading rate is high enough to prevent plastic deformation, then probably the full stress concentration should be considered. Now tests have been made recently by Campbell and somebody else in England on the effect of high loading rate on plates. I don't know to what extent this would be applicable and they have indicated that probably a value of two to three times the yield limit can be supported if this limit is applied extremely shortly. It mentioned the time range of microseconds. There is work on slip delay which has been started at the University of California some years ago and there is a possibility probably of correlating loading rates with the possibility of slip delay or slip initiation if you are in a range where slip cannot occur before the load - at a time when the load is already removed then you probably will have no plastic effect. If slip occurs you will have flow or the partial plastic effect and here it would depend on your actual loading rate applied and the question can hardly therefore be answered in a general way.

Gentlemen, we are just on time for lunch. The meeting is adjourned.

INTRODUCTION TO SESSION IIA

COLONEL HARVEY P. HUGLIN

WRIGHT AIR DEVELOPMENT CENTER

In this afternoon's session on "Fatigue Loads-Measurement and Prediction", your session chairman will be Mr. John H. Meyer. Mr. Meyer at the present time is General Manager of Atom Apply, a Missouri firm providing engineering consulting services in the structural, mechanical and nuclear fields. He was formerly with the McDonnell Aircraft Corporation, where as structural manager, he was responsible for the structural design and integrity of fighter and transport aircraft produced for the Air Force and Navy. In this capacity Mr. Meyer became intimately acquainted with the fatigue problem area especially from the structural design and operational viewpoint. He is a graduate of the University of Missouri, a structural engineer of that state and an Associate Fellow of the Institute of Aeronautical Sciences. His latest publication entitled: "Thermo-elastic Wing Structural Design" appeared in the Journal of Aeronautical Sciences in 1958.

STRUCTURAL FATIGUE UNDER RANDOM LOADING

By

Melvin Stone

Douglas Aircraft Company, Inc.
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ABSTRACT

The establishment of fatigue life due to random environment requires the understanding of input spectra, dynamic-elastic and aerodynamic properties (transfer functions), response characteristics and damage rates. The general treatise and problems confronting the designer in the determination of fatigue spectra under random loadings are discussed. Comparison is made of incremental bending moments on a swept-wing bomber using rigid and elastic transfer functions. The advantages of power spectral and discrete representation are shown in determining fatigue input spectra. Information that can be gained from the structural integrity program to establish fatigue criteria is outlined. Additional recommendations are made to further realize the full potential of existing programs.

INTRODUCTION

For some time designers have been engaged in research involving fatigue evaluation. The necessity for establishing a comprehensive and sound program has long been recognized. An increase in the state-of-the-art can be accomplished with the presently planned governmental programs if the interested and competent persons in industry, government and the universities work together. A careful analysis of this complex problem is required, as well as a thorough knowledge of the available information so that the overall effort can be minimized to a manageable task.

The purpose of this paper is to examine existing methods and techniques. Sample calculations illustrate the importance of properly considering the flexibility of the structure. The derivation and representative form of the input spectra for fatigue criteria is discussed. The advantages of the fatigue research and development studies and the structural integrity program are shown.

Random analysis techniques have been used in all examples throughout the paper; however, whenever possible, discrete analysis techniques have also been included.

Extremely important problems concerning aircraft fatigue life still confront the designer. Why, after so many years of aircraft design experience, are we still troubled with aircraft fatigue problems?

The parameters affecting fatigue life such as the atmosphere, runway roughness, and pilot's technique have not changed. Changes have, however, taken place in performance, configurations and mission requirements.

The increase in performance, mainly higher operating speeds, and the optimization of structural weight makes the flexibility of the airplane important in predicting fatigue life.

The speed, weight and time spent during maneuvers are usually different than those assumed during the original design and result in higher 1 "g" stress levels, principally because many other uses are found for the aircraft in service.

If one considers the changes that have taken place, then present analytical methods and techniques used to predict aircraft fatigue life should be examined.

BASIC STEPS IN FATIGUE ANALYSIS

There are many who think that it is not necessary to make a rigorous analysis. In fact, simple rules such as low stress levels and careful attention to detail design are the golden rules used by many manufacturers. Before we accept any rules or establish design criteria, however, let us study the fatigue results of a swept-wing bomber. For convenience, the analysis will be shown in power spectral form rather than discrete form. An example of a swept-wing bomber has been chosen to illustrate the methods of response to random atmospheric turbulence using the following basic steps:

1. The load producing input conditions

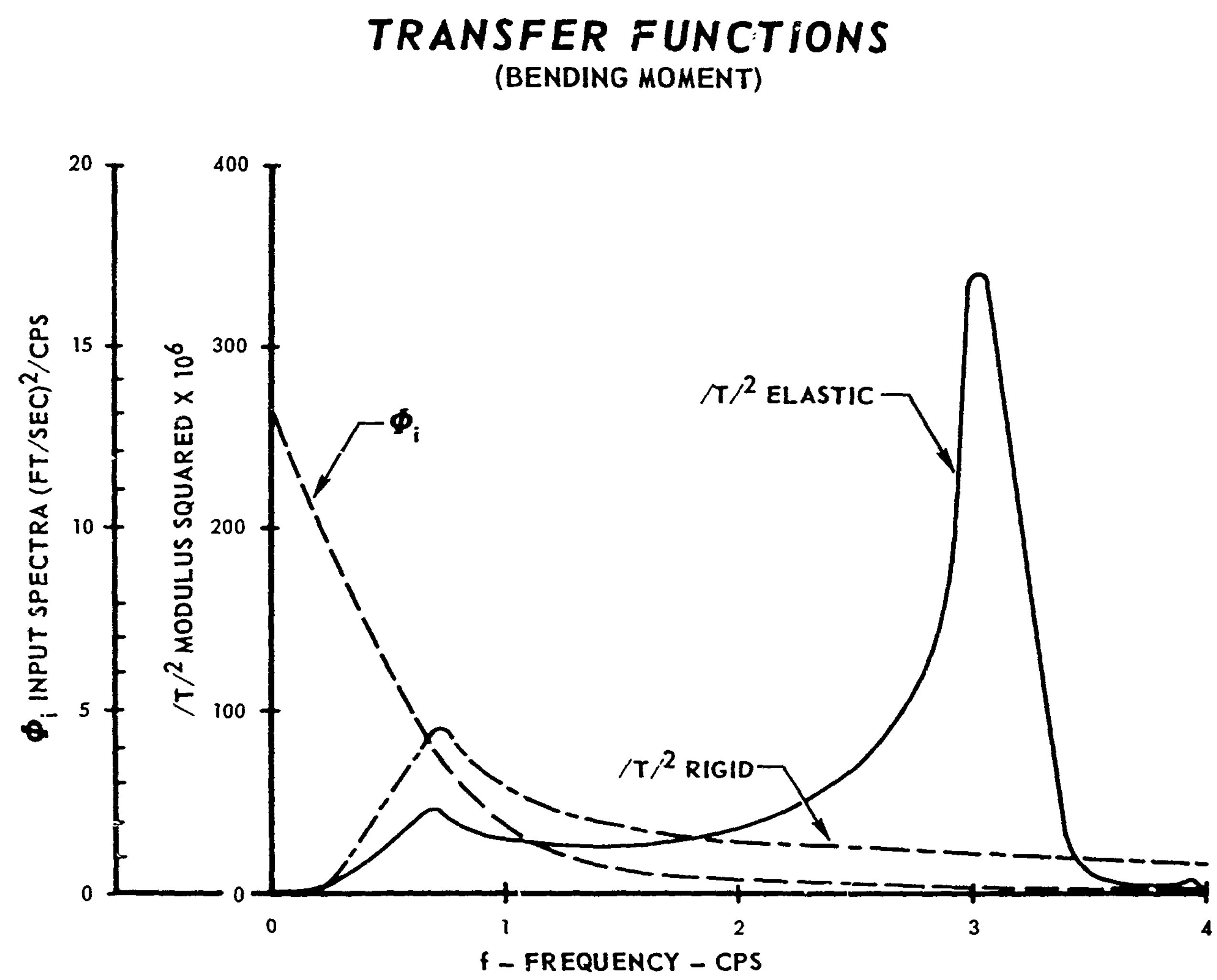
2. Dynamic-elastic and aerodynamic properties (transfer functions)

3. Response characteristics

4. Fatigue analysis (damage rates)

LOAD INPUT CONDITIONS AND TRANSFER FUNCTIONS

Figure 1 shows the input spectrum for atmospheric vertical gust velocity typical of turbulence found in low level flights over farmlands. It also shows a comparison of rigid and elastic transfer functions for a specific weight, altitude, C.G., and speed condition.



TRANSFER FUNCTIONS
(BENDING MOMENT)
Φi INPUT SPECTRA (FT/SEC)2/CPS
/T/2 MODULUS SQUARED X 106
20
15
10
5
0
400
300
200
100
0
Φi
/T/2 ELASTIC
/T/2 RIGID
0
1
2
3
4
f – FREQUENCY – CPS

Figure 1

The gust, though random, has been assumed to be vertical and symmetric with no spanwise variations, and only symmetric rigid and elastic modes of the airplane have been excited. The penetration effect of various portions of the wing and tail entering the gust at different times has been included in the response calculations. The transfer functions, which are the dynamic-elastic characteristics and the aerodynamic properties of the airplane, were obtained by means of equations from flutter analysis.

As a basis of evaluation, the rigid transfer function has been determined by using two degrees of freedom (rigid body vertical translation and pitch modes) whereby the elastic transfer functions also include flexibility. The aerodynamic forces were computed for two-dimensional, incompressible flow methods with lift-curve slope corrections for three-dimensional and Mach number effects. The actual shake test modes, frequencies and dampings were used.

The first peak is associated with the short period mode of the airplane and the second peak with the first wing bending mode of the airplane. At the short period mode the elastic transfer function is less than the rigid transfer function due to aeroelastic relief. The effect of first wing bending mode is shown at the higher frequencies.

RESPONSE CHARACTERISTICS

The frequency characteristics of the output of a linear system subjected to a stationary random input is given by reference 1.

$$\Phi_o(\omega) = \left|T(\omega)\right|^2 \Phi_i(\omega)$$

Where $\Phi_i(\omega)$ = power spectral density of input

$\Phi_o(\omega)$ = power spectral density of output

$\left|T(\omega)\right|^2$ = modulus-square of frequency response or admittance

The outputs of interest are the sectional accelerations and bending moments at various stations on the airplane that are generated as a result of the input to the system.

The procedure used was:

a) Compute the frequency response functions $\left|T(\omega)\right|^2$ for the various output quantities.

b) Multiply the frequency response functions by the given gust spectrum input to obtain various output or response spectra.

The effect of elasticity is to relieve the response at the airplane short period mode frequency and to increase the response at the higher frequencies.

The output spectrum indicates the extent to which various frequency components are present in the response and it allows the determination of various statistical properties of the response time history such as incremental C.G. acceleration and incremental bending moment.

The load amplification at the C.G. and wing root caused by elastic mode resonance are offset by the large relief obtained in the short period mode and, therefore, the overall effect of elasticity is reducing (see Figure 2). However, at the outboard wing stations the effect of elasticity is load amplifying. It should be noted that in other swept aircraft the effect of elasticity can be amplifying even at the root of the wing (reference 2). All loads and accelerations obtained are incremental and do not include the 1 "g" steady-state flight conditions. At higher altitudes where less aerodynamic damping and static elasticity exists, the elastic mode resonance can overcome the relief in the short period mode and the effect of elasticity can be load amplifying.

The source of the larger amplitudes of incremental C.G. accelerations and bending moments are representative of high input power in the short period mode and the smaller values of incremental load are representative of less input power at the higher frequencies. In fact, the elastic load curves cross over the rigid load curves because of the effect of first wing bending mode at the higher frequencies. The loads which are the number of response peaks that occur per second above a given incremental bending moment represented by the abscissa were obtained from Rice's equation, reference 3.

FATIGUE ANALYSIS (DAMAGE RATES)

Once load spectra is established, the stress spectra can be determined by analytical fatigue analysis. The relationship between load and stress is the primary function of the stress analyst and many methods are available for performing this task. Finally, it is possible to evaluate fatigue life of the aircraft from tests relating load spectra with damage rates at various locations of the structure. A thorough treatise of the damage theory and testing procedures will be covered during these proceedings; however, based on present damage theory, the effect of elasticity shows a decrease in damage for the swept bomber.

INPUT SPECTRA

Flexibility is extremely important whether the representation is by a discrete or power spectral form. The discrete approach has many advantages: 1) non-linear representation, 2) good check on experimental transfer functions, and 3) better relationship of maximum stress amplitude to minimum stress amplitude which is important to good fatigue analysis. The power spectral approach also has advantages. The common denominator of all of these problems is that the time history of the forcing function cannot be specified in detail since it depends on chance. In the past, the difficulty has been overcome for gust response by assuming discrete gust profiles. These assumed gust shapes do not really resemble measured gust shapes and are not, therefore, well grounded physically. A random gust analysis differs from a discrete gust analysis in that it recognizes at the outset the random features: i.e., an airplane has known characteristics subjected to a forcing function of gust velocity with known statistical averages that determine the statistical characteristics of the response. The statistical characteristics of gust velocities are provided by its power spectral density function. Figure 3 shows a typical plot of both discrete and power spectral approach for vertical gusts; however, the taxi and maneuver spectra can be shown in a similar form. In the case of this discrete approach, an input shape has to be defined.

In the case shown in Figure 4 the discrete signal for the runway profile can be used directly with airplane transfer functions to determine loads and stresses. Although the plot is typical of runway roughness, it can be shown as a time signal for maneuver and gust.

INCREMENTAL BENDING MOMENT

CONTINUOUS GUSTS – 1000 HOURS

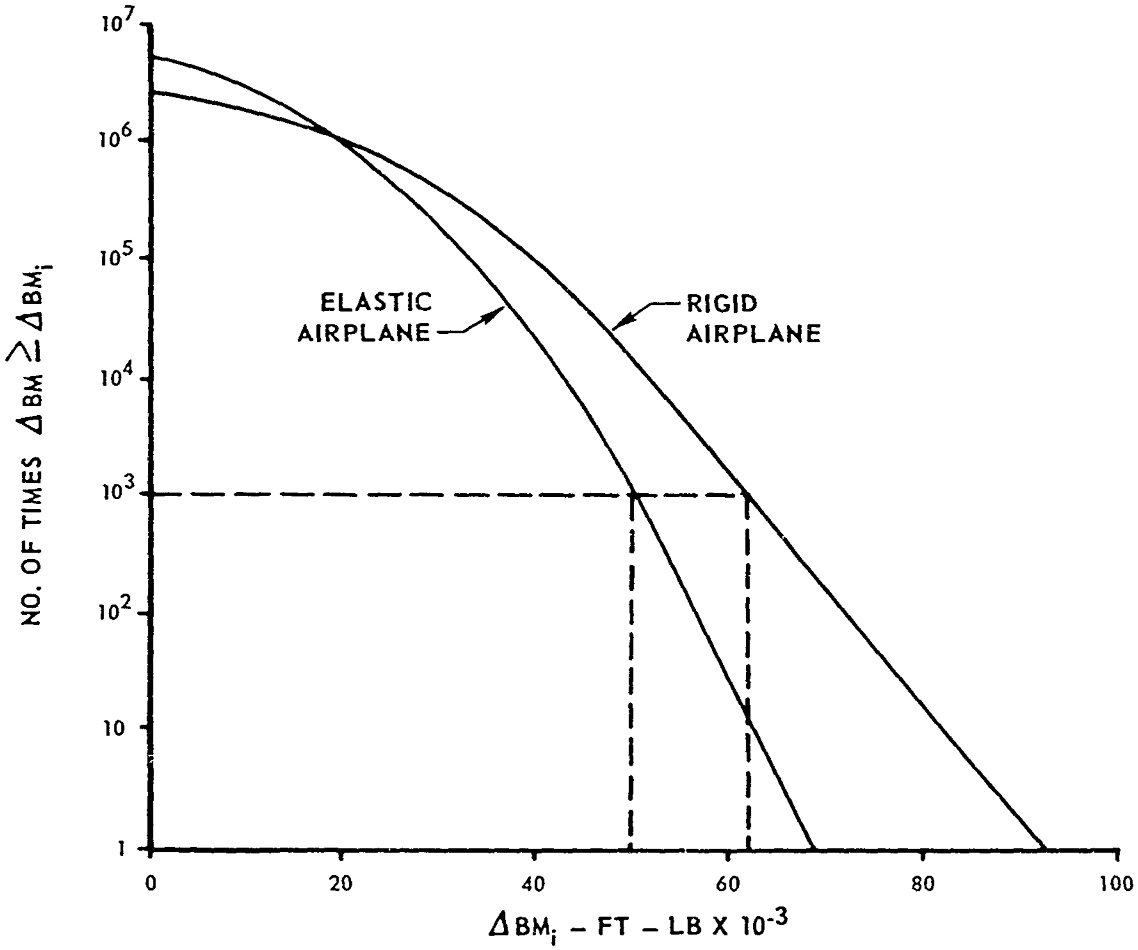


Figure 2

DISCRETE AND POWER SPECTRAL PRESENTATION

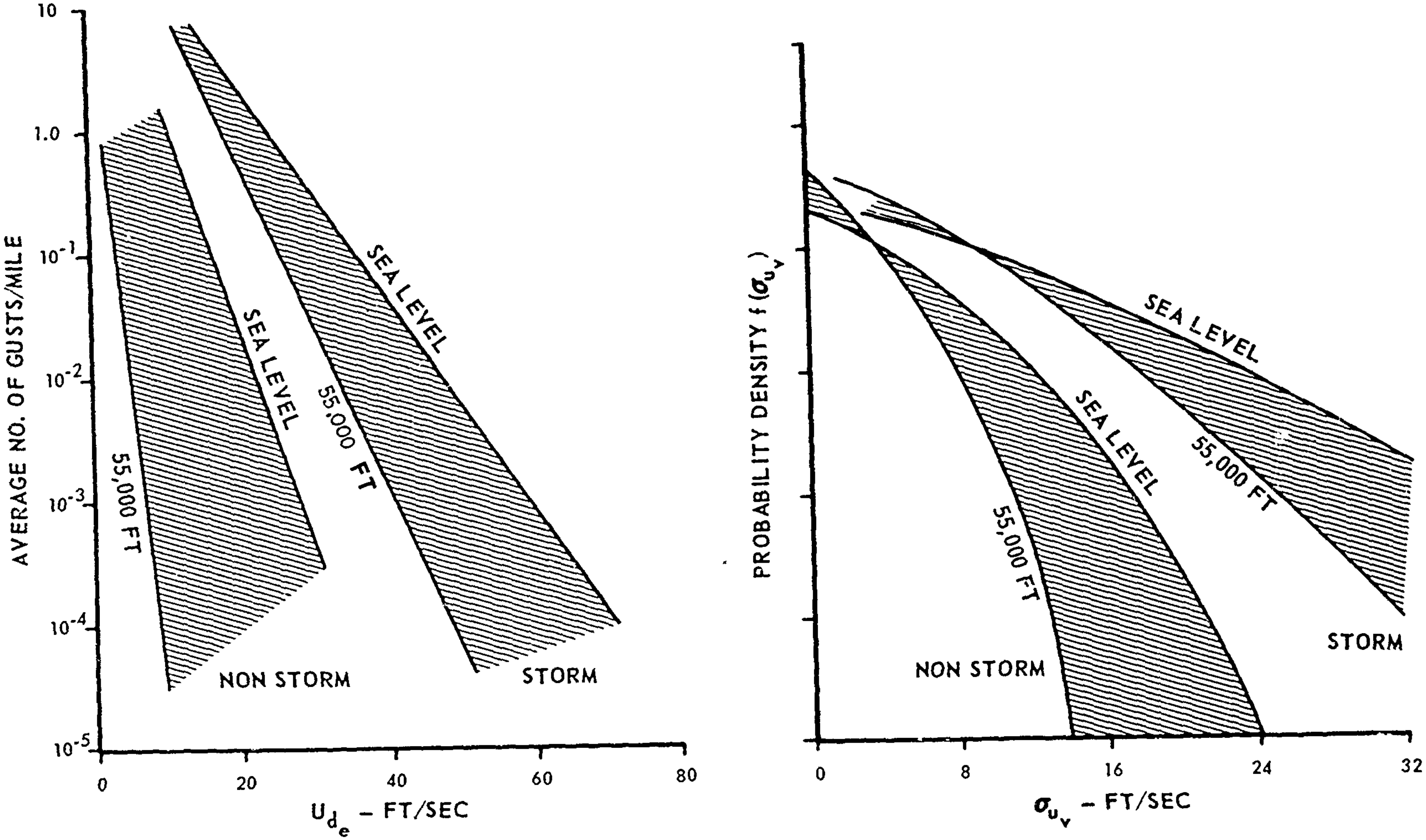


Figure 3

DISCRETE RUNWAY ROUGHNESS PRESENTATION

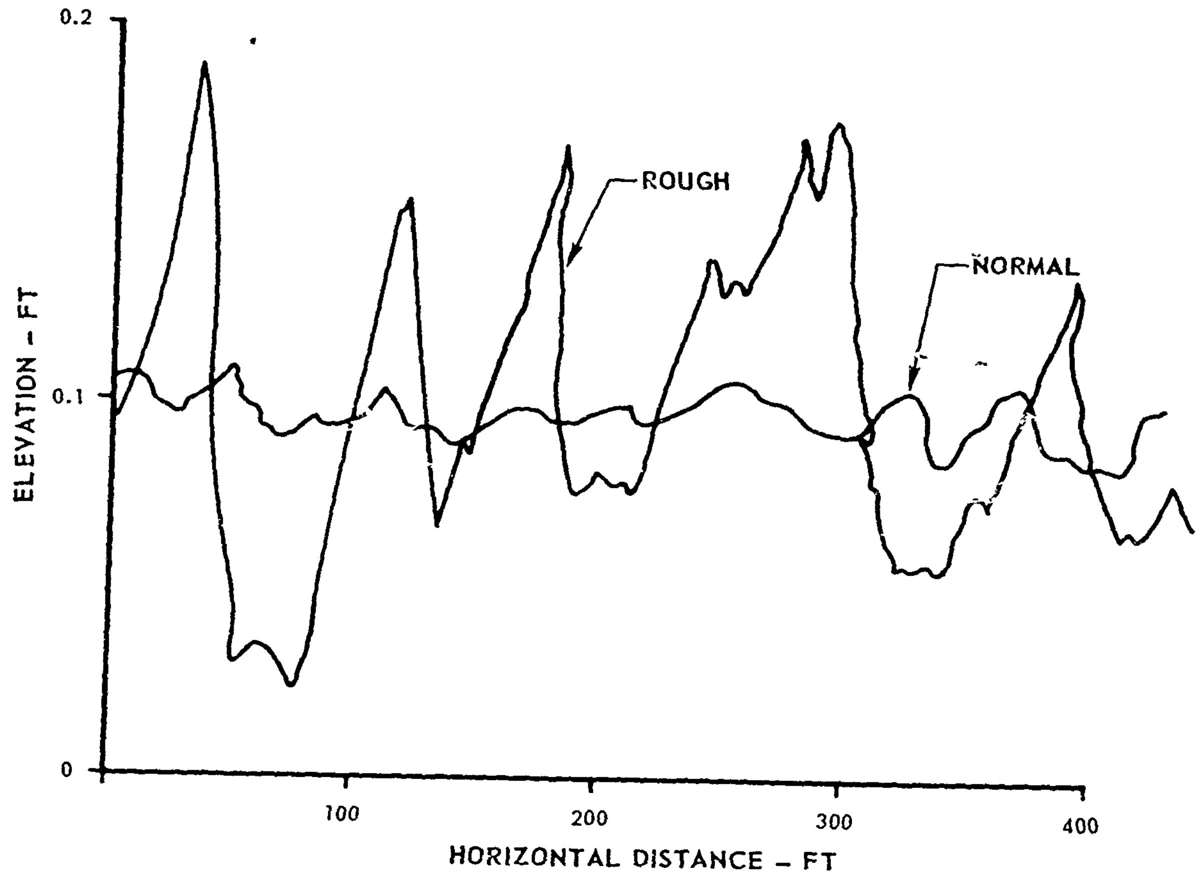


Figure 4

Realizing the need for optimum fatigue criteria, let us look at load producing input conditions. The major load input conditions are gust, maneuver, taxi and sinking speed (reference 4). The runway roughness or taxi input can be completely divorced from the airplane and expressed in its power spectral or discrete form. The NASA and the Air Force have collected a large amount of data on the runway roughness problem (references 5, 6 and 7), and additional data will be collected by the Air Force by use of a cart equipped with automatic data handling instrumentation.

It is felt that the information collected to date is adequate for design criteria. The power spectral representation can probably be put in a form to normalize the standard deviation of the runway roughness spectra. It is felt that the runway terrain should also be shown in discrete form so that the analyst can use non-linear representation of structure.

The sinking speed probability data collected to date is ample for design criteria. The data collected to date, however, can be further classified according to characteristics of aircraft (reference 6).

Besides taxi and sinking speed data, other statistical data such as landing and taxi weights, weight distribution, C.G. position, forward landing speed, taxi distance and velocity, time, etc., are extremely important to define proper airplane transfer functions.

GUST INPUT

Gust spectra should be represented in both discrete and power spectral form and left up to the analyst for its proper usage. The discrete form is quite familiar and has the advantage of being able to use non-linear terms.

Some flight test data have been assembled in power spectra form whereby the atmosphere has been divorced from the airplane by properly instrumenting the aircraft and correcting for the aircraft motion relative to the ground. The B-66 Low Level Terrain program has collected the three orthogonal components of the turbulent velocity, vertical, lateral and head-on gust spectra data over various levels of terrain and meteorological conditions (reference 8). This program should yield valuable information on spectra and cross-spectra of the gust velocity components and the effects of spanwise variation of head-on and vertical components. The effects of meteorological conditions such as mean wind speed, lapse rate, altitude, flight path and terrain characteristics will be determined by this program. Project Low Blow, reference 9, has also collected the much needed low level data in the atmosphere. However, a higher degree of accuracy is still required at the extremely low frequency content of the spectra. This is important because it contains the highest power and tunes more readily with the short period mode of an aircraft. Normally, the amount of power contained in the spectra above 3 cps is minimum and does not require a complete description of the elastic modes beyond this value.

Much of the gust input data has been derived by using the response data, especially at the higher altitudes. Techniques as shown in reference 3 provide a description of the statistical properties of the turbulent gust velocities by using experimental C.G. acceleration response data and the transfer functions of the airplane. This method of obtaining gust design criteria depends on the proper use of transfer functions. An estimate of the probability density of vertical gusts is obtained by using the swept bomber elastic and rigid transfer functions.

Assuming a population distribution for $\sigma_{\Delta n_1}$ the distribution for $f\left(\sigma_{\Delta n}\right)$ is

$$f\left(\sigma_{\Delta n}\right) = \frac{1}{a}\sqrt{\frac{2}{\pi}} \; e^{-\frac{\sigma_{\Delta n}^{2}}{2a^{2}}}$$

Using a = .135 g's which is a representative B-66 load factor in flights through low level continuous gusts, then the frequency distribution for σ_{U_v} is

$$f\left(\sigma_{U_v}\right) = \frac{1}{b}\sqrt{\frac{2}{\pi}} \; e^{-\frac{\sigma_{U_v}^{2}}{2b^{2}}}$$

$$\text{where} \qquad b = \frac{a}{A} = \frac{.135}{A}$$

$$f\left(\sigma_{U_v}\right) = \frac{A}{.135}\sqrt{\frac{2}{\pi}} \; e^{-\frac{\sigma_{U_v}^{2}}{\frac{2(.135)^{2}}{A^{2}}}}$$

The ratios A_E (Elastic Transfer Function) and A_R (Rigid Transfer Function) have been computed for the case shown in Figure 1.

$$A_E = .0238 \text{ g's/ft/sec}$$

$$A_R = .0301 \text{ g's/ft/sec}$$

The distribution for the standard deviation $\left(\sigma_{U_v}\right)$ of vertical gust velocity is shown in Figure 5 using both the rigid and elastic transfer functions for the swept bomber. For this particular case, the rigid value of Expected Standard Deviation (b_R) is 20.6 per cent less than the elastic values (b_E).

Depending on dynamic pressure, structural stiffness, etc., these differences in elastic effects may be completely reversed (reference 10).

The choice of the correct gust input is extremely important for the specific type of airplane because of the per cent of time spent in clear air turbulence and in thunderstorms vs. the total flight time.

MANEUVER INPUT

The present maneuver criteria has been established from V-G and VGH recorded data collected during flight (reference 4). In many instances, these data have not included enough of the associated operating conditions to be able to verify or properly classify design criteria. It is felt that it is not enough to indicate the class of airplane, such as fighter, bomber , transport, etc., or merely VGH data. If the individual characteristics of the airplane can be separated on the basis of

CUMULATIVE PROBABILITY DENSITY OF VERTICAL GUST

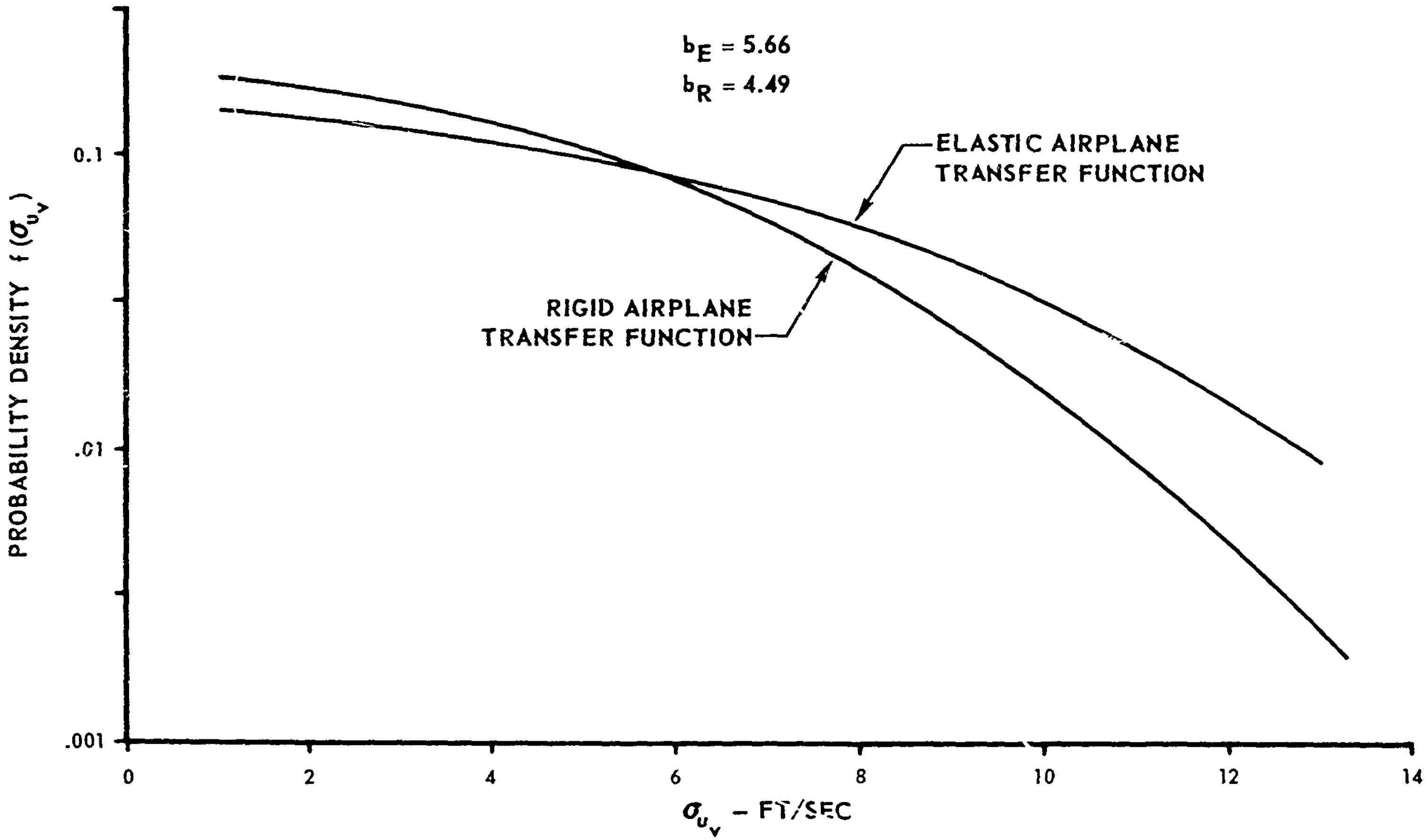


Figure 5

stability derivatives or additional parameters, then it is possible to standardize on maneuver spectra that has more significance than the present criteria. As discussed before, the maneuver criteria can be shown in power spectral and discrete form (reference 11). To date, except in a few instances, maneuver data for fatigue analysis have been used as steady-state conditions. However, the various structural components of the airplane can be critical for transient maneuvers and maneuvers plus gust. The horizontal tail load can be critical for first peak tail loads with only a small load factor recorded at the center of gravity of the airplane during a transient maneuver condition. Therefore, the loads on wing and tail surfaces should be derived from the measurements of the dynamic pressures, the three linear accelerations, the three angular accelerations, and the corresponding three angular velocities. Figure 6 is an example of individual probability curves that, if available, would allow the analyst to derive horizontal tail loads for use in fatigue damage calculations (reference 12). The proposed 8-channel Emerson recorded data can supply the necessary statistical data, including operating conditions, so that realistic maneuver design criteria can be established.

FATIGUE SPECTRA

A suggested procedure for the determination of fatigue life can be accomplished as follows:

1. Define input spectra for all environments in power spectra or discrete form.

2. Define completely the mission profile for taxi, take-off, climb, enroute, descent, landing, and taxi on landing, including corresponding weight, C.G. position, altitude, time and distance in each segment.

3. Compute the frequency of occurrence of C.G. acceleration, bending moments, shear, etc. by using the appropriate transfer functions in each case.

4. Sum the stress vs. frequency of occurrence along the profile for the structural component for a given number of mission (layers).

CONCLUSIONS

Fatigue analysis has been evolved over many years. A review of the available literature and experiments achieved during the past several years demonstrates the need for a fresh and integrated approach.

Existing data does not cover a sufficient range of conditions nor a sufficiently large statistical sample to establish firm design criteria, especially with increased vehicle speed, but does serve as a guide in the accomplishment of the structural integrity program.

A group of interested persons from industry, universities and government working together as a team can acquire knowledge from these service loads programs that can establish a firm foundation on which to base future fatigue analysis. Although this paper deals primarily with fatigue criteria and proper aircraft representation, it should be mentioned that the fatigue damage of a vehicle depends largely on the proper sequence representative of service loads.

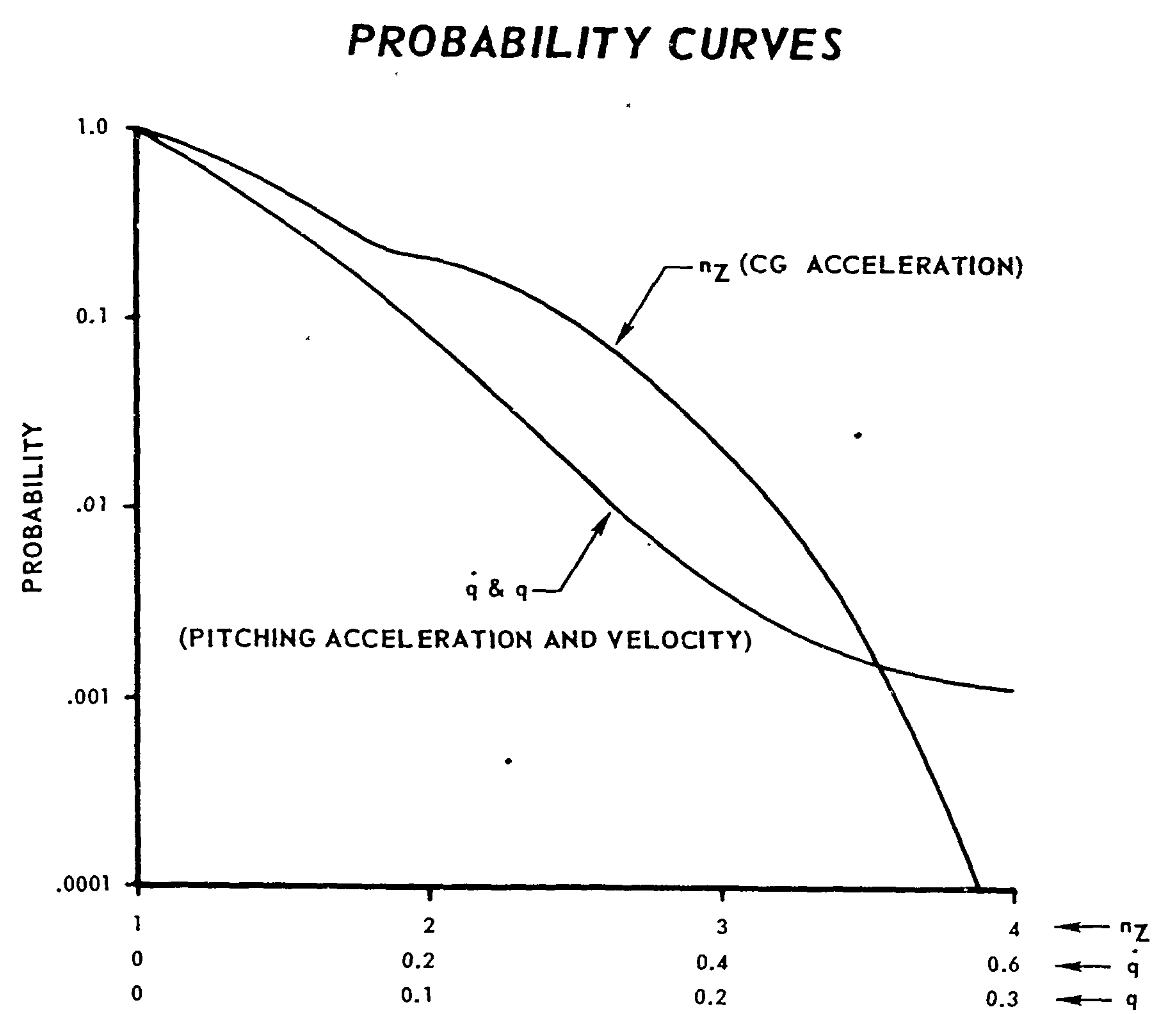

PROBABILITY CURVES
1.0
0.1
.01
.001
.0001
PROBABILITY
n_Z (CG ACCELERATION)
$\dot{q}$ & q
(PITCHING ACCELERATION AND VELOCITY)
1 2 3 4 ← n_Z
0 0.2 0.4 0.6 ← $\dot{q}$
0 0.1 0.2 0.3 ← q

Figure 6

Therefore, the service loads program, including fatigue research and development, can provide an advantageous integrated system for yielding the following information.

- Experimental verification of the input probability density using C.G. acceleration data, as well as assessment of structural degrees of freedom and spanwise averaging.

- Experimental verification of analytical transfer functions.

- Statistical data such as weight distribution, C.G. position, time, velocity, and distance in each segment of maneuver.

- Discrete and/or power spectral representation of gust spectra, maneuver spectra, and taxi spectra.

- Percentage of time in rough and normal operations such as storm and non-storm data.

- Adequacy of present design criteria.

- The necessity of a rigorous analysis in addition to good design practices.

- Basic stability and flutter data.

- Data leading to the establishment of logical threshold values.

- Selection of proper data forms that permit a simple conversion of data into fatigue input spectra.

- Further classification of fundamental input forms for various types of aircraft.

- Determination of the elastic modes that must be included, depending upon power input.

- Effect of aircraft capabilities in pitchup, stall, buffet boundary with spectra input for cutoff of Δn.

RECOMMENDATIONS

In order to derive the maximum benefits from the structural integrity program and other allied research and development studies, it is necessary to compare analytical and experimental results. The specific response should be bending moment and stress levels because this describes the true load history and strain history of the structure. Pertinent load data should be collected experimentally for comparison with analysis. In order to substantiate the comparison it is necessary to check various types of aircraft. The following procedures are recommended:

1. Compute the output response for bending moment and stresses at various stations along the wing by the three input representations: discrete (assumed shape), power spectral, and discrete time history.

2. Obtain experimental values of wing bending moments and stresses at the same stations computed analytically. Calibrate the wing for a limited flight test program. All pertinent instrumentation, including the 8-channel Emerson recorder, should be recorded simultaneously.

3. Check the accuracy of analytical transfer functions with experimental values in order to reduce the errors in the comparison between analytical and experimental results.

4. Compare analytical and experimental values to help determine the degree of sophistication required for the method of input representation and fatigue analysis.

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PREDICTION OF SONIC EXPOSURE HISTORIES

by

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Recognizing that the vibration field in present and projected aircraft and missiles is excited primarily by the surrounding acoustic field, it is of the utmost importance to be able to predict the exposure field as a function of the flight profile. The acoustic excitation is caused by the propulsion noise and by turbulence in the aerodynamic boundary layer. This paper discusses the relationships between vehicle and engine parameters and the resulting acoustical field with reference to the development of methods of predicting sonic exposure, and to areas which require additional experimental or theoretical work.

INTRODUCTION

The advances in aircraft performance since the introduction of the turbojet engine, together with the more recent advances in missiles and space vehicles resulting from development of the rocket motor, have been accompanied by a significant increase in noise. This increase of noise, coupled with the necessity of minimizing vehicle weight to maximize performance has, in several cases, resulted in vehicle malfunctions and/or material failures. Fortunately, the majority of the actual material failures resulting from acoustical loading have occurred in aircraft skin where they could be discovered by visual inspection and corrected prior to the development of a catastrophic failure. However, the ensuing requirement for extreme thoroughness in inspection and the concurrent design, evaluation and application of modifications which will successfully withstand the actual environment has been expensive. Furthermore, the malfunction of electronic or other equipment aboard a missile, has sometimes resulted in complete failure of the missile system, which is not only expensive in dollars, but also in time.

Consequently, considerable effort has been devoted to the development of methods which will enable a designer, first to predict the sonic loading for a particular aircraft and then to utilize this prediction in the design and proof test of aircraft skins and components. These methods have resulted in a considerable advance in the state of the art. However, they have been based necessarily on empirical correlations. Further, their accuracy is not of the same order of magnitude which is normally considered for structural engineering calculations. Note that the ±3 db stated by the acoustician actually represents a factor of 2 in pressure or acceleration.

This paper will discuss briefly the major sources of sonic loading and give some examples of the correlations which may be used for predictions in the early design stage.

MAJOR ACOUSTICAL SOURCES

The major sources of sonic exposure for vehicles operating in the earth's atmosphere are the vehicle's propulsion system, boundary layer, and wake. With the exception of propeller noise, turbojet compressor noise, and certain types of boundary layer phenomena, these major sources are all characterized by pressure and velocity amplitudes which fluctuate randomly with respect to both frequency and time. In addition, these sources all result from turbulent flow, although the transfer of turbulent energy from the flow to the vehicle differs for the various sources.

The difficulties in developing accurate prediction methods are directly related to our ignorance of turbulent phenomena. Although considerable theoretical and experimental work in this area has been accomplished in the past eight years there is, as yet, no generalized and comprehensive explanation for many of the various generating mechanisms which result in sonic loading. For this reason, the models used for acoustical prediction are based primarily upon empirical fit of data to various convenient characteristic flow parameters. Since the resulting empirical relationships are not necessarily supported by theory, it is often dangerous to predict the noise of sources where one or more flow parameters significantly differ from those studied in the empirical derivations.

PROPULSION NOISE

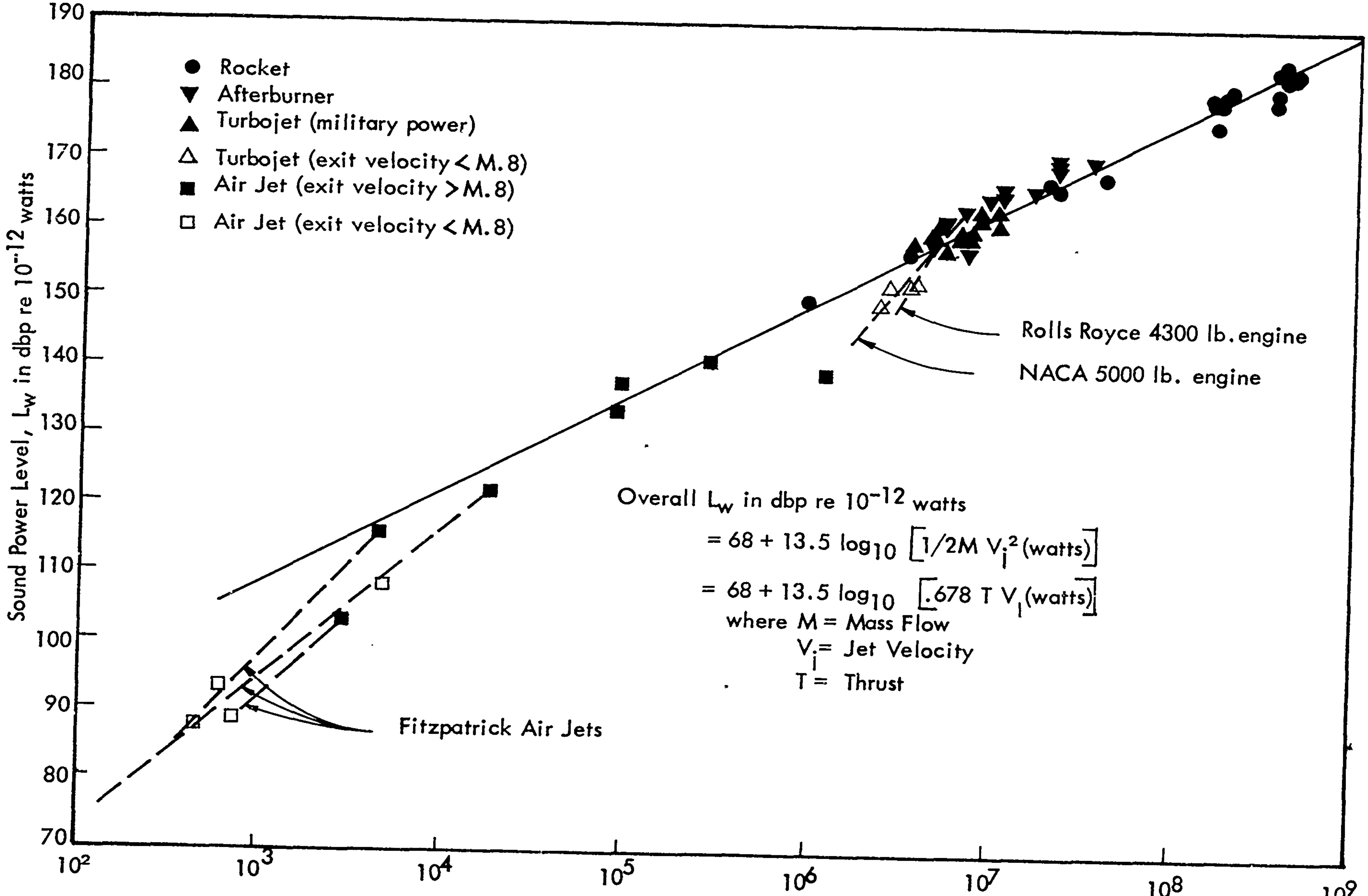
The most powerful acoustical sources developed by man are the propulsion systems for modern aircraft and missiles. Current propulsion systems at maximum thrust produce acoustical power in the range of 100,000 to 3,000,000 watts. This range includes afterburning J-57 turbojet engines which produce approximately 100,000 watts, 5000 HP turboprops with supersonic prop tips which give the same order of acoustical power, and 130,000 lb. rockets which generate approximately 3,000,000 watts of acoustical power. Fortunately for the designer, the sonic loading from a propeller is well understood by theory and may be readily calculated for individual configurations (see Ref. 1 for a review and additional references). Consequently, this discussion will consider only the less well understood area of turbojet and rocket noise.

The theoretical relationships between jet flow parameters and noise generation have been rigorously derived by Lighthill (Refs. 2 and 3). These relationships indicate that the total acoustical power for subsonic flow should increase with at least the 8th power of a characteristic jet velocity. Callagan of NASA (Ref. 4) and others have demonstrated that an 8th power of velocity is a valid parameter for both air jets and turbojets and thus could be used for prediction. However, application of this relationship to supersonic jets or rockets leads to unrealistically high values of acoustic power. Consequently, the WADC conducted an extensive survey of rocket and jet noise sources (Refs. 5 and 6) to obtain the basic data required to give a better understanding of the noise generation for these important propulsion systems.

Figure 1, given in the 1956 Fall Acoustical Society of America meeting, summarizes the total acoustical power measured by several workers for jet flows which were primarily either sonic or supersonic at the nozzle. It is clear that the rocket, turbojet and model jet data all fall along the empirical curve given by the formula in the figure whenever the initial jet velocity exceeds approximately Mach.8. However, when the initial velocity is less than Mach .8 the acoustical power is less than indicated by the curve. This results from the fact that the noise produced by the subsonic jets is related to approximately the 8th power of velocity, whereas the noise of the sonic or supersonic jet appears to be related to a much lower power of velocity, approximately the 4th. This is indeed fortunate for the development of lightweight missile structure.

The frequency spectrum of the power generated by jet flows has been shown to be random with a relatively broad maximum region. Several reports, including Refs. 1, 4, and 6, have related the power spectrum to a non-dimensional frequency parameter given by the flow Strouhal number. However, the maximum energy for the rocket occurs at a lower Strouhal number than the maximum energy for a turbojet, as shown in Ref. 6. Recently, a more general relationship was found (Ref. 7) which enables the empirical comparison of the frequency spectra for all jet flows thus far studied. An example of this relationship is given in Figure 2. Here, the spectrum is non-dimensionalized by a modified Strouhal number (S') obtained by utilizing a characteristic jet velocity and diameter, together with the ratio of the critical speed of sound in the flow to that in the surrounding atmosphere. As can be seen, the power spectral density increases with the square of frequency below an S' of approximately .1 and after the relatively broad maximum region decreases with the inverse square of frequency. The relationship clearly illustrates that an increase in characteristic diameter will lower the frequency of maximum acoustic energy, whereas an increase in velocity will increase the frequency of maximum energy.

In order to predict the sonic loading for a particular configuration, it is necessary to know the spacial distribution of the noise, as well as the total power and spectra. The noise produced by jet flows is generated by turbulence in the mixing regions of the jet. The maximum turbulent energy has been found by Laurance of NASA (Ref. 8) to be at the tip of the jet core which is the region of maximum momentum transfer. Corresponding detailed near field noise studies by Howes and Callagan of NASA (Ref. 9) have demonstrated that the maximum noise emanates from the same region. Further, these studies with subsonic


Rocket
Afterburner
Turbojet (military power)
Turbojet (exit velocity < M.8)
Air Jet (exit velocity > M.8)
Air Jet (exit velocity < M.8)
Rolls Royce 4300 lb. engine
NACA 5000 lb. engine
Overall L_w in dbp re 10^{-12} watts
$= 68 + 13.5 \log_{10} \left[1/2 M V_j^2 \text{(watts)}\right]$
$= 68 + 13.5 \log_{10} \left[.678\ T\ V_j \text{(watts)}\right]$
where M = Mass Flow
V_j = Jet Velocity
T = Thrust
Fitzpatrick Air Jets
Sound Power Level, L_w in dbp re 10^{-12} watts
TOTAL JET STREAM POWER WATTS
190
180
170
160
150
140
130
120
110
100
90
80
70
10^2
10^3
10^4
10^5
10^6
10^7
10^8
10^9
Figure 1

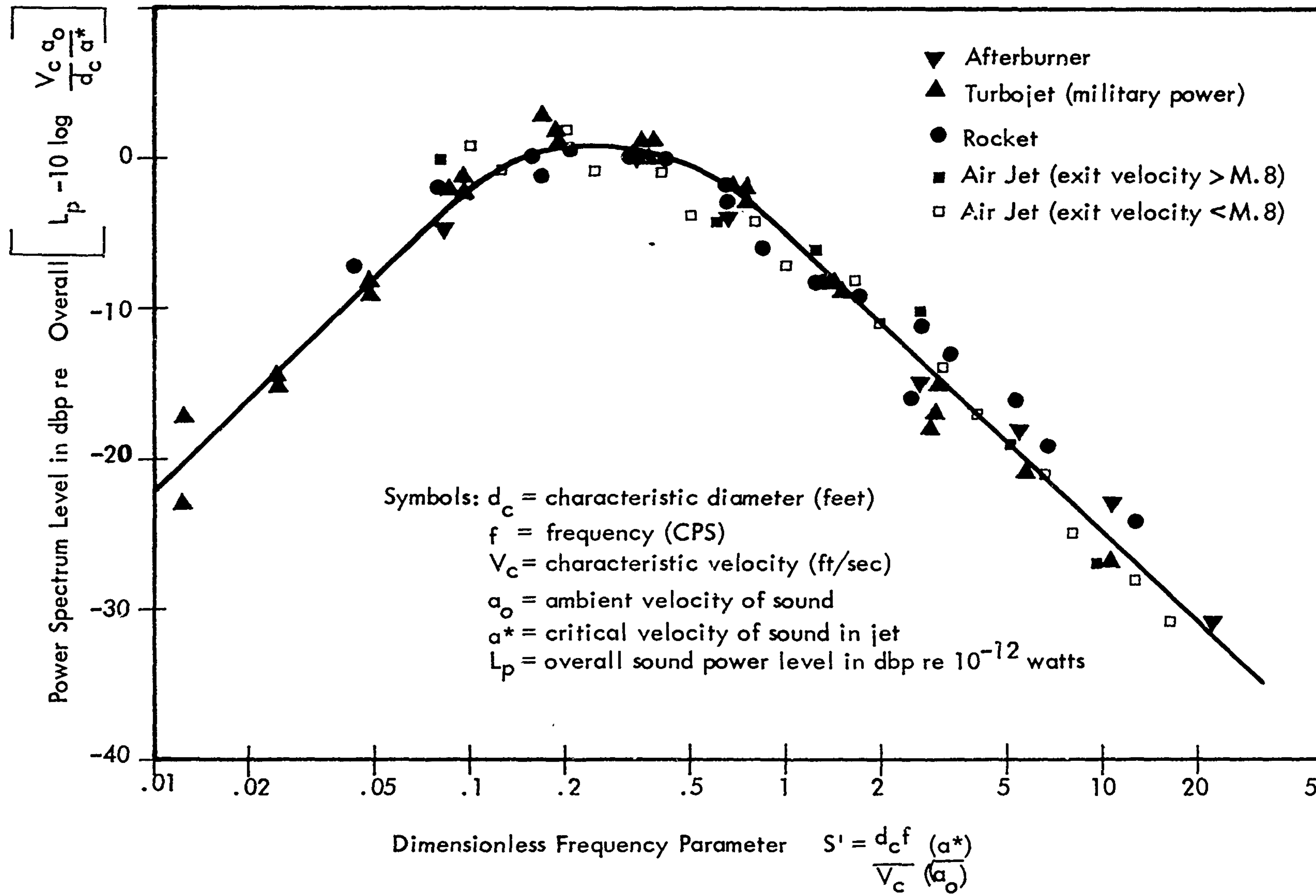

Afterburner
Turbojet (military power)
Rocket
Air Jet (exit velocity > M.8)
Air Jet (exit velocity < M.8)
Symbols: d_c = characteristic diameter (feet)
f = frequency (CPS)
V_c = characteristic velocity (ft/sec)
a_o = ambient velocity of sound
a^* = critical velocity of sound in jet
L_p = overall sound power level in dbp re 10^{-12} watts
Power Spectrum Level in dbp re Overall $\left[L_p - 10 \log \frac{V_c}{d_c} \frac{a_o}{a^*}\right]$
0
-10
-20
-30
-40
.01
.02
.05
.1
.2
.5
1
2
5
10
20
50
Dimensionless Frequency Parameter $S' = \frac{d_c f}{V_c} \left(\frac{a^*}{a_o}\right)$

Figure 2

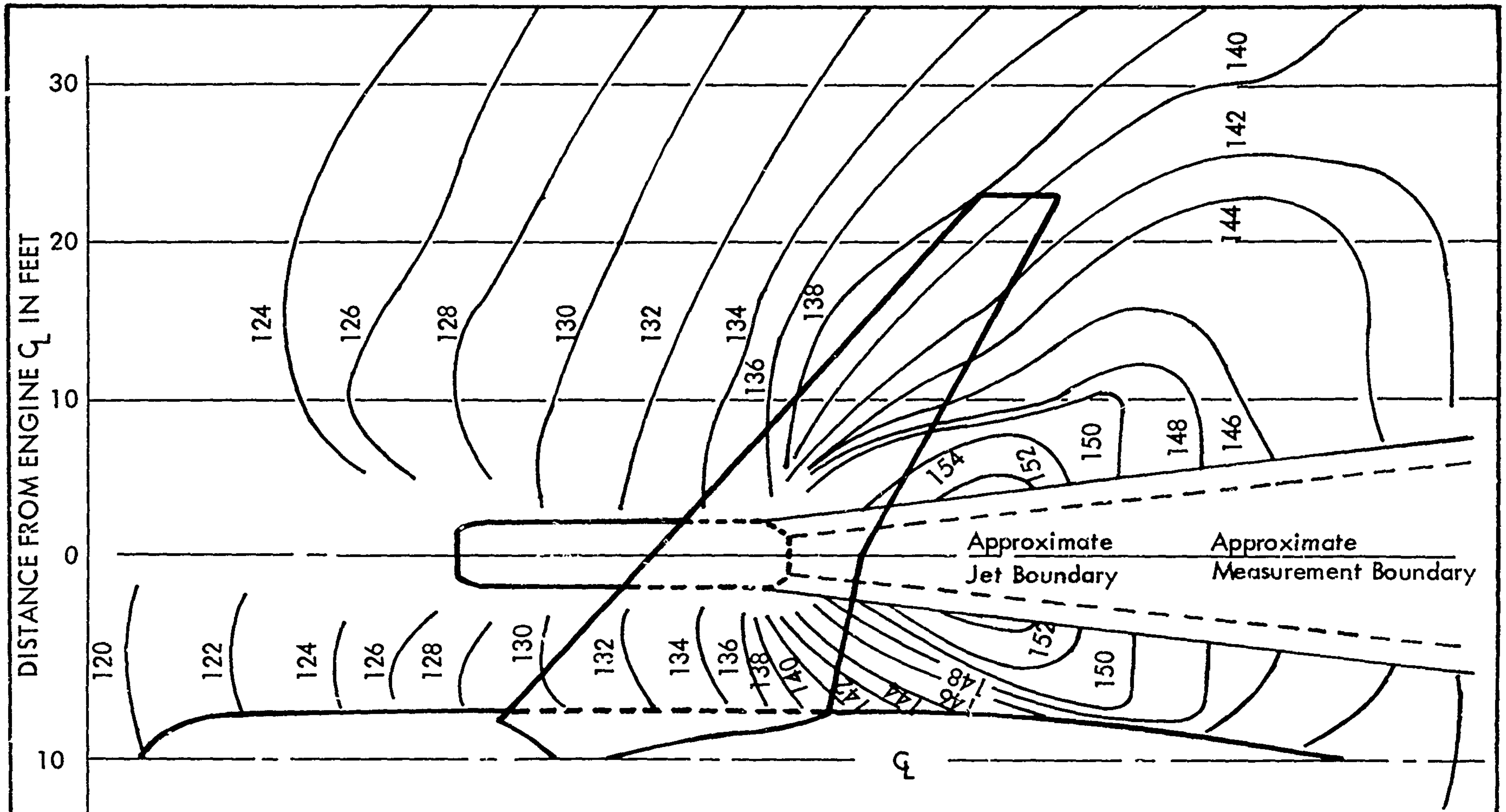
or sonic jets also indicate that the high frequency turbulence and noise are generated very close to the nozzle, whereas the lowest frequency noise is generated 10 to 20 jet diameters downstream. In addition, the data of Ref. 9 show a good correlation of longitudinal distribution of frequency maxima between the turbulence and noise data and enable fairly accurate calculation of the source location for the turbojet noise.

An example of the spacial distribution of acoustical energy is given in Figure 3 where contours of equal overall sound pressure level (SPL), obtained from a detailed near field noise survey of a bare J-71 engine by the WADC, are overlaid on a plan view of a B-66 aircraft. This example illustrates the downstream position of the maximum sound pressure levels just discussed. Note that the measurements did not extend to the boundary of the jet as did those of Ref. 9 and therefore do not give the absolute maximum levels at the boundary. The figure also shows that the characteristic tendency of the jet to radiate the greatest proportion of its sound energy toward an angle of 140° to 150° relative to its intake, again emphasizing the desirability of placing the engines at the aftermost point on the vehicle to minimize sonic exposure.

Figure 4 gives an example of the prediction of noise alongside the B-66 fuselage from the J-71 data. It is clear from the data that the levels forward of the engine are greater on the aircraft than in the free field because of the multiple reflection surfaces (ground, fuselage, wing and pod). In addition, some discrepancy is noted aft of the nozzle. However, it is considered that predictions of near field turbojet noise can be made within at least the accuracy shown here, by utilizing this type of data together with the NASA data of Ref. 9 with an appropriate non-dimensionalized distance parameter expressed in exit nozzle diameters.

Data regarding the near field noise of rockets is much more limited than those regarding turbojet noise. However, the available data (Refs. 10, 11 and 12) clearly illustrate that the maximum SPLs are found much farther downstream than those of the turbojet. Further, the study shows that these maximum levels occur at the tip of the supersonic core of the rocket flow. Since the flow downstream of the supersonic core is similar to the normal subsonic or sonic turbojet, the available near field data strongly suggest that the noise from a rocket is generated in the subsonic turbulent portions of the flow. This assumption has been used in Ref. 7 to derive the characteristic velocity and diameter of the rocket flow where the characteristic velocity (v_c) is taken as the critical sonic velocity in the flow and the characteristic diameter (d_c) is taken as the constant mass flow diameter for this velocity.

Unfortunately, no study to date has given the distribution of frequency maxima as a function of longitudinal distance along the flow for hot rockets. However, because the distance for the maximum overall noise source is relatively far downstream and because the area of interest for sonic exposure lies forward of the nozzle, the error is not too great if the source location for all frequencies is approximated as the tip of the supersonic core. From the data of Ref. 13 the distance (S) from the nozzle to the tip of the core



PLAN VIEW OF B-66 FUSELAGE WITH J-71
OVERALL SOUND PRESSURE LEVEL CONTOURS
IN db re .0002 DYNE/CM^2

Figure 3

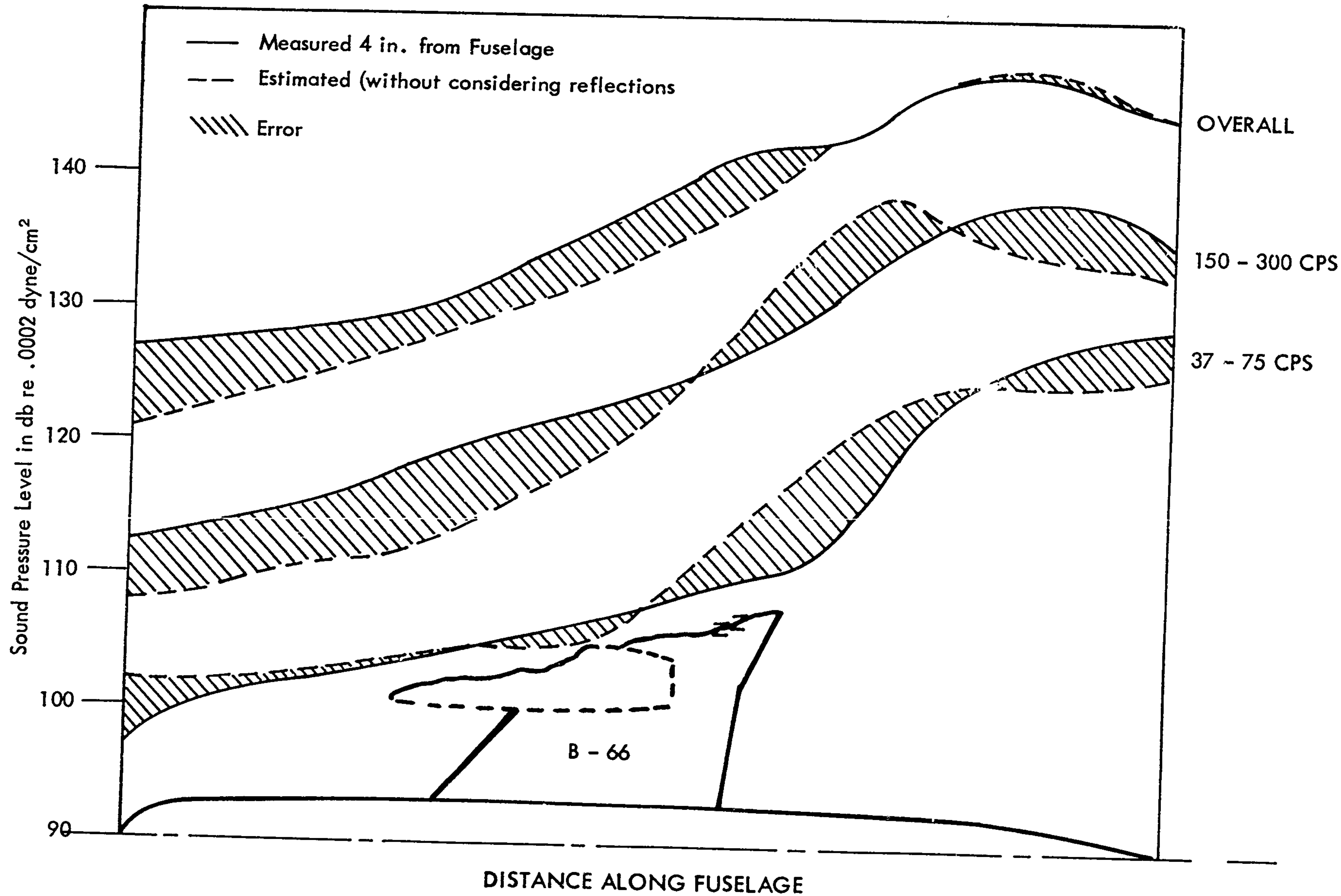

Measured 4 in. from Fuselage
Estimated (without considering reflections
Error
OVERALL
150 - 300 CPS
37 - 75 CPS
140
130
120
110
100
90
Sound Pressure Level in db re .0002 dyne/cm^2
B - 66
DISTANCE ALONG FUSELAGE

Figure 4

is given approximately by:

$$S = 6.5\, d_e \left[1+(M_e-1)^2\right] \qquad \text{Equation 1.}$$

where d_e is the nozzle exit diameter and
M_e is the nozzle exit Mach number.

With this approximation, the overall SPL forward of the nozzle along the rocket's axis (0°) may be found by:

$$\text{OA SPL}_{0°} = L_p - 20 \log x - 10 \qquad \text{Equation 2.}$$

where L_p is the total acoustic power from Figure 1 and x is the distance from the source to the point of interest.

This result will be approximately correct as long as the distance x is large enough to assure that the point is in the far field, where the SPL decreases in accordance with the inverse square spreading loss, 6 db per double distance. As can be seen in Figure 5, the far field forward of the flow is reached at a distance of about 10 wavelengths. When the distance is less than the 10 wavelengths, the SPL decreases at a greater rate than 6 db/ double distance and an appropriate correction must be added to Equation 2.

The spectrum forward of the nozzle may be found from Figure 6, which applies to single nozzle rockets. This spectrum, although similar in form to the power spectrum of Figure 2, includes the directivity effects for the 0° radial and is thus specialized for the forward position. It is noted that the spectrum for rockets with multiple nozzles will be somewhat broader than that for the single nozzle. This results from the larger characteristic diameter of the multiple nozzle (Ref. 14) and, in fact, the function of multiple rocket nozzles can be considered similar to the multitube noise suppression nozzles in use on the commercial jet airliners.

It should be noted that these examples of predictive methods do not include the screech noise sometimes found in afterburning engines, or the results of reignition aft of the nozzle. This latter phenomenon, reported in Ref. 6, was responsible for an increase of approximately 10 db for positions forward of the nozzle (Ref. 15) over the levels of an equivalent non-reigniting rocket. The occurance of these unpredictable phenomena clearly recommends that noise measurements be made on each specific type of motor at the earliest practicable stage of system development.

It is well known that the noise of jets and rockets decreases with increasing flight speed. Figure 7 illustrates the decrease which would be expected forward of the nozzle for these two cases based on two considerations, decreased relative characteristic jet flow velocity and the increased distance between source and vehicle positions forward of the

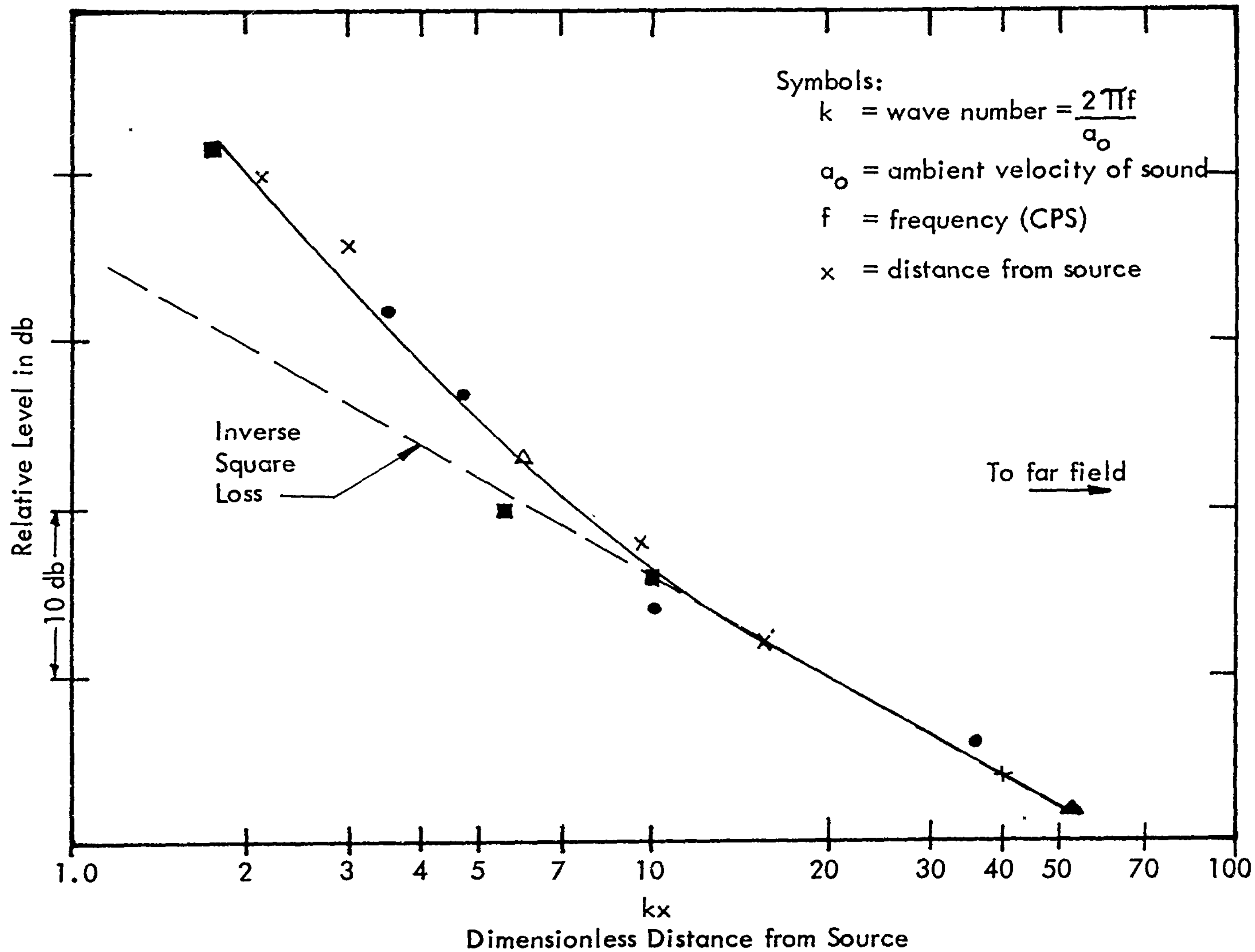

Symbols:
k = wave number = 2πf / a_o
a_o = ambient velocity of sound
f = frequency (CPS)
x = distance from source
Inverse Square Loss
To far field
Relative Level in db
10 db
1.0
2
3
4
5
7
10
20
30
40
50
70
100
kx
Dimensionless Distance from Source

Figure 5

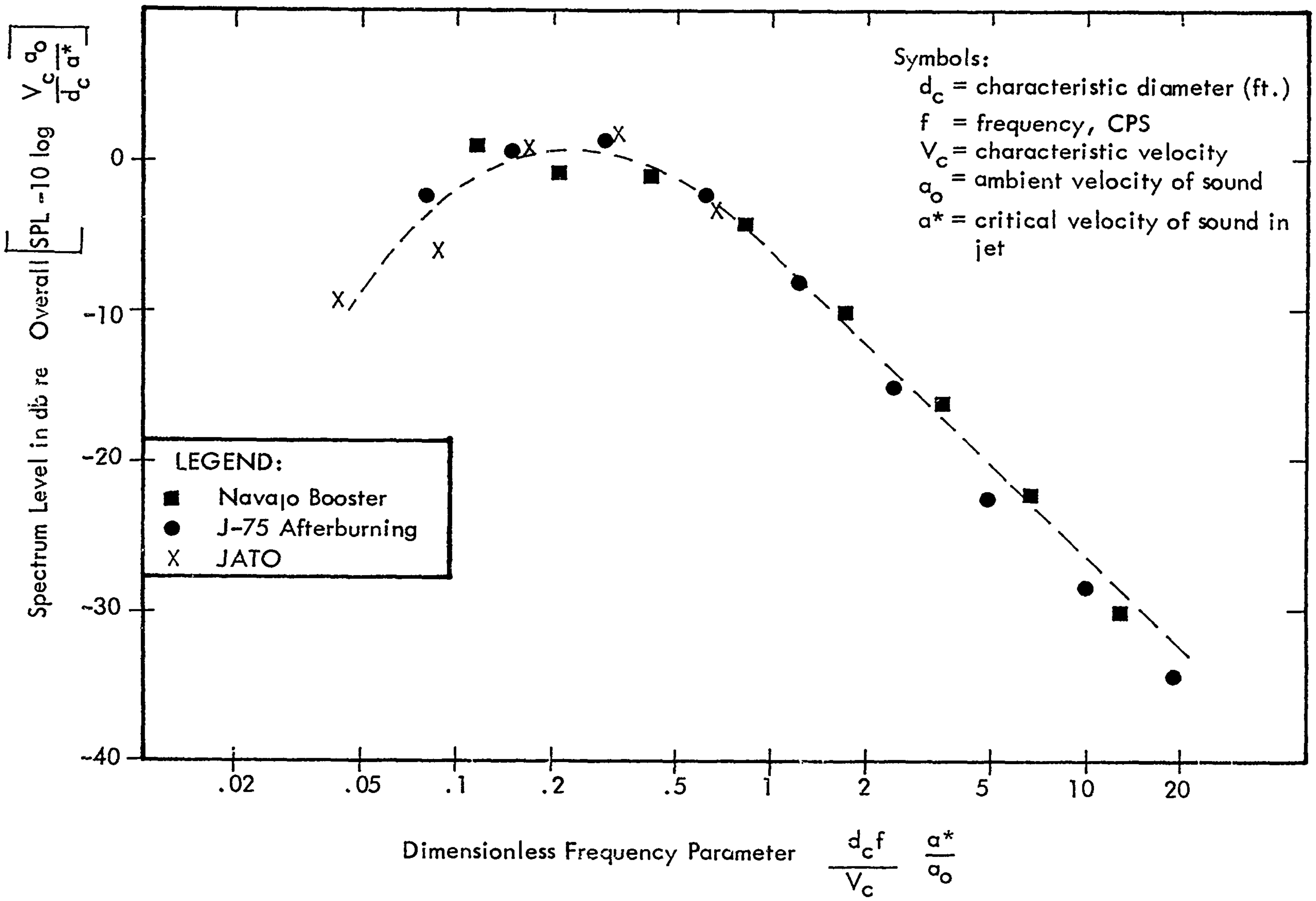
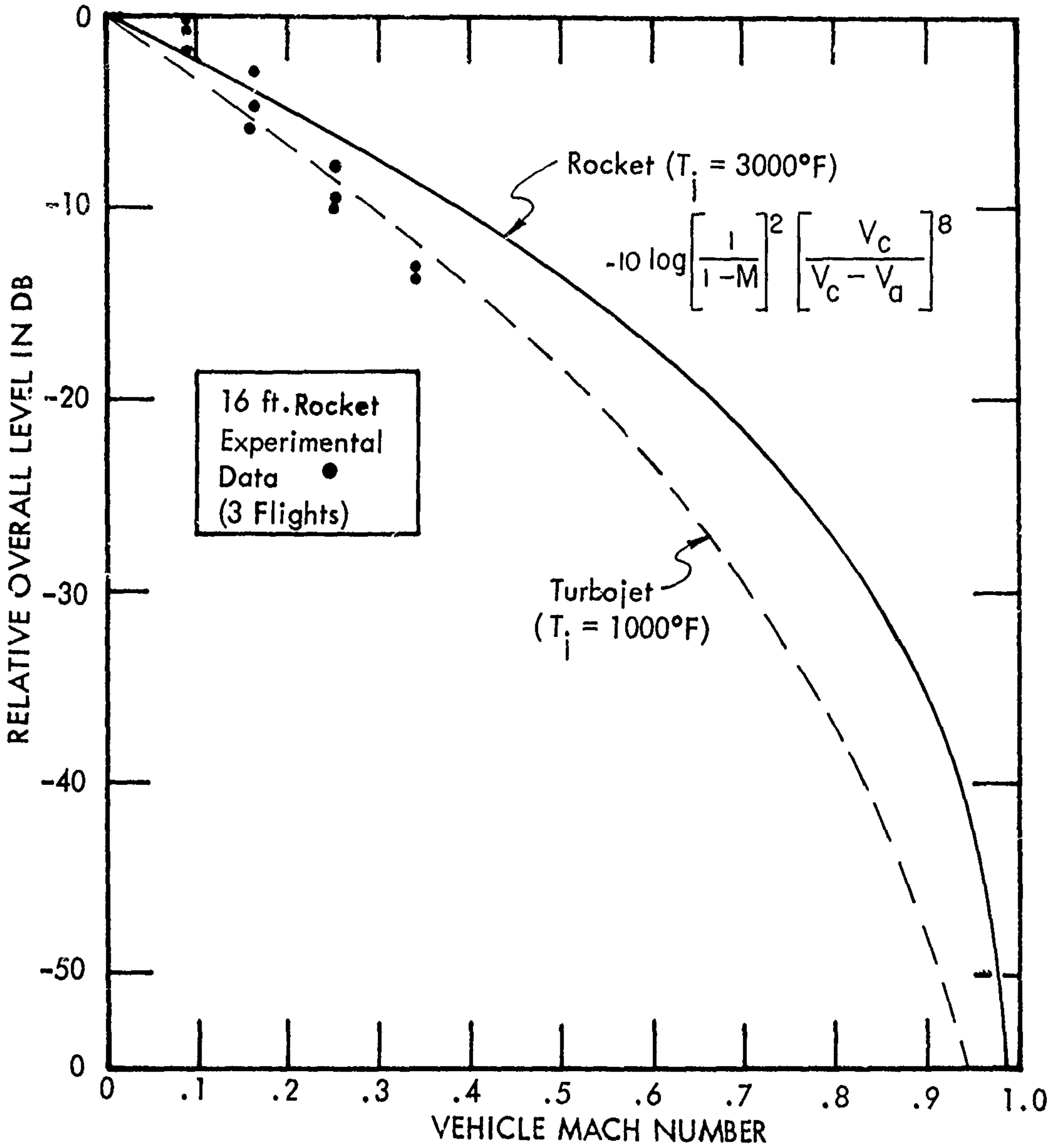

Symbols:
d_c = characteristic diameter (ft.)
f = frequency, CPS
V_c = characteristic velocity
a_o = ambient velocity of sound
a^* = critical velocity of sound in jet
LEGEND:
Navajo Booster
J-75 Afterburning
X JATO
Spectrum Level in db re Overall $\left[SPL - 10 \log \frac{V_c}{d_c} \frac{a_o}{a^*}\right]$
0
-10
-20
-30
-40
.02
.05
.1
.2
.5
1
2
5
10
20
Dimensionless Frequency Parameter $\frac{d_c f}{V_c} \frac{a^*}{a_o}$

Figure 6



Rocket (T_j = 3000°F)
$-10 \log \left[\frac{1}{1-M}\right]^2 \left[\frac{V_c}{V_c - V_a}\right]^8$
16 ft. Rocket Experimental Data ● (3 Flights)
Turbojet (T_j = 1000°F)
RELATIVE OVERALL LEVEL IN DB
0
-10
-20
-30
-40
-50
0
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
VEHICLE MACH NUMBER

Figure 7

nozzle. The figure also gives some experimental data from a small missile which indicates a greater decrease of noise with forward speed than indicated by the calculated curves.

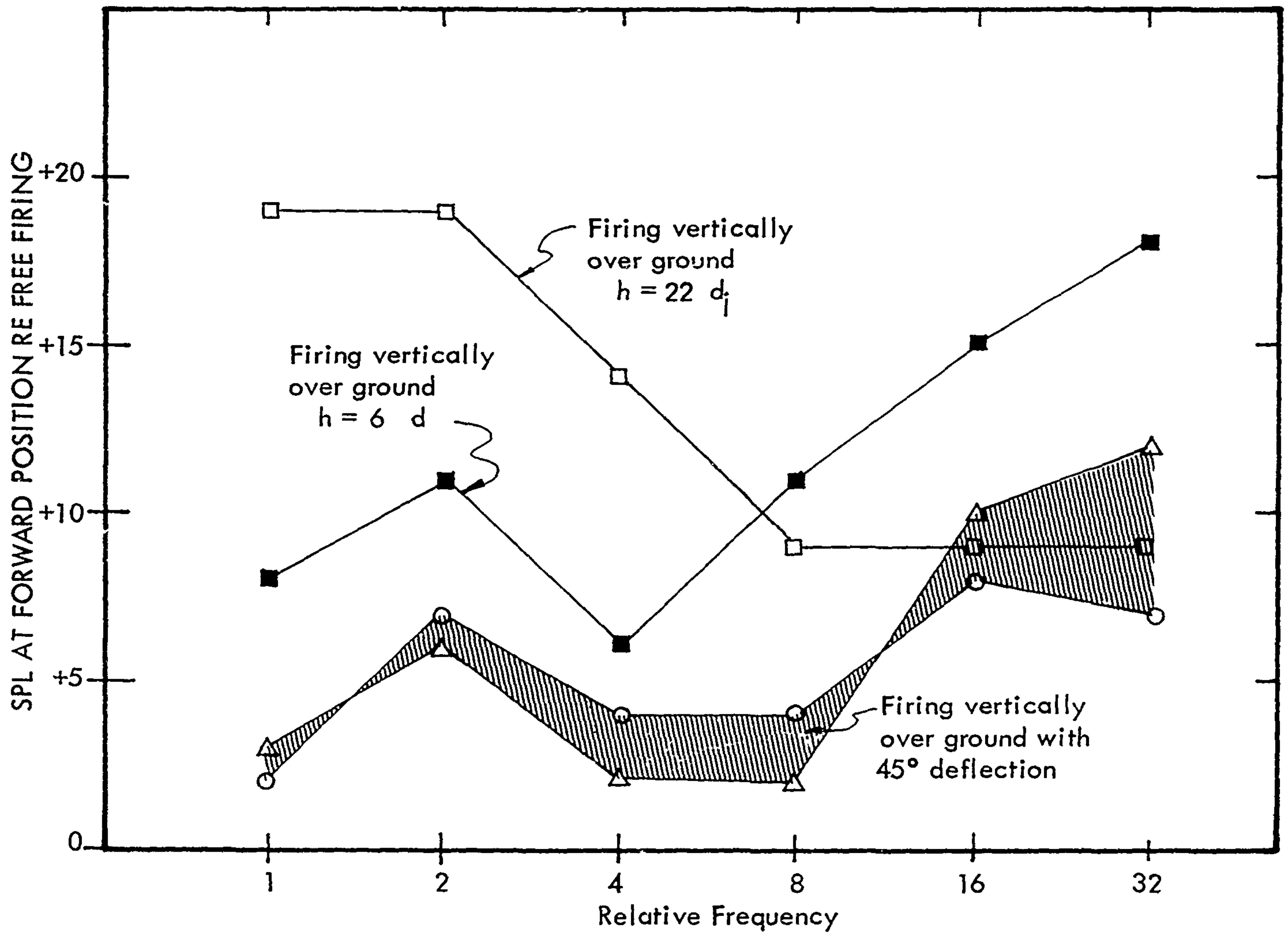
A recent extensive series of experiments has been made to determine the effect of the several launch configurations on the noise produced by rockets. These experiments were performed by the WADC on JATO booster rockets and the results are scheduled for early publication. Figure 8 gives a few of the results forward of the nozzle in the region of a simulated nose cone. The data strikingly indicate the increase of noise forward of the nozzle during launch relative to the noise which would exist for a free rocket stream. Paradoxically, the intersection of the jet with a ground plane or deflection device actually results in a reduction of total acoustical power relative to that produced by the free flow. However, since the rocket's stream flowing over the ground radiates considerably more energy vertically near the upright missile than is normally radiated from the free flow, the noise at forward positions tend to increase. These results demonstrate that the maximum sonic environment for missiles resulting from the propulsion system occurs during launch and that the actual environment is extremely dependent upon configuration. Hence, model experiments with simulated rocket flows (temperature, Mach number, etc.) are necessary to predict launch environments. In addition, attention to the launch configuration can result in designs which minimize the levels during launch, as indicated by a study of the complete WADC data.

BOUNDARY LAYER AND WAKE NOISE

As the vehicle's flight velocity increases and the noise from the propulsion system is reduced, another turbulent noise source becomes important. This is the turbojet boundary layer which grows along the exterior surface. Although considerable data is available for subsonic boundary layers, little is available for the supersonic flight velocities of current interest.

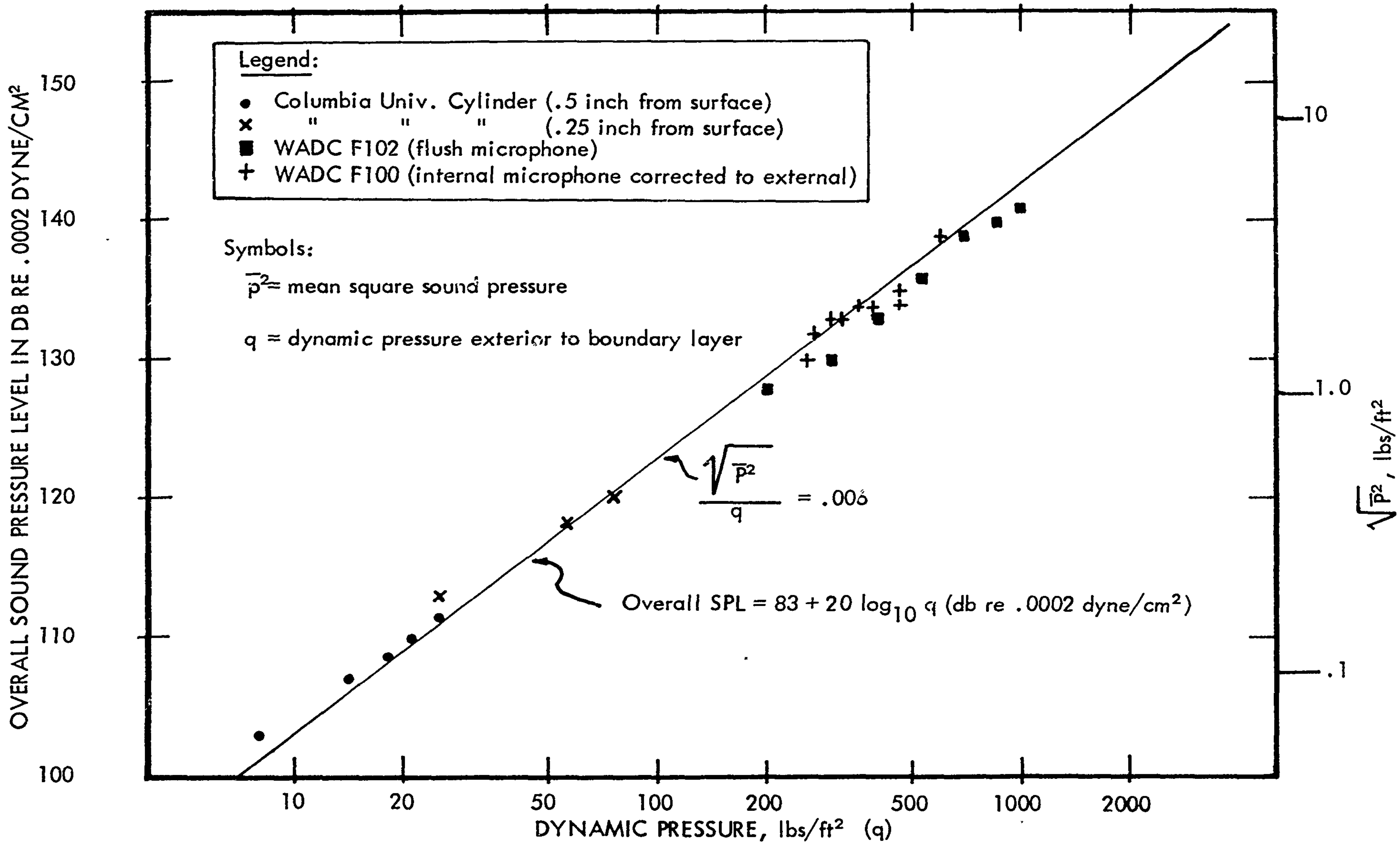
Figure 9 illustrates an empirical relationship between free stream dynamic pressure and the overall external SPL measured by various experimenters (Refs. 16, 17 and 18). These data as well as that of other investigators (Refs. 19 and 20) give a value for the ratio of RMS overall sound pressure to dynamic pressure of approximately .006.

There is much more agreement on the previous relationship than can be found when examining the measured frequency spectra. Figure 10 give a normalized frequency spectra based on the data used in Figure 9. As can be seen, the data covers several decades in frequency and does not exhibit any tendency toward decreasing at the lower frequencies. Further, the nature of the non-dimensional frequency parameter indicates a more rapid increase in the high frequency levels proportional to the overall as the vehicle speed increases until the vehicle's speed is sufficient to shift the non-dimensional frequency parameter to the left side of the curve. Note that the frequency parameter also indicates a dependency upon boundary layer thickness so that the lower frequencies would be more



SPL AT FORWARD POSITION RE FREE FIRING
+20
+15
+10
+5
0
1
2
4
8
16
32
Relative Frequency
Firing vertically
over ground
h = 22 d
Firing vertically
over ground
h = 6 d
Firing vertically
over ground with
45° deflection

Figure 8



Legend:
Columbia Univ. Cylinder (.5 inch from surface)
" " " (.25 inch from surface)
WADC F102 (flush microphone)
WADC F100 (internal microphone corrected to external)
Symbols:
$\overline{p^2}$ = mean square sound pressure
q = dynamic pressure exterior to boundary layer
$\frac{\sqrt{\overline{p^2}}}{q} = .006$
Overall SPL = 83 + 20 $\log_{10} q$ (db re .0002 dyne/cm²)
OVERALL SOUND PRESSURE LEVEL IN DB RE .0002 DYNE/CM²
DYNAMIC PRESSURE, lbs/ft² (q)
$\sqrt{\overline{p^2}}$, lbs/ft²
150
140
130
120
110
100
10
20
50
100
200
500
1000
2000
10
1.0
.1

Figure 9

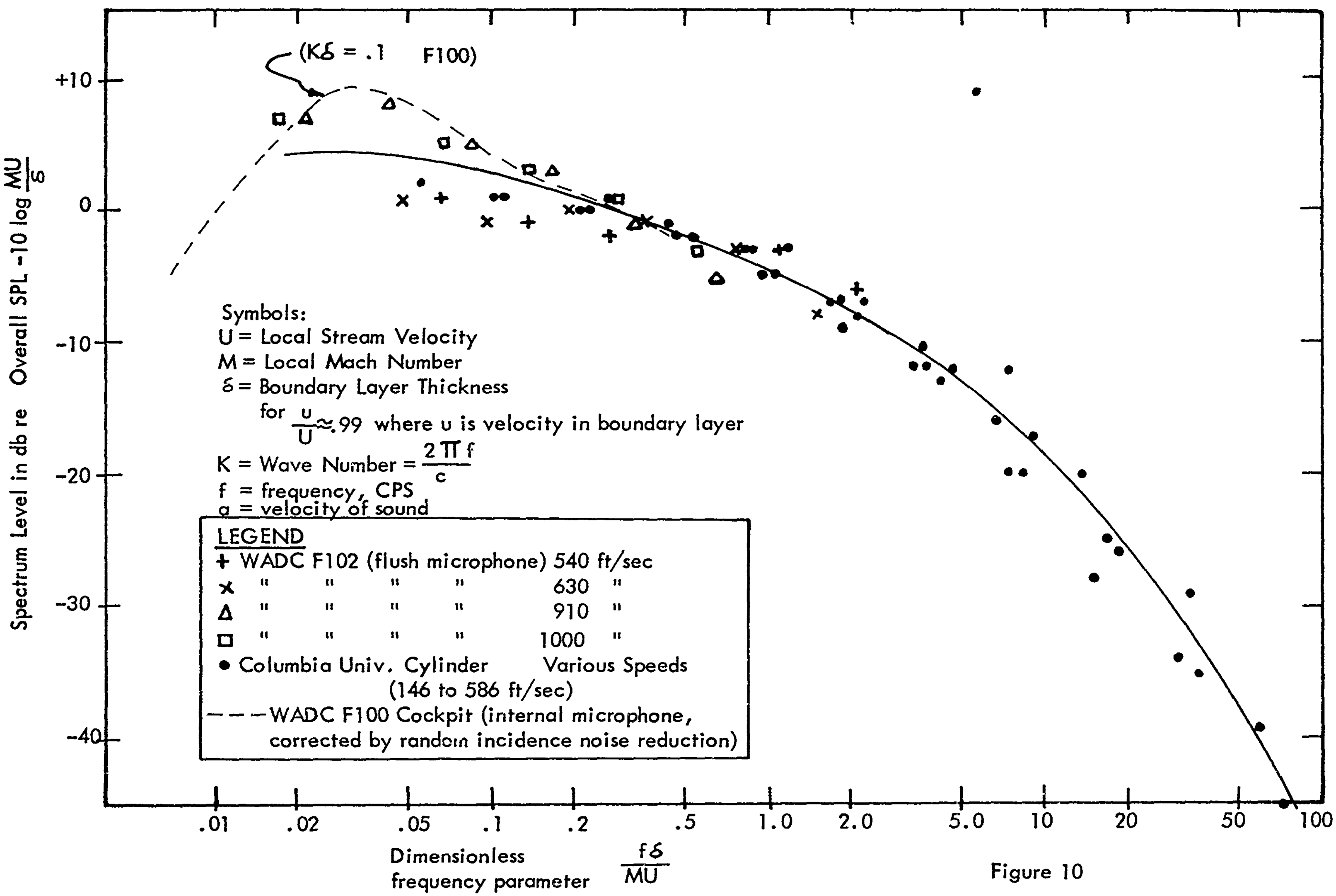

(Kδ = .1 F100)
Spectrum Level in db re Overall SPL -10 log $\frac{MU}{\delta}$
+10
0
-10
-20
-30
-40
Symbols:
U = Local Stream Velocity
M = Local Mach Number
δ = Boundary Layer Thickness
for $\frac{u}{U} \approx .99$ where u is velocity in boundary layer
K = Wave Number = $\frac{2\pi f}{c}$
f = frequency, CPS
c = velocity of sound
LEGEND
+ WADC F102 (flush microphone) 540 ft/sec
× " " " " 630 "
△ " " " " 910 "
□ " " " " 1000 "
● Columbia Univ. Cylinder Various Speeds (146 to 586 ft/sec)
- - - WADC F100 Cockpit (internal microphone, corrected by random incidence noise reduction)
.01
.02
.05
.1
.2
.5
1.0
2.0
5.0
10
20
50
100
Dimensionless frequency parameter $\frac{f\delta}{MU}$

Figure 10

prominent at aft vehicle stations than at forward stations. It is noted that although the wind tunnel data of Refs. 19 and 20 (not shown in the figure) are similar in the low frequency region, they decrease more rapidly in the high frequency region than the data of Figure 10. This conflict makes it clear that the understanding of the behavior of boundary layer spectrum is very limited at this time and that Figure 10 must be considered very preliminary.

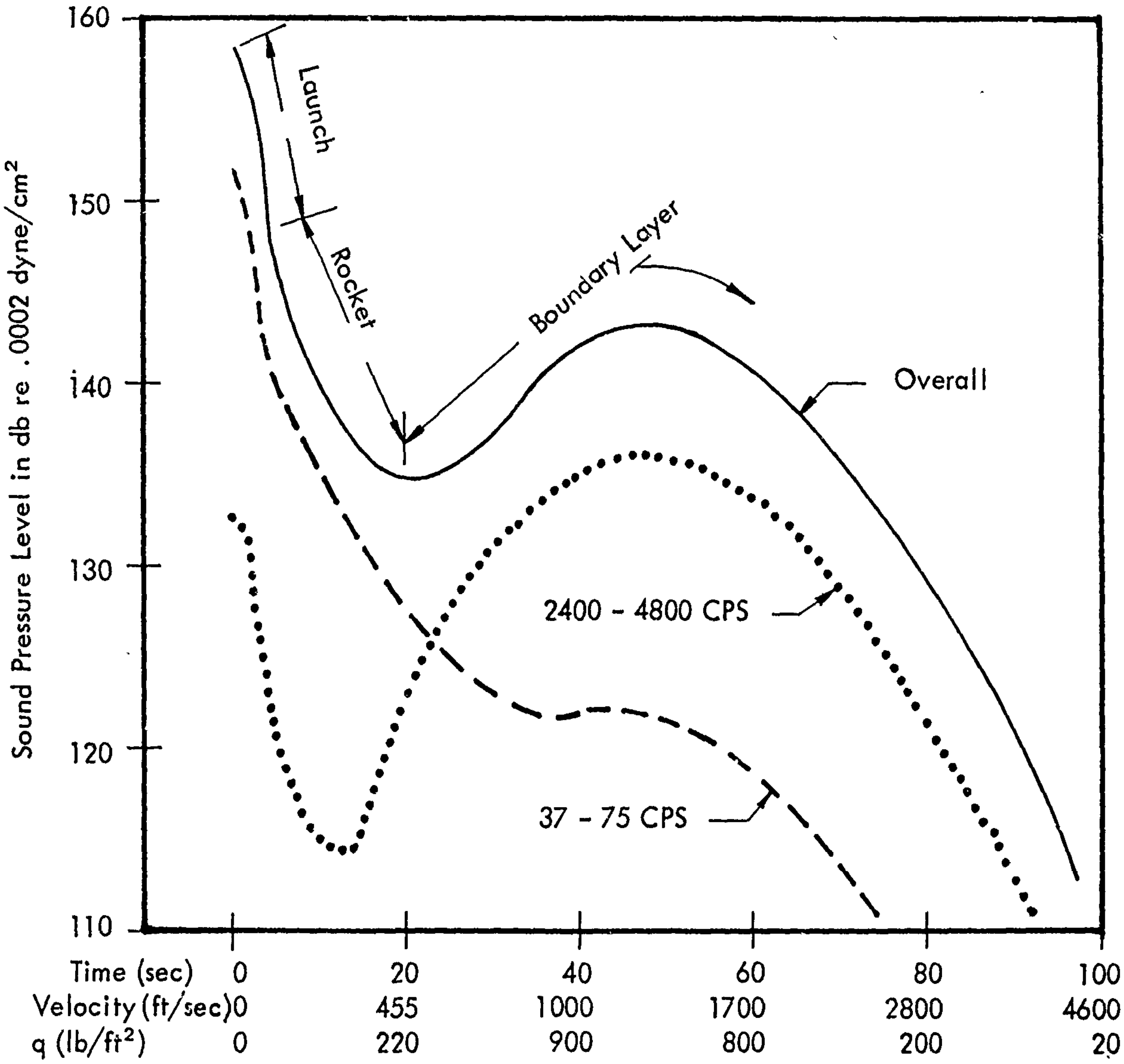
Another important factor must be considered when utilizing external measurements of boundary layer noise. This is the fact that the boundary layer sound pressures measured are not sound in the conventional sense, but rather are turbulent fluctuations which are correlated for only a few boundary layers downstream. Hence, in the low frequency range, the fluctuating pressure over a surface is in phase for only a fraction of a wavelength. Consequently, the transfer of low frequency energy from the boundary layer to the surface is expected to be much less than in the normal acoustical case (Ref. 21). This may account for the low frequency cutoff shown in the extrapolation of internal cockpit noise shown by the dashed line in Figure 10. Note that the data begin to match the measured external data when the boundary layer thickness is greater than 1/10th of a wavelength.

Because very little data is available for flight speeds exceeding Mach 1, it is impossible to give empirical relationship for the supersonic case. However, it would appear that the subsonic relationship can be utilized until better data becomes available, by substituting local boundary layer parameters in place of the free stream parameters in the above relationship. In addition, it must be remembered that these data apply to relatively simple shapes. Before they can be applied to new designs, the appropriate similarities must be found by the aerodynamicist. Consideration must also be given to shock wave noise which is in addition to that considered.

Another source of sonic loading which is believed to have been found on some recent test vehicle flight is wake noise. This source results from turbulent eddies and vortices in the subsonic wake behind a bullet-shaped missile. Although little is known regarding this phenomenon, it may be responsible for exciting the axial compressional mode in a vehicle, and measurements of the SPL in this region would appear very worthwhile. Note that similar phenomena may be anticipated along the wake of any projection from the surface of the aircraft or missile. In some cases the vortices in these wakes will give an almost discrete frequency noise which will vary with flight velocity.

SUMMARY - EXAMPLE OF SONIC EXPOSURE

The various empirical relationships which have been given enable a first approximation for a flight sonic exposure history. One example has been calculated in Figure 11 for a possible launch profile. This example clearly illustrates the three major phases of sonic exposure which have been discussed. Here the maximum noise occurs during launch and decreases rapidly as the missile rises, until the rocket stream no longer impinges on the ground. As the missile gains more altitude and velocity the noise from the rocket stream



160
150
140
130
120
110
Sound Pressure Level in db re .0002 dyne/cm²
Launch
Rocket
Boundary Layer
Overall
2400 - 4800 CPS
37 - 75 CPS
Time (sec) 0 20 40 60 80 100
Velocity (ft/sec) 0 455 1000 1700 2800 4600
q (lb/ft²) 0 220 900 800 200 20

Figure 11

continues to decrease, but at a lower rate than the initial decay. Then the boundary layer noise begins to overshadow the rocket noise, first at the higher frequencies, and finally at the low frequencies. This boundary layer noise increases with the increase in dynamic pressure until the maximum dynamic pressure is attained. Thereafter the noise decreases as the missile leaves the atmosphere. Upon re-entry, the boundary layer noise will increase again, depending upon the vehicle's re-entry profile.

The various relationships which have been given are intended primarily as a frame of reference for discussion of the prediction of sonic exposure, rather than a description of a comprehensive prediction method.

However, they do exhibit a form of prediction method and serve to illustrate the major gaps in present knowledge of sonic exposure. Perhaps the most serious area which requires investigation for missile applications is the effect of launch configurations and the development of launch pads which minimize sonic exposure during this crucial flight period. Another important area is the understanding of boundary layer turbulent phenomena and the mechanism of the transfer of energy to the vehicle. Hence, much experimental and theoretical work toward understanding of sonic exposure remains as a continuing challenge. In addition, it is clear that noise measurements should be included in each weapons system development to allow early anticipation of unexpected problem areas and to insure that the final design and test criteria are based on the actual environment.

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FLIGHT MEASUREMENTS OF DYNAMIC LOADS AND STRAINS

By

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This paper presents a general description of strain gauge techniques and measurements taken in flight on Lockheed test aircraft. It covers a philosophy of where and when to make load or strain gauge measurements. Application of the technique and procedures to the solution of some detailed structural problems is dealt with. Beyond the detailed strain measurements, some case histories of basic load determination are discussed. Relationship and correlation between analytical data and test data is touched on; and the matter of load probability prediction is discussed with respect to some special flight load problems.

Until a very few years ago, the flight test engineer was seriously hampered in measuring the magnitude of dynamic loads and strains on an aircraft in flight, through lack of adequate instruments. He was confined to use of the V-G recorder, plus taking pressure surveys. Both methods imposed severe limitations on determination of actual loads and strains on the structure. Sometimes a photographic camera was used for measurement of deflections, but this system, too, had serious limitations.

Development of the wire resistance strain gauge provided a system that was nearly ideal for solution of many instrumentation problems associated with flight load measurement. Two main advantages of the strain gauge are the fact that it is remote reading and that its frequency response is excellent. A companion

development, wonderfully suited to the recording of strain gauge outputs, was the oscillograph incorporating multiple high-sensitivity galvanometers. Together, these two devices have proved to be a great advance in the problem of determining dynamic loads and strains on an aircraft in flight. This system is certainly not the ultimate tool for the job, but it has provided a means of measuring more things more accurately.

Like most experimental operations, strain gauge work requires that a great deal of importance be placed on detail, and there are a great many details that are vital to successful strain gauge work.

In measuring loads on an aircraft structure, of first and primary importance is the selection of a proper location for the gauge. Pains must be taken to place the gauges where they will sense the proper strain. Such things as local buckling, stress concentration, local heating from propulsion units (or other sources) must all be considered in selecting gauge location. An equally important step is proper installation of the gauges themselves, which must be done with a great deal of care. In many types of measurements, the method and type of calibration used is also quite important.

In all flight load measurement work at Lockheed, primary emphasis on strain gauge work in flight has been directed toward the determination, or checking, of external basic loads. Our general philosophy has been that if we can know the basic loads accurately, then detail stresses or stress distributions can be determined more easily on the ground during static test work. In most of our work this philosophy has been followed successfully. In some instances, however, detail stress measurements were required and were made. In general, such detail measurements were associated with problems involving local structural cracking, as a result of fatigue. This paper will present some typical cases which involve both basic load measurement and also detail stress measurements, in illustration of work accomplished in this field. We believe the results show conclusively that there is no present-day easy substitue for strain gauges in performing this work.

A first example of strain gauge use in measuring detail stresses concerns a fatigue problem which involved cracks in a wing rear beam web. The aircraft was a Constellation Model C-121C and structural cracking was encountered at flight times as low as 600 to 700 hours. By analytical approach to the problem, the designers evolved two different changes in the structure in an effort to cure the web cracking. One of these was a field modification which installed an additional stiffener between the original stiffeners to obtain a nine-inch spacing. The second was a correction to aircraft on the production line which consisted of an increase to the beam web thickness from .040 to .051, while maintaining the nine-inch stiffener spacing. Both corrections were installed without recourse to measurements, either before or after the fixes were made. Neither was successful since cracking of the structure continued in service at fairly low flight hour intervals.

At this point, a flight test program was instituted to measure vibratory stresses in the web at various engine speeds since it was believed that cracks were associated

with engine operation. Measurements were taken on an airplane which incorporated the production change of web thickness increase to .051. Thereafter, several different structural configurations were tested in an effort to arrive at changes which would result in low stress levels. Results of these measurements are shown in Figure 1. It may be seen that the addition of a single stiffener did not significantly reduce the vibratory stresses, thus corroborating service experience. The addition of two stiffeners, however, did significantly decrease stresses in the web. Application of the additional stiffeners to aircraft in service solved the problem completely. An interesting sidelight of this program is that the high stresses were obtained only when the wing integral fuel tanks were nearly full. With the tanks empty, measured stresses were very low for all configurations tested.

Another example in which a fatigue problem was solved by measurement of detail strain with strain gauges, concerns skin cracking at the corners of passenger door cutouts. Gauges were installed at the corners of both forward and aft cabin door frames, as shown in Figure 2. Oscillograph records of strain variations were obtained for a variety of flight and ground operating conditions. The former included pull-up maneuvers, sideslips, cabin pressurization, and rough air conditions; while the latter covered landing, taxiing, turning, and engine run-up.

The maximum strain changes which were measured occurred at the rear cabin door during a hard landing. All three instrumented corners experienced equivalent stress changes around 40,000 psi, as shown by Figure 2. Under no circumstances did the forward cabin door experience stress oscillation over about 6,000 psi.

Based on these measurements, a doubler was designed and installed, and further fatigue failures at the cutouts were eliminated. In this particular case, it was virtually impossible to calculate how the door corners could get stresses of an order high enough to create a fatigue problem. On the other hand, actual measurements taken on the airplane showed clearly that a flight or ground maneuver which excited the fuselage in torsion could in fact cause serious stress changes in the corners of the aft passenger door cutout.

Turning now from the measurement of detail stresses, we shall consider the determination, or checking, of external basic loads. One of these is a dynamic loading condition which is practically impossible to analyze. This condition is the accelerated stall buffet. Present accepted design practice is to add arbitrary factors on top of normal design loads to cover the unknown buffeting load quantities. Unfortunately, these arbitrary design factors have proved overly conservative for some airplanes, while for others they have fallen short of the actual loads encountered by the aircraft in a deep penetration into accelerated stall. At Lockheed, we have obtained and analyzed measurements in accelerated stall for a number of airplanes including fighters, trainers, transports, and patrol types. Such measurements, in accelerated stall buffet, provide excellent examples of basic structural load measurements which are considerably more useful than detail strain or stress measurements.

C-121 CONSTELLATION WING BEAM STRESSES - EFFECT OF STIFFENERS ON BEAM WEB

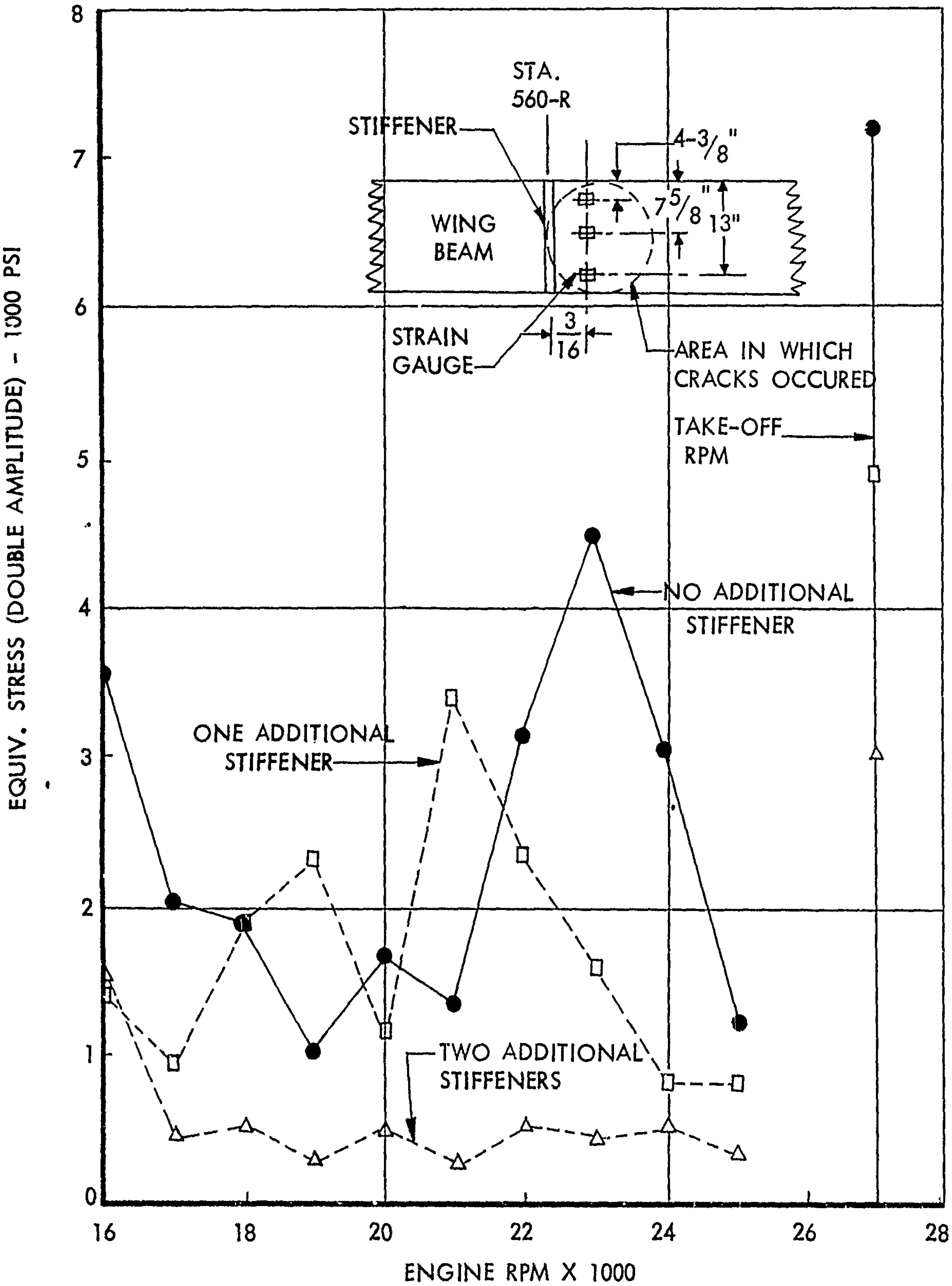


FIGURE 1

STRESS VARIATIONS - AT DOOR CORNERS DURING HARD LANDING

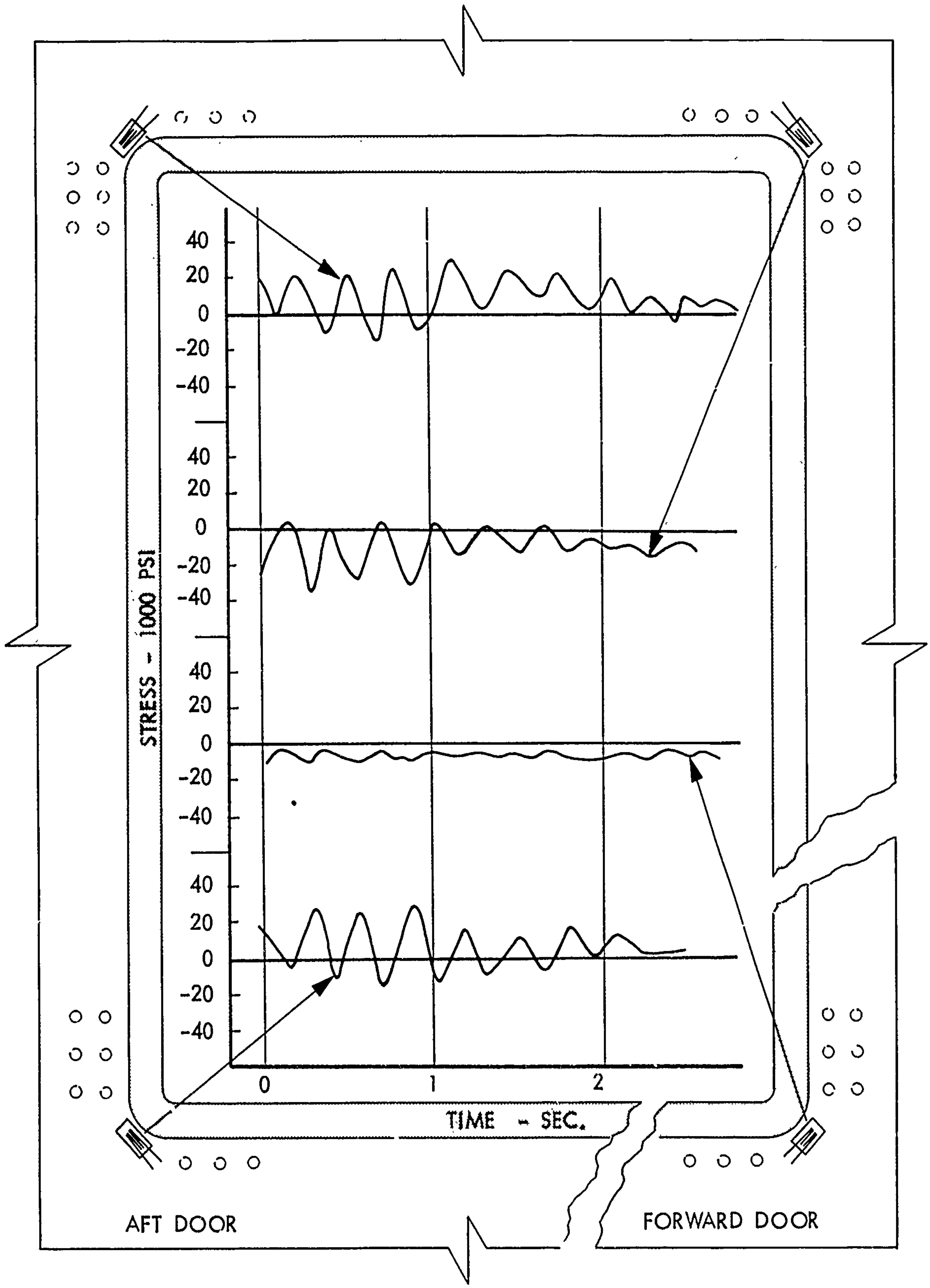


FIGURE 2

Furthermore, they can be better correlated with available structural strength information. Therefore, in problems associated with aerodynamic buffet, it is invariably our practice to measure basic structural loads rather than detail stresses. In illustration of progress at Lockheed in this field of testing, examples of measurements obtained on a four-engine radar patrol airplane and also those taken on a jet trainer airplane will be considered here.

Early tests on the patrol airplane indicated that, essentially, two structural areas were equally critical. These were the outer stabilizer in torsion, and the aft fuselage in combined shear and torsion. Analysis of the load measurements were made in general accordance with procedures outlined by the NACA in TN-3080, which covers analysis of buffeting loads on a fighter airplane. From this statistical analysis of the major parameters measured, those affecting buffeting loads were determined to be: airspeed, and incremental lift coefficient above that for start of buffet (ΔC_N). Thus, all measured loads were divided by equivalent airspeed, and plotted against ΔC_N, as shown typically for aft fuselage torsion in Figure 3. A statistical analysis of the data was then made to determine the average trend line, and also lines for various probabilities of occurrence. Note that considerable scatter is evident in the data, indicating that some sort of a probability analysis was in order.

For purposes of defining a reasonable condition under which we were willing to conduct an actual structural demonstration, a probability factor of one in ten was chosen. In other words, for a given demonstration condition, it was expected that the limit strength would be exceeded once out of each ten tries, and that the remaining nine would be at (or less than) limit.

The net results of the analysis in terms of a V-N diagram are shown in Figure 4. Note that based on a ten percent probability, the allowable load factor line shows that a sizeable penetration can be made into buffet at the lower airspeeds with the allowable penetration reducing with increase of airspeed.

Next, a series of tests leading up to structural demonstration were conducted on a jet trainer. Requirements of the demonstration included an abrupt application of stick force and holding this force until maximum load factor was obtained. This type of maneuver automatically implies a very deep penetration into accelerated stall.

Analysis of preliminary tests on this airplane clearly indicated that the parameters of airspeed and ΔC_L did not show sufficient correlation with the measured horizontal stabilizer bending moments to allow use of the data as basic parameters. Further analysis showed that the two main parameters which actually governed the buffet loads were: incremental angle of attack above that for maximum lift coefficient, and dynamic pressure. In addition to the effect incremental angle of attack has on airflow separation over the wing, it has another effect: it is a major factor controlling position of the horizontal tail with respect to the wing wake. In this case, higher angles placed the horizontal tail deeper in the wake.

MAXIMUM AFT FUSELAGE TORSION LOAD ÷ EQUIVALENT AIRSPEED vs NORMAL LIFT COEFFICIENT INCREMENT (ΔC_N)

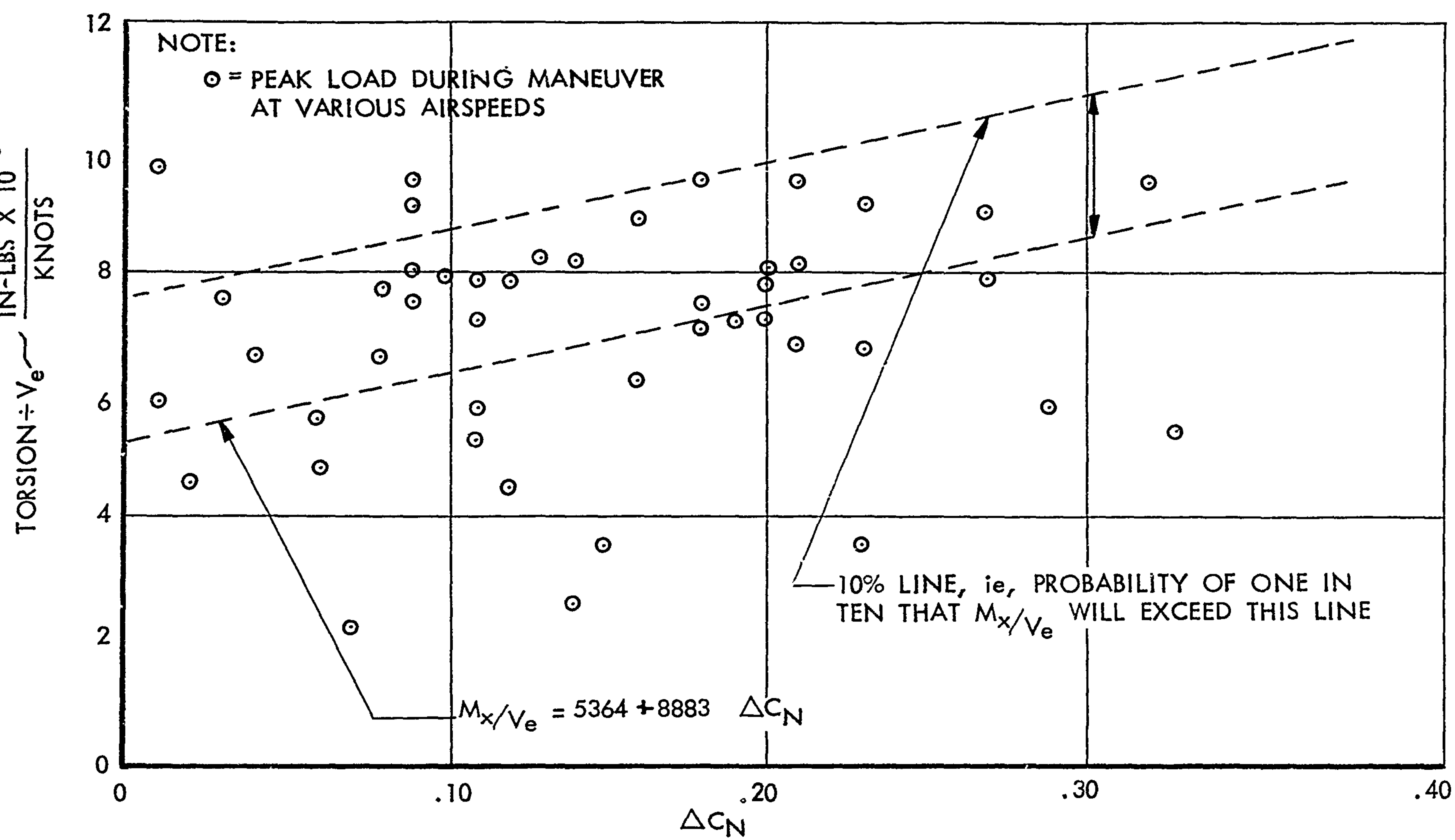


FIGURE 3

V-n DIAGRAM - PREDICTED LOAD FACTOR FOR LIMIT LOAD DURING BUFFET IN SYMMETRICAL PULL-UPS

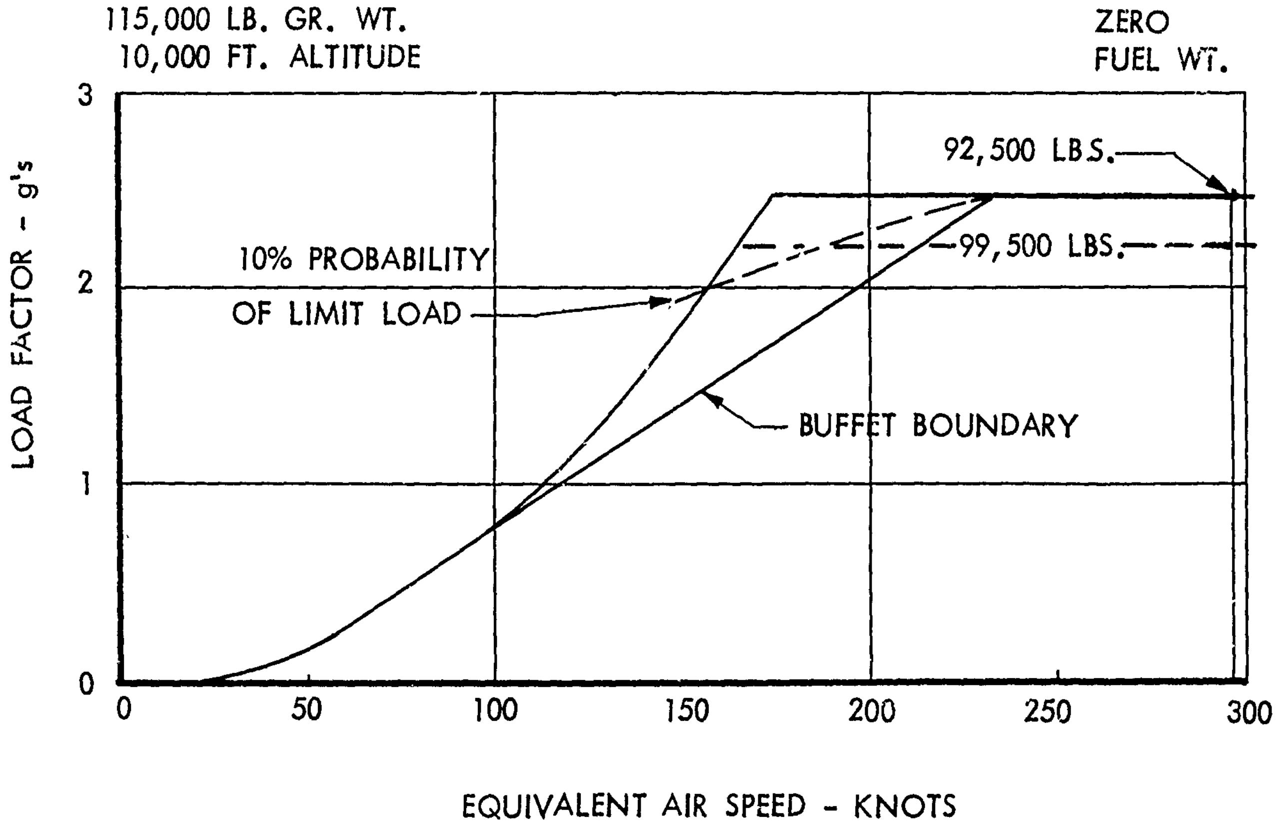


FIGURE 4

A typical plot of stabilizer bending moment divided by dynamic pressure (q) versus incremental angle of attack is shown in Figure 5, for both the left and right stabilizer measurements. Note that the scatter has been reduced considerably as compared with the plot of load divided by speed versus ΔC_N for the patrol airplane. In fact, for a buffeting condition, the scatter is remarkably small.

In the test program, various rates of application of stick force were used in making the maneuvers. These were divided into two basic categories: extremely abrupt and rapid. To illustrate the effect that rate of stick force application has on the loads, the same measured loads used in deriving the previous plot versus incremental are charted versus airspeed in Figure 6. It may be seen that the loads for rapid maneuvers are well below the line defined by the extremely abrupt maneuvers. In all cases, the airplane went well beyond stall but for the rapid maneuvers the incremental angle of attack attained was not so great.

It proved unnecessary to perform a statistical analysis for probability on this airplane in view of the excellent correlation of data.

The net result, in terms of V-N diagram, is shown by Figure 7. A definite speed was determined, up to which unrestricted extremely abrupt maneuvers could be performed with a very low probability of exceeding limit. It was possible to satisfactorily demonstrate this point with the measured loads falling very close to limit. In the area between the point for unrestricted maneuvers and the design load factor cutoff, deep penetration into buffet beyond full stall is allowed, but the pilot is cautioned not to use abrupt stick application.

Another structural area in which strain gauges have played, and are continuing to play, a most important role is that of measuring loads and strains on landing gears. Many fatigue problems have been avoided, and several solved, by such strain gauge measurements. Marginal conditions of shimmy or brake chatter have been detected which could have caused failures in service had they not been discovered during flight test programs. On at least one occasion, a very high loading of a nose gear was detected during the early phase of testing a new airplane. This loading condition had been overlooked in the design analysis and, when uncovered with strain gauges, a rational analysis was found to agree remarkably well with the actual measurements.

Strain gauges properly placed and used are not only a valuable tool to solve fatigue problems that occur on aircraft in service, but also, and even more important, they can uncover and predict fatigue-causing conditions during the development and prototype program. In addition, of course, strain gauges can determine or verify basic loads on the structure plus measure dynamic loads or strains which are practically impossible to calculate.

STABILIZER BENDING ÷ DYNAMIC PRESSURE VS. INCREMENTAL ANGLE OF ATTACK

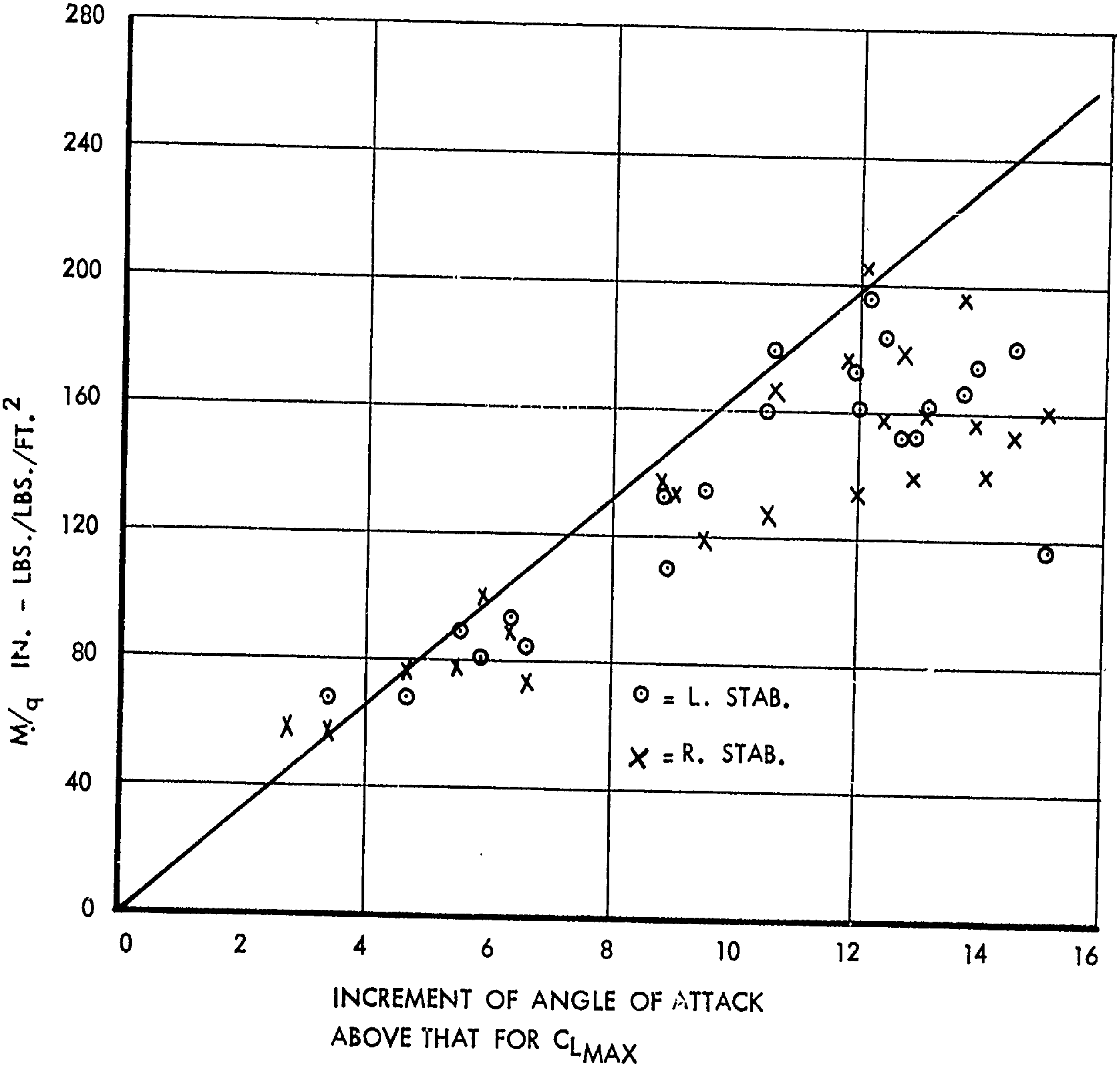


FIGURE 5

MAXIMUM STABILIZER BENDING MOMENT

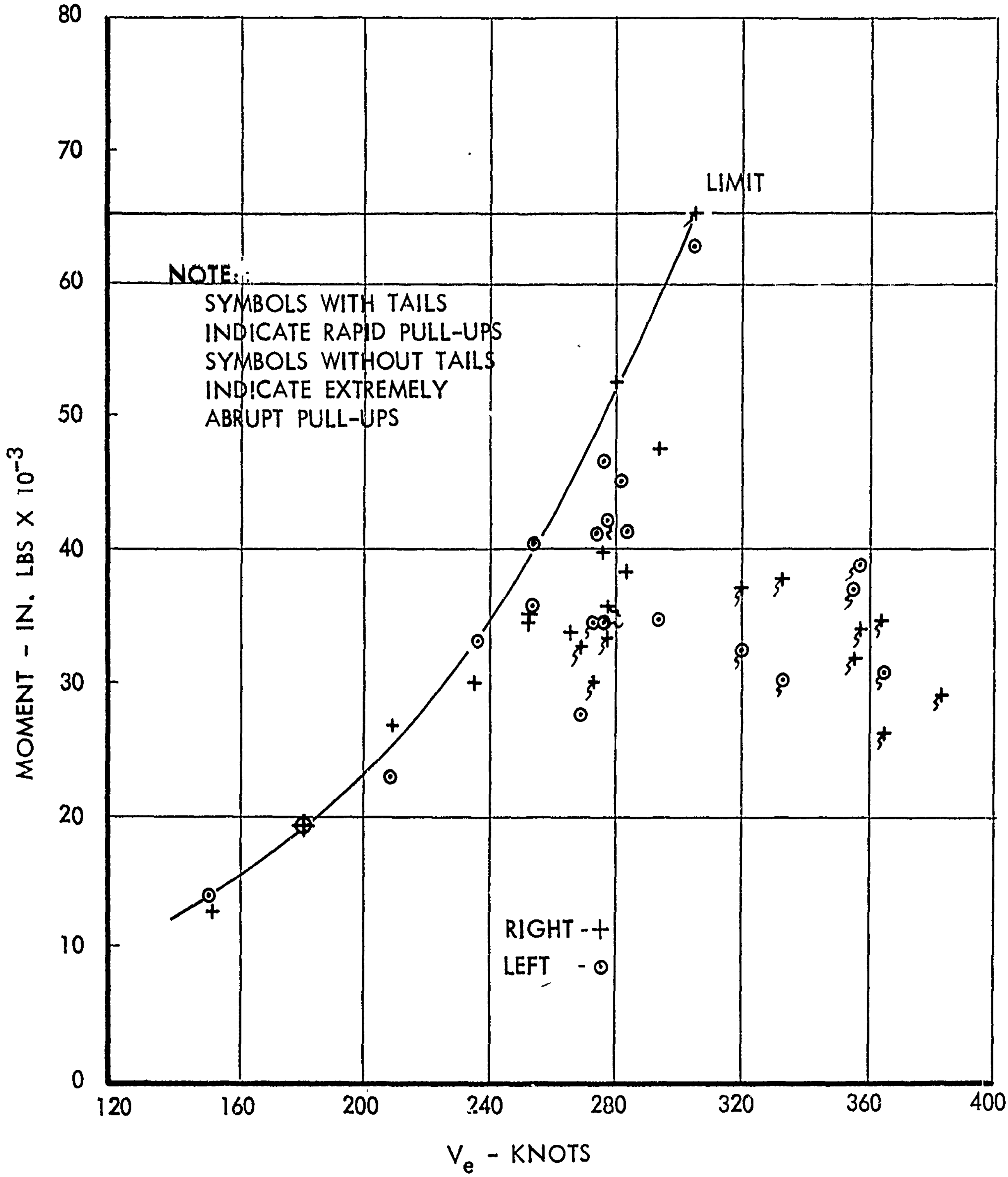


FIGURE 6

TYPICAL V-n DIAGRAM SHOWING POSITIVE HIGH ANGLE OF ATTACK CORNER-CUTOFF DUE TO BUFFET STRESSES

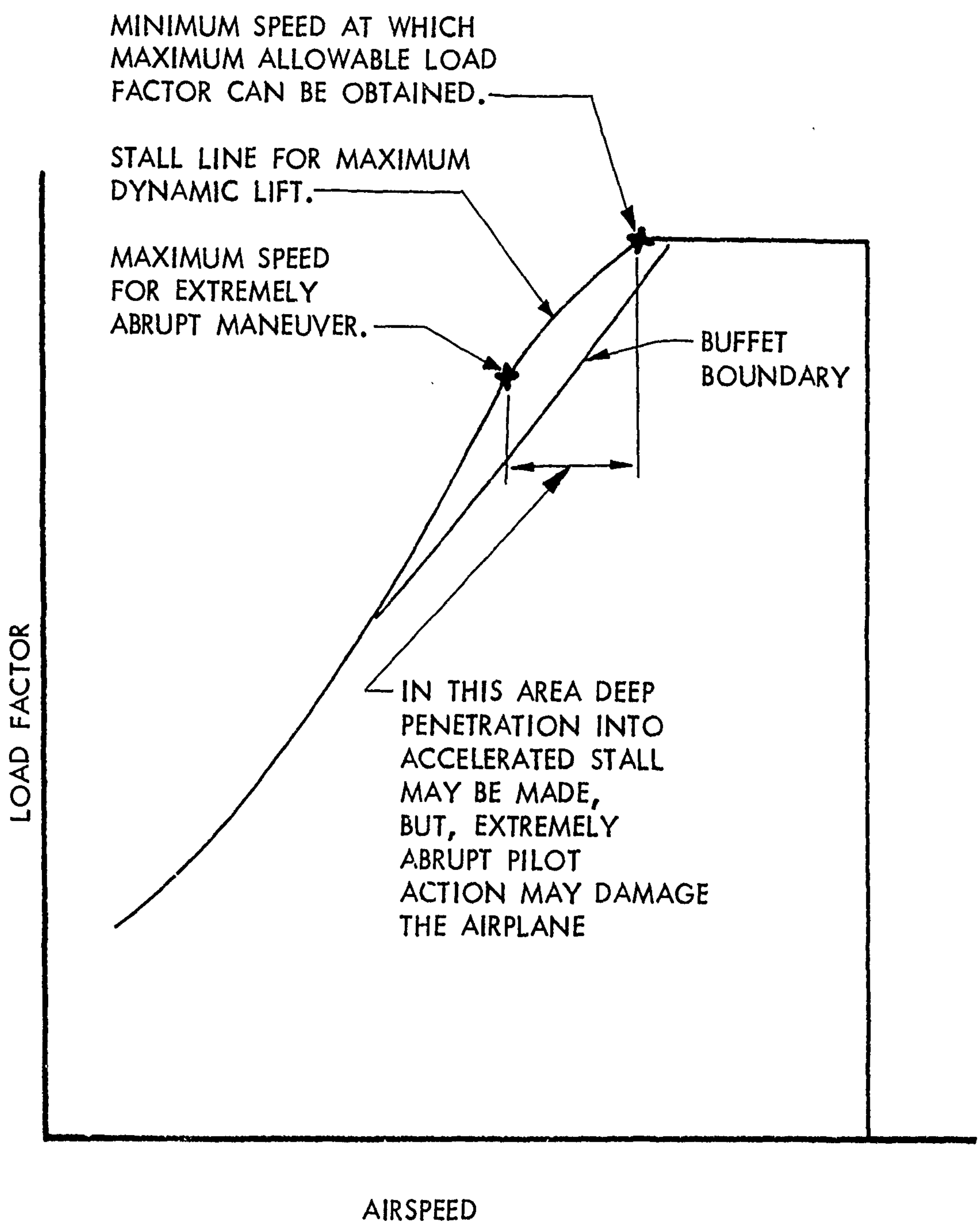


FIGURE 7

PREDICTION OF GUST LOADS IN AIRPLANE

AND MISSILE OPERATIONS

By John C. Houbolt and Roy Steiner

NASA Langley Research Center

SUMMARY

A condensed procedure is given for estimating the gust loads that are experienced by airplanes and missiles during operation. A turbulence model consisting of a continuous probability density distribution of root-mean-square gust velocity and the use of power spectral techniques form the basis for the approach. The treatment of the turbulence encountered is given completely by two sets of curves dealing with the variation of nonstorm and storm properties with altitude, one set being proportion of time involved, the other being the composite root-mean-square gust value. The treatment involves both repeated loads and probability considerations.

INTRODUCTION

Reference 1 and its predecessor, reference 2, are excellent companion reports which develop procedures for estimating the gust loads that are likely to be experienced by airplanes or missiles during their operation. This report has the same subject, and its main purpose is simply to summarize the basic findings of reference 1. In making this condensed version, a slightly different development is used; thus, the "cause" or gust field is herein assumed known and we proceed directly to determine the "effect" or statistical characteristics of the airplane response or load history. The evaluation of the details and makeup of the "causes," which forms a large part of reference 1, is thus avoided. Also, this treatment is cast completely in terms of power spectral techniques and no reference or consideration is given to discrete or derived gust velocities. As in reference 1, the study is mainly applicable to horizontal or nearly horizontal flight and applies only in a crude way to the near-vertical flight paths.

OUTLINE OF TREATMENT

The development of the report proceeds along the following lines. We first consider the atmosphere to be made up of discrete patches of disturbances of different mean-square intensity, Gaussian and stationary

in character, and then consider this discrete patch model to be replaced by a model which has a continuously variable distribution of root-mean-square gust velocity. Nonstorm and storm turbulence are treated separately but alike. Composite mean-square and basic peak count relations of vertical velocities are then derived. Airplane response is considered next, and this is followed by a treatment which combines nonstorm and storm encounter to yield the basic repeated loads data. Finally, probability considerations of the repeated loads are given.

COMPOSITE GUST ENCOUNTER

In the flight of airplanes or missiles, experience has taught us that turbulence encounter can be divided basically into two parts, nonstorm and storm. Each of these may be represented by a model comprised of a succession of discrete patches of turbulence of different mean-square intensity. Figure 1 depicts such a model in terms of vertical velocity versus distance traveled, where it is understood that the successive patches do not necessarily increase in intensity as shown in the figure.

The composite root-mean-square value of vertical velocity for this model is given by the equation

$$\sigma_c{}^2 = \frac{1}{d}\left[\int_{x_0}^{x_1} u_0{}^2(x)dx + \int_{x_1}^{x_2} u_1{}^2(x)dx + \int_{x_2}^{x_3} u_2{}^2(x)dx + \int_{x_3}^{x_4} u_3{}^2(x)dx\right]$$

$$= \frac{1}{d}\left[d_0(0) + d_1\sigma_1{}^2 + d_2\sigma_2{}^2 + d_3\sigma_3{}^2\right]$$

$$= p_1\sigma_1{}^2 + p_2\sigma_3{}^2 + p_3\sigma_3{}^2 \qquad (1)$$

where p_n represents the proportion of time spent in the patch having a root-mean-square value of σ_n, see sketch lower left of figure 1. For the limiting case in which the turbulence model is represented by a continuous variation in root-mean-square gust velocity, see sketch at lower right of figure 1, equation (1) would assume the form

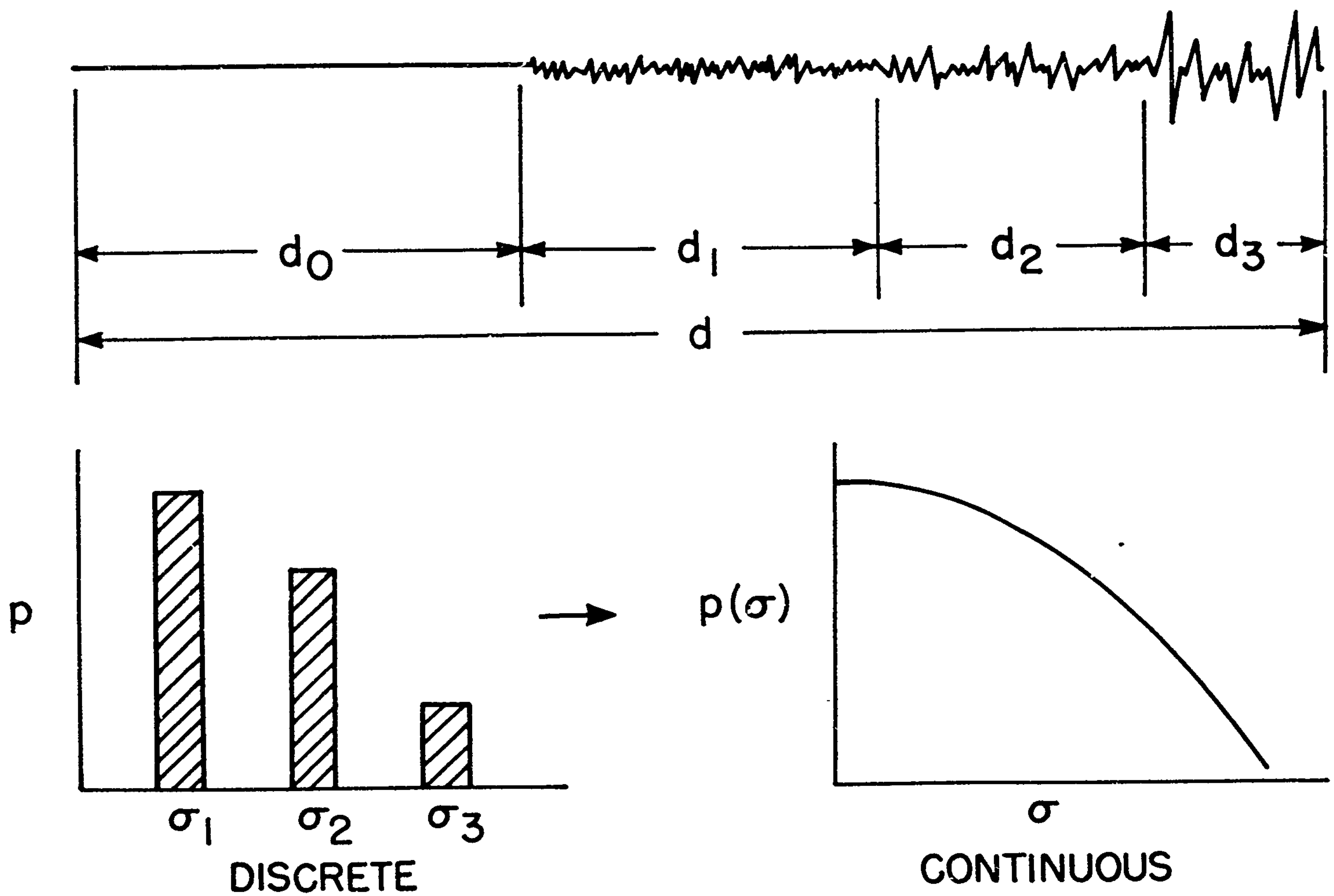


Figure 1.- Composite gust encounter. NASA

$$\sigma_c{}^2 = \int_0^\infty \sigma^2 p(\sigma)d\sigma \tag{2}$$

where $p(\sigma)$ is the probability density distribution of root-mean-square gust velocity.

We next derive information on the peak count of vertical velocity. To do this we first assume that the spectra for all the patches of turbulence in our discrete patch model are all similar in shape and differ only in level. Then if we assume further that each patch is a stationary Gaussian disturbance, then the equation for approximating the number of peaks per foot which exceed a given level of vertical velocity follows as

$$G(u) = \frac{1}{d}\left[d_1 G_o e^{-\frac{u^2}{2\sigma_1{}^2}} + d_2 G_o e^{-\frac{u^2}{2\sigma_2{}^2}} + d_3 G_o e^{-\frac{u^2}{2\sigma_3{}^2}}\right] \tag{3}$$

where G_o depends only on the spectrum shape and is the expected number of times per foot that $u(x)$ crosses the value zero with positive slope, and is given by the equation

$$G_o = \frac{1}{2\pi}\left[\frac{\int_0^\infty \Omega^2 \phi_u(\Omega)d\Omega}{\int_0^\infty \phi_u(\Omega)d\Omega}\right]^{1/2}$$

where ϕ_u is the spectrum of gust velocity and Ω is the spacial frequency, $\Omega = \frac{2\pi}{L}$ (L being the wavelength of a given frequency component). It is remarked that equation (3) is exact for calculating the number of crossings per foot with positive slope of given values of u, but is also a good approximation for determining the number of peaks above a given level so long as u is greater than σ_n.

In a manner similar to that used for equation (1), equation (2) may be written

$$G(u) = G_o \left[p_1 e^{-\frac{u^2}{2\sigma_1{}^2}} + p_2 e^{-\frac{u^2}{2\sigma_2{}^2}} + p_3 e^{-\frac{u^2}{2\sigma_3{}^2}} \right]$$

and for the limiting case of a continuous variation in root-mean-square value this equation may, in turn, be written

$$G(u) = G_o \int_0^\infty p(\sigma) e^{-\frac{u^2}{2\sigma^2}} d\sigma \tag{4}$$

The experience to date in analyzing many records taken during routine airplane operation indicates that $p(\sigma)$ may be approximated with good accuracy by the equation

$$p(\sigma) = \frac{1}{a_1}\sqrt{\frac{2}{\pi}}\, e^{-\frac{\sigma^2}{2a_1{}^2}} \tag{5}$$

On substituting this equation in equation (2), we find that $a_1 = \sigma_c$; with this fact we then substitute equation (5) into equation (4) and find that the equation for peak count reduces remarkably to the simple equation

$$G(u) = G_o e^{-\frac{u}{\sigma_c}} \tag{6}$$

This equation forms the basis for this entire report. Airplane operational studies show that σ_c is a function of the altitude of operation and this variation will be shown subsequently.

A summary of equations (2), (4), (5), and (6) is given in figure 2.

COMPOSITE GUST RESPONSE

To determine the overall operational load histories we simply combine the responses that develop in the exposure to the different Gaussian gust disturbances. Figure 3 depicts schematically this operation from a power spectral point of view. Horizontally the contribution from the various patches are added together to give the composite picture shown in the right-hand column. Vertically the diagram depicts the basic power spectral relation which states that the spectrum φ_x of output response (acceleration, moment, stress, etc.) is simply the spectrum φ_u of input disturbance multiplied by the square of the frequency response function T (in this case the response of the airplane due to a unit sinusoidal gust). For the other quantities in the figure the σ_n's, σ_c, G, and G_o are as before and φ_1 is the shape of the gust input spectrum such that when it is multiplied by the mean-square gust velocity the actual spectrum is obtained. The two equations for peak count shown apply to the case of a continuous variation in root-mean-square gust velocity and specifically for the distribution given by equation (5).

The equation for peak count of the output response is derivable as follows. We first recognize that for any individual patch of turbulence the mean-square value of output response is related to mean-square value of input gust disturbance by the relation

$$\sigma_{x_n} = \sigma_{u_n}\left[\int_0^{\infty} T^2(\Omega)\phi_1(\Omega)d\Omega\right]^{1/2}$$

$$= A\sigma_{u_n}$$

where A, the square root of the integral, is in the nature of a gust response factor which depends on airplane weight, wing area, speed, air

$$\sigma_C^2 = \int_0^\infty p(\sigma)\,\sigma^2\,d\sigma$$

$$G(u) = G_0 \int_0^\infty p(\sigma)\,e^{-\frac{u^2}{2\sigma^2}}\,d\sigma$$

FOR $$p(\sigma) = \frac{1}{a_1}\sqrt{\frac{2}{\pi}}\;e^{-\frac{\sigma^2}{2a_1^2}}$$

THEN

$$a_1 = \sigma_C$$

$$G(u) = G_0 e^{-\frac{u}{\sigma_C}}$$

Figure 2.- Basic peak count relations. NASA

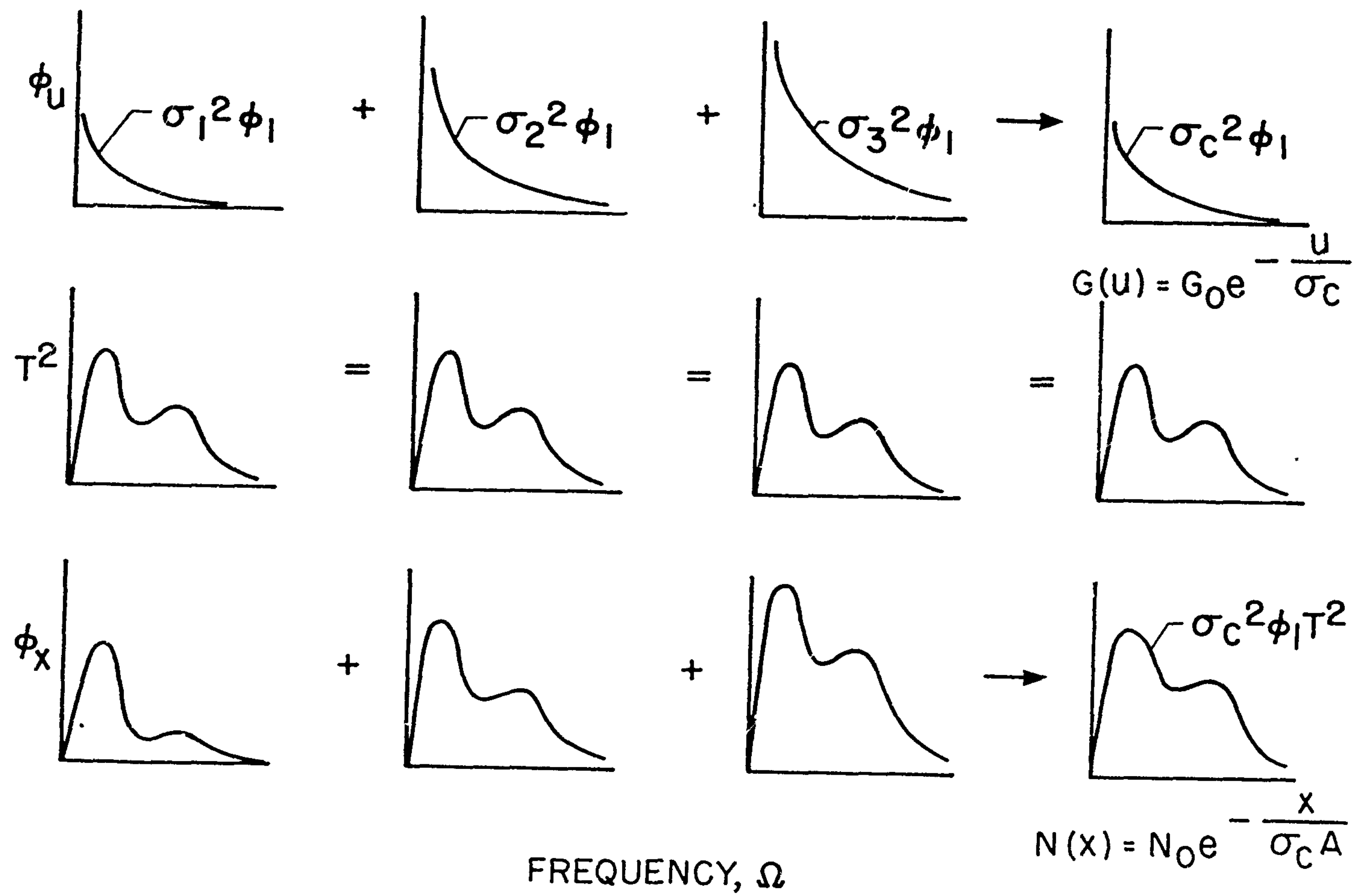


Figure 3.- Composite response to gusts. NASA

density, etc., and also directly reflects the number of degrees of freedom that are taking part in the response. Then, on the basis of encountering discrete patches, the number of peaks per foot in the response that exceed a specified value is, for the larger values of the response,

$$N(x) = \frac{1}{d}\left[d_1 N_o e^{-\frac{x^2}{2\sigma_1{}^2A^2}} + d_2 N_o e^{-\frac{x^2}{2\sigma_2{}^2A^2}} + d_3 N_o e^{-\frac{x^2}{2\sigma_3{}^2A^2}}\right]$$

$$= N_o\left(p_1 e^{-\frac{x^2}{2\sigma_1{}^2A^2}} + p_2 e^{-\frac{x^2}{2\sigma_2{}^2A^2}} + p_3 e^{-\frac{x^2}{2\sigma_3{}^2A^2}}\right)$$

where N_o is the expected number of times per foot that x crosses the value zero with positive slope and is given by

$$N_o = \frac{1}{2\pi}\left[\frac{\int_0^\infty \Omega^2 T^2(\Omega)\phi_u(\Omega)d\Omega}{\int_0^\infty T^2(\Omega)\phi_u(\Omega)d\Omega}\right]^{1/2}$$

For the continuous distribution of root-mean-square gust velocity and specifically for the distribution given by equation (5), this equation becomes

$$N(x) = N_o\int_0^\infty p(\sigma)e^{-\frac{x^2}{2\sigma^2A^2}}\,d\sigma$$

$$= N_o e^{-\frac{x}{\sigma_c A}} \qquad (7)$$

as was to be shown. In the present form and all subsequent applications of this equation in this report $N(x)$ refers to the number of exceedances

per foot for positive x, or for negative x. If the total number of peaks per foot falling above positive x and below negative x is desired, the results of equation (7) must be doubled.

REPEATED LOADS DURING COMBINED NONSTORM AND STORM OPERATION

It was mentioned earlier that turbulence encounter for airplanes in routine operations seems to divide naturally into two parts, nonstorm and storm. Further, it seems that the distribution functions for the root-mean-square gust velocities are similar for the two cases, the main difference being a larger composite root-mean-square gust velocity for storm turbulence than for the nonstorm counterpart.

On a statistical average basis the relation for peak count in combined storm and nonstorm operation follows by applying equation (7) to both cases and then combining the results as follows

$$N(x) = P_1 N_o e^{-\frac{x}{\sigma_{c_1} A}} + P_2 N_o e^{-\frac{x}{\sigma_{c_2} A}}$$

$$= N_o f(x) \qquad (8)$$

where here $N(x)$ represents the number of peaks per foot of the airplane response that lie above a value x (for x large), P_1 and P_2 represent the proportion of flight distance in nonstorm and storm turbulence, respectively, and σ_{c_1} and σ_{c_2} are the respective composite values of root-mean-square gust velocity.

The proportions P_1 and P_2, as well as the composite root-mean-square gust velocities σ_{c_1} and σ_{c_2}, have been found to vary with altitude. One of the main contributions of reference 1 was the evaluation and presentation of P_1 and P_2 for both airplane and missile operations; plots of these values as a function of altitude, as well as equation (8), are reproduced in figure 4(a); for design purpose enlarged plots of the proportions are shown in figures 4(b) and 4(c). It should be mentioned that the much smaller proportion of time spent in storms

$$N(x) = \underbrace{P_1 N_0 e^{-\frac{x}{\sigma_{c_1} A}}}_{\text{NON-STORM}} + \underbrace{P_2 N_0 e^{-\frac{x}{\sigma_{c_2} A}}}_{\text{STORM}} = N_0 f(x)$$

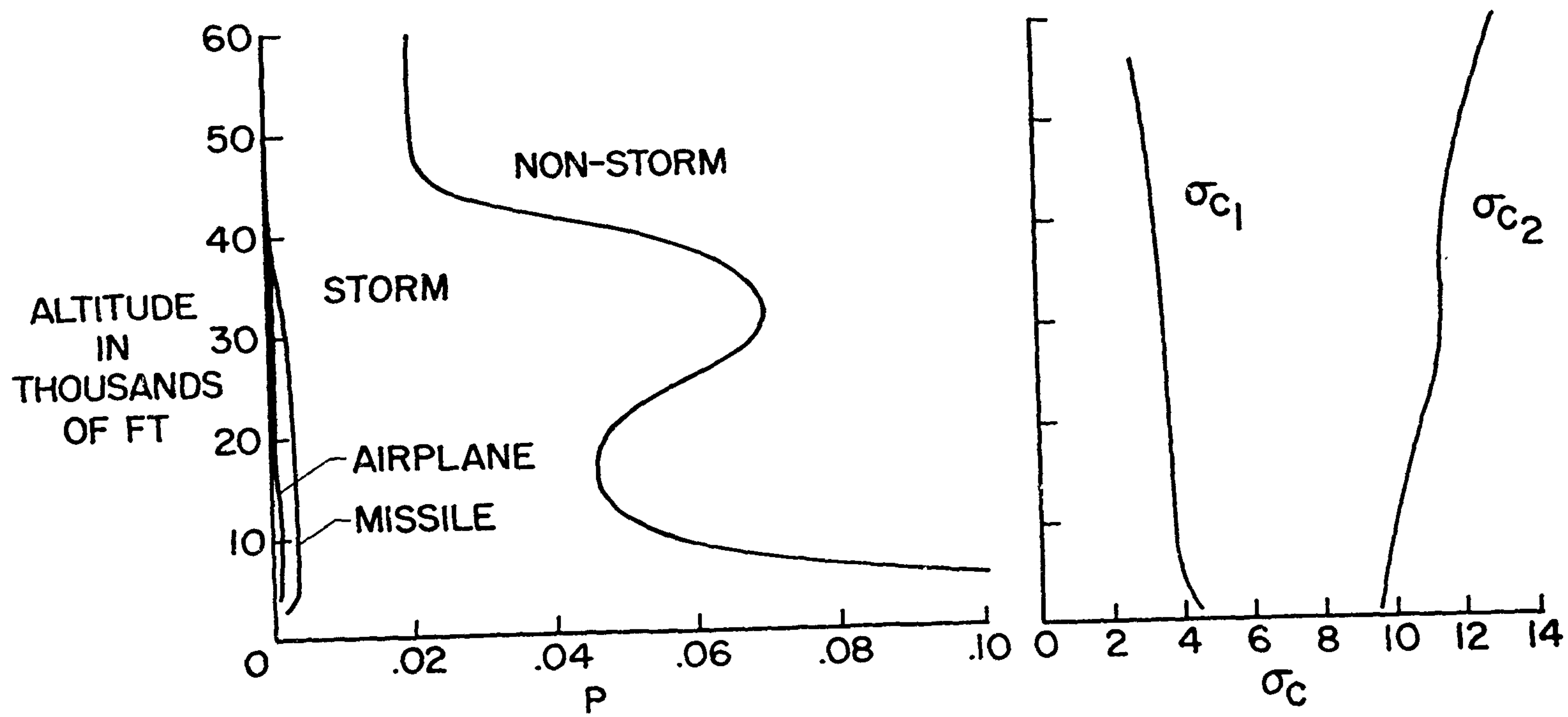
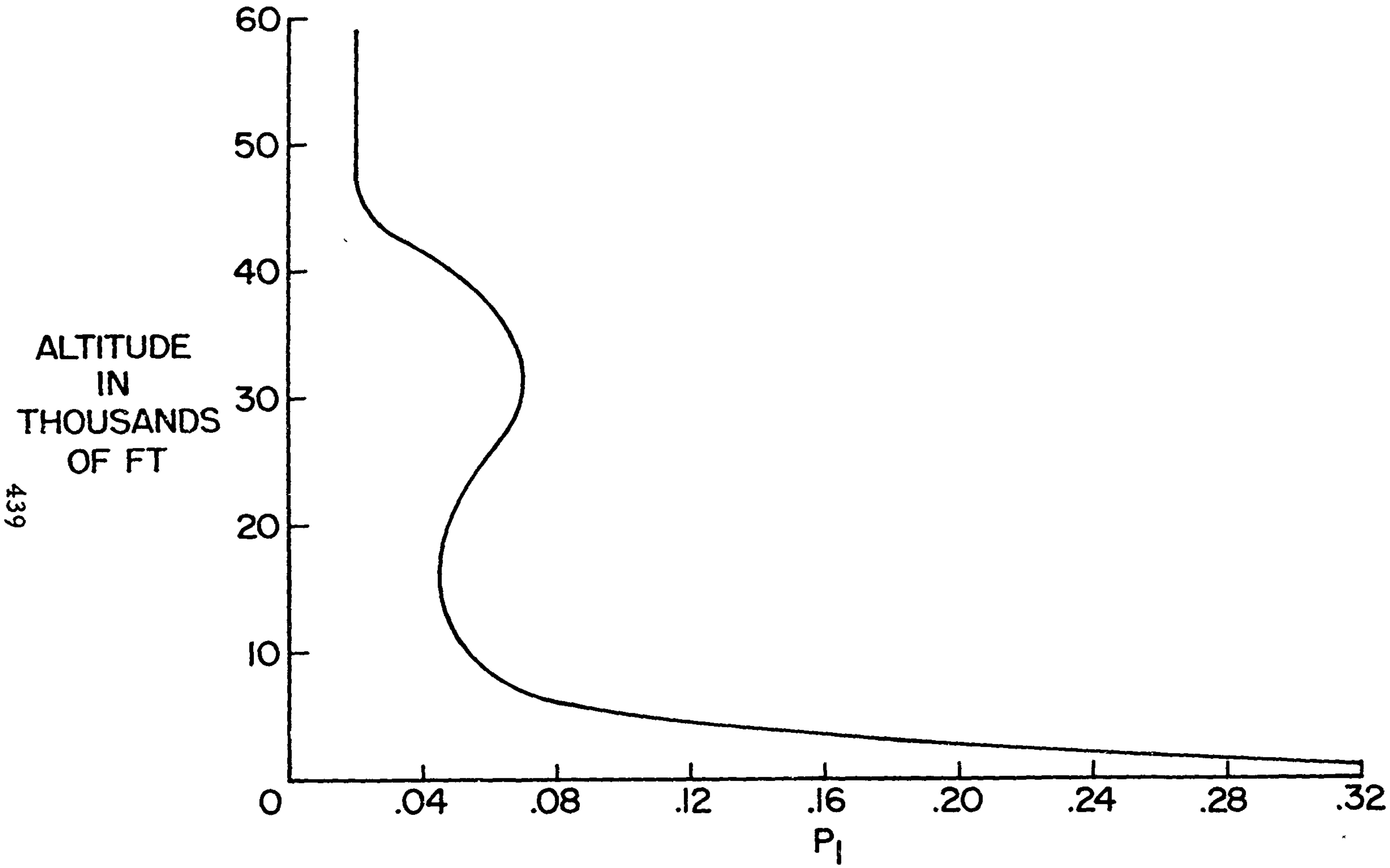


Figure 4a.- Combined nonstorm and storm. NASA



NASA

Figure 4b.- Proportion of flight distance in nonstorm turbulence.

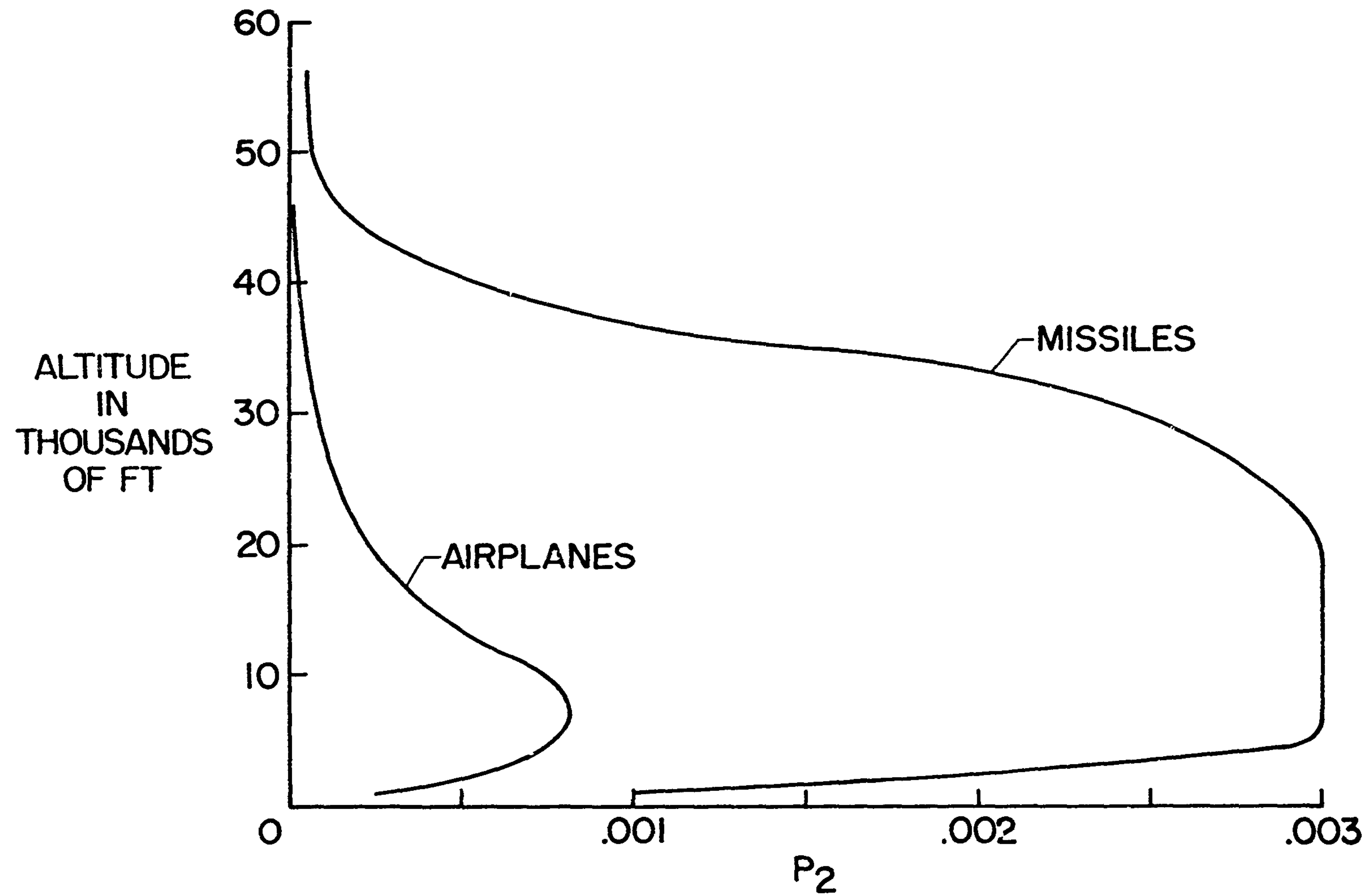


Figure 4c.- Proportion of flight distance in storm turbulence. NASA

by airplanes as compared with probable missile experience simply reflects the storm avoidance procedures (sight, radar, ground support) that may be realized by airplanes. In regard to σ_{c_1} and σ_{c_2} these values were also evaluated in reference 1, although not directly recognized as such. If we compare equation (8) with equation (21) in reference 1 then we see that $\sigma_{c_1} = b_1$ and $\sigma_{c_2} = b_2$ (a more formal derivation than the term by term comparison can be made to show these equivalences, but is not included here). A plot of variations of σ_{c_1} and σ_{c_2} with altitude is shown in the lower right of figure 4(a) (see table I of ref. 1 for evaluation of b_1 and b_2).

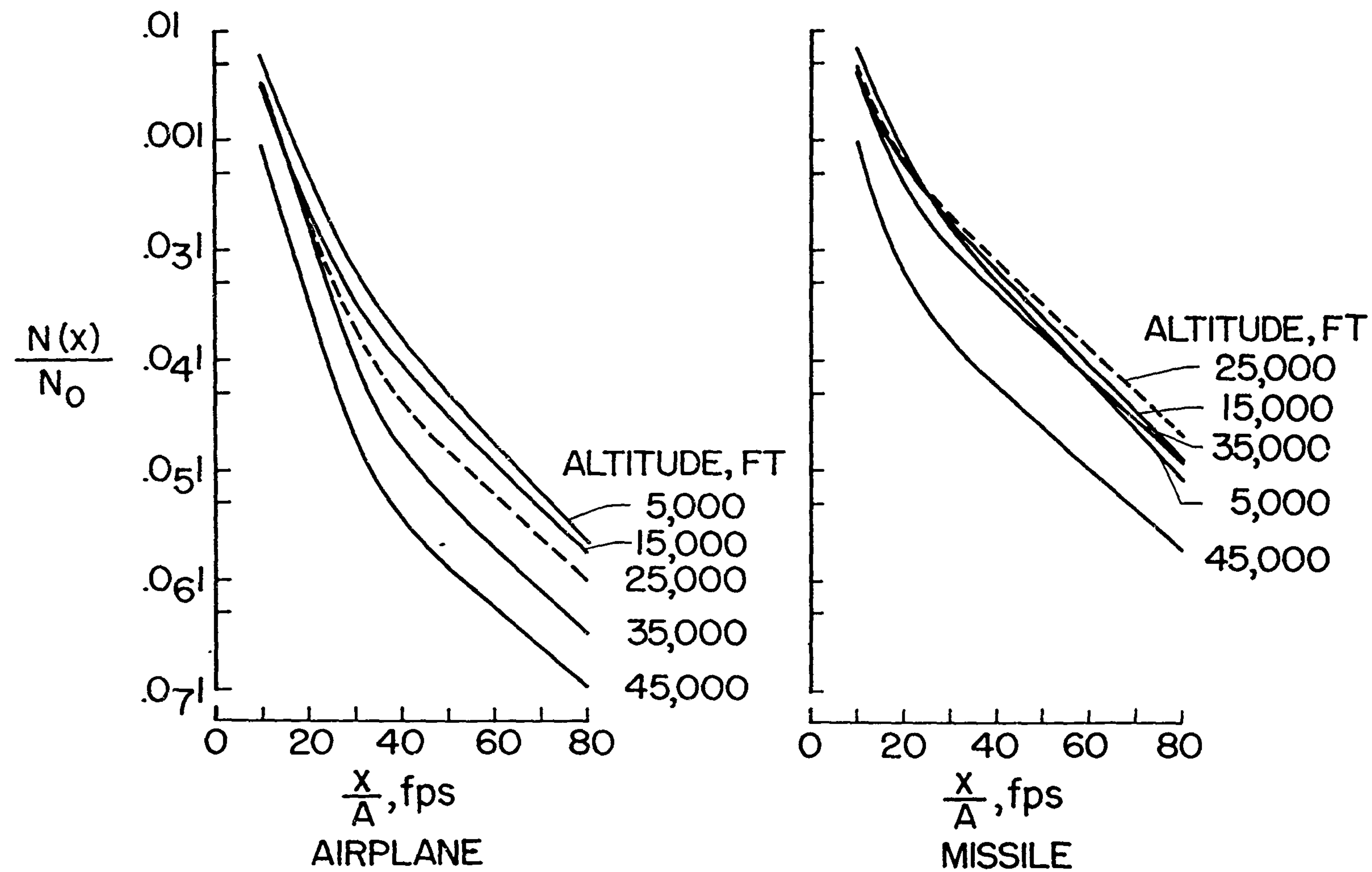
Thus we have in figure 4d the complete and succinct method for determining the statistical data on peak count for flights of airplanes or missiles at any given altitude; this figure is thus the meat of the present report.

Of course, any mission or succession of missions usually involves a large altitude variation. One further extension of the above procedures is therefore necessary where overall mission is the basic point of study. This extension is embodied in the following equation

$$N_T(x) = \frac{1}{D}\left[D_1 N_o f_1(x) + D_2 N_o f_2(x) + \ldots\right]$$

$$= N_o F(x) \tag{9}$$

where D_1, D_2, . . . are the distances traveled in each altitude bracket considered, and D is the total flight distance. Equation (9) thus gives, on the average, the composite total number of peaks per foot that exceed a given value x during the complete mission of the airplane or missile; again, from the point of view of peak counts, this equation applies to the larger values of x. The complete repeated load history from a statistical sense is thus contained in equation (9). The equation and the type of plot it produces is shown at the top of figure 5. For total peaks including positive and negative x, equation (9) should be multiplied by 2.



NASA

Figure 4d.- Generalized curves for gust load experience, as evaluated directly from equation and curves given in figure 4a.

From equation (9) the probability of exceeding a given value of x during any one or several flights may be estimated by the following means, part of which is duplicated on the bottom of figure 5. Consider a value x_d; then the number of feet D_d that must be flown to exceed this value once per foot on the average is defined by the relation

$$D_d N_T(x_d) = 1$$

or

$$D_d = \frac{1}{N_T(x_d)} \tag{10}$$

It follows then that the probability of exceeding the value of x_d in only D feet of flight is given approximately by

$$P_{ex} = \frac{D}{D_d} \tag{11}$$

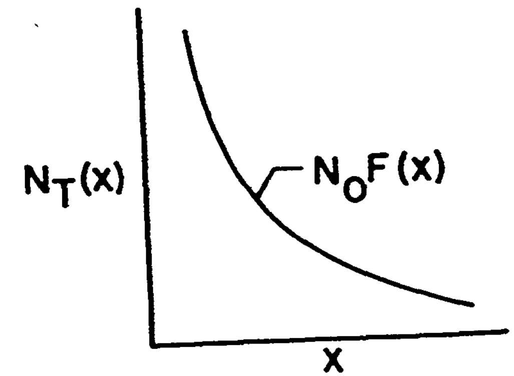
provided $D_d \gg D$, which is assumed to be the case of interest. (Note: This formulation assumes that the exceedances of x_d are distributed at random, which is strictly not the case because of the nonstationary make-up of equation (9), but the results obtained are probably not greatly in error.)

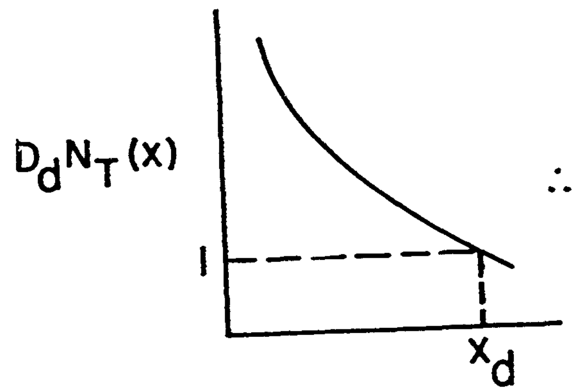
The substitution of equation (10) into (11), and the use of equation (9), gives the following simple formula for computing the probability of exceedance

$$P_{ex}(x_d) = DN_oF(x_d) \tag{12}$$

To show the nature of the results obtained from equation (12), a vehicle with $N_o = 0.002$ was assumed to fly at 25,000 feet altitude.

$$N_T(x) = \frac{1}{D}\left[D_1 N_0 f_1(x) + D_2 N_0 f_2(x) + \ldots\ldots\right] = N_0 F(x)$$





$$P_{ex} = \frac{D}{D_d} = DN_T(x_d)$$

$$\therefore P_{ex}(x_d) = DN_0F(x_d)$$

Figure 5.- Repeated loads and probability considerations. NASA

With center-of-gravity acceleration as the response quantity of interest, the probability of exceedance curves shown in figure 6 were obtained, one set pertaining to the vehicle as an airplane, the other as a missile. These charts now form an instructive chart for evaluating the vehicle probable flight experience. As a specific illustration the charts show that the probability of exceeding a value of $x/A = 50$ in 1000 miles of flight is 0.016 as an airplane and 0.317, or 20 times as much, as a missile. Further, if the flight distance is doubled to 2000 miles then a value of $x/A = 60$ is determined for the airplane for the same probability value as determined before, representing an increase in x of only 20 percent.

CONCLUDING REMARKS

It should be understood that the procedures given herein are intended to give statistical information on repeated load occurrences; that is, on the average values are found which do not necessarily reflect the experience in any one out of, say, a hundred flights.

The assumption of a constant average velocity of flight is tacitly made herein. In the spectral considerations the frequency argument $\Omega = \frac{2\pi}{L} = \frac{\omega}{V}$ is assumed used throughout, where ω is circular frequency and V velocity. The velocity is thus brought in through the frequency response function and its effect is to expand or contract the frequency response function along the Ω axis for decreasing or increasing velocities, respectively. An average velocity of flight is perhaps satisfactory for most operations. A slightly better choice would be to take an average value for nonstorm turbulence and a different average value for storm turbulence. The effect would be simply to use different values of N_o in the two terms of equation (8).

A similar point has to be made about the variation of N_o with altitude, since in the treatment it was also tacitly assumed to be invariant with altitude. A slightly more precise way, but probably one of second-order importance, would be to evaluate N_o at each altitude in question and modify equation (9) accordingly. The effect of taking velocity and altitude variation on N_o into account should be noted to be nothing more than slightly modifying P_1 or P_2 in equation (8), or the ratios $\frac{D_n}{D}$ in equation (9).

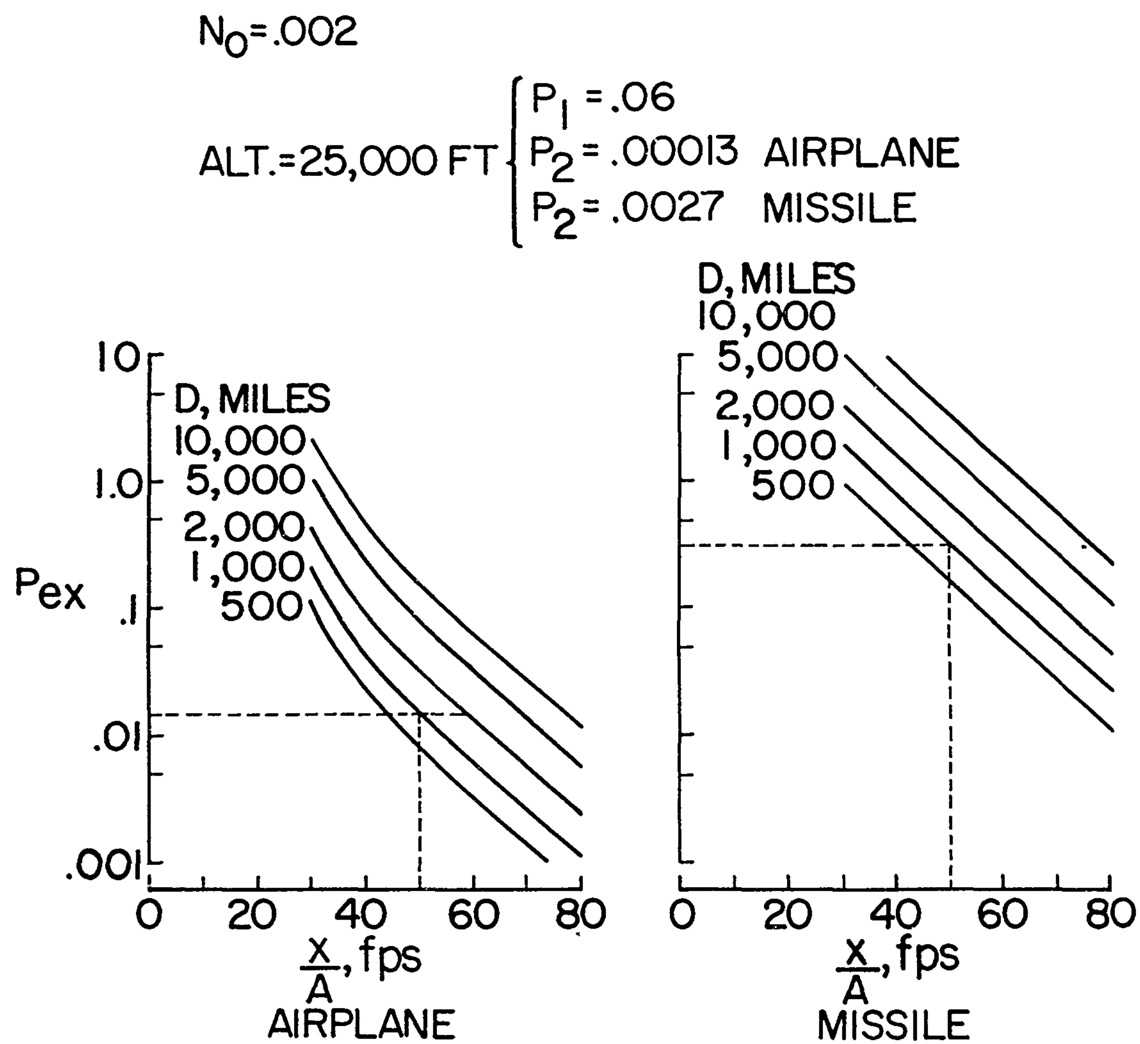


Figure 6.- Specific probability study. NASA

Again it is mentioned that results contained herein are based on airplane measurements obtained in essentially horizontal flight and, hence, appear applicable to missile flight operations (as well as airplane) involving only horizontal or near-horizontal flight.

Some further comments on application and limitations are given in reference 1.

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THE INFLUENCE OF DYNAMIC LOADS ON AIRCRAFT FATIGUE

by

C.E. Jackson[a], K.R. Thorson[b], J.E. Wherry[c], J.B. Dempster[d]

ABSTRACT

Presented in this paper is a discussion of a philosophy and the techniques used in airplane flight test programs designed to obtain load experience data on flexible aircraft. In addition, a method of reducing raw loads data to a form directly usable in fatigue studies is described in the text. Two methods of idealizing typical random load data (peak count and load cycles) for use in fatigue analyses or testing are compared. A method is discussed for converting airplane center of gravity acceleration peak count distributions to structural load cycle frequency distributions. Other useful information that may be obtained during flight test programs of this type is reported. Examples of data are presented showing the low altitude derived gust velocity cumulative frequency distribution and the root-mean-square gust velocity probability density distribution as obtained from the current flight test programs including comparative data from other programs.

Results of the programs to date have provided valuable data for use in fatigue prevention studies. They have also pointed up areas which should be considered for future research. These areas are discussed.

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A	ratio of the response root-mean-square to the disturbance root-mean-square
$C(\Delta n)$	reading correction, $f_m(\Delta n)/f_v(\Delta n)$
$D(f)$	the dynamic magnification factor as a function of frequency
$F(f,\Delta n)$	c.g. acceleration peak count distribution
$F(f,\Delta M)$	load peak count distribution
$F(f,\Delta M,\bar{m})$	load cycle frequency distribution
M	load (bending moment, shear, torsion, strain, etc.)
$M(t)$	load time history
N	cumulative frequency
N_o	number of crossings of zero response level with a positive slope per unit of time or distance
P_i	the fraction of the total time spent in the i^{th} Gaussian component
Q	the load per unit c.g. acceleration under steady state conditions
$R(f)$	ratio of fleet to control accelerations as a function of frequency, $\Delta n_F(f)/\Delta n_c(f)$
U_{de}	derived gust velocity
f	frequency
$\hat{f}(\sigma_v)$	probability density distribution of the root-mean-square gust velocity
m	load time history mean, 1 g - level flight
$\bar{m}$	load cycle mean
n	airplane center of gravity load factor, g's
Δ	incremental
σ	root-mean-square

Subscript C   control

Subscript F   fleet

Subscript M   measured

Subscript S   static

Subscript U   uncorrected

Subscript i i^{th} Gaussian component

Subscript m   machine

Subscript u   gust velocity

Subscript v   VGH

The remaining symbols used in this paper are explained within the text.

INTRODUCTION

Recent experiences on a variety of fleet aircraft have served to emphasize the need for increased attention to prevention of structural fatigue. Satisfaction of this need requires the ability to accurately predict the service life of a design. The first step in this process of prediction is to obtain an accurate description of the loads involved.

The theories now in use for structural life estimations are based on the concept that all applied loads contribute to the fatigue experience. In practice, however, some loads are small enough to have a negligible effect. In general, the airframe will be subjected to a composite load spectrum consisting of random loads originating from:

1. gusts
2. buffet
3. maneuvers
4. landings
5. taxi
6. sonic vibration
7. other sources

In the present studies being conducted at Boeing, it has been established that the significant fatigue loads on the primary structure are caused by gusts, maneuvers and taxi. Throughout these studies the loads have been represented as

1. ground-air-ground loads (typical loads associated with the transitions from taxi to flight and from flight back to taxi)
2. flight maneuver loads (excluding the ground-air-ground loads)
3. gust loads

Current work indicates that these contribute approximately 97% of the load experience to critical areas of the wing, empennage and fuselage. Moreover the ground-air-ground and gust loads are closely associated with the airplane dynamics.

Figure 1 illustrates the relative importance of these loads on typical structures of the B-47 and B-52. Shown in this figure are the percentages of the total damage expected on the fleet airplanes due to ground-air-ground, maneuver, and gust loads. It can be seen that there is a substantial difference in the contribution of each of the three load types. The reason that the ground-air-ground cycle is more significant on the B-52 than on the B-47 can be explained by the fact that the B-52 carries internal wing fuel, and the design conditions for take-off and landing require these wing tanks to be nearly full. The differences shown between the mid span and the root of the B-47 are primarily due to the higher gust dynamic magnification factor experienced at the outboard station. This effect can best be illustrated by the data shown in Figure 2. Plotted there is the relative service life as a function of the gust dynamic magnification factor for the B-47 wing root under the same flight usage shown in the preceeding figure. The importance of this factor is indicated by the rapid decrease in relative service life with increasing gust dynamic magnification factor.

The effect of mission profile on the relative importance of the three load types may also be substantial. Figure 3 illustrates the difference in relative damage due to the three load types for high and low altitude training missions. The effect of increased turbulence, expected at low altitudes, is indicated by the higher percentage attributed to gusts during the low altitude training mission. It is evident therefore that accurate descriptions of the ground-air-ground load cycles, maneuver load cycles, and gust load cycles are required for service life studies.

The major portion of this paper will be devoted to the discussion of methods which are devised to provide accurate load history data from which fatigue load spectrums may be obtained for flexible airplane structures. In particular the discussion will center around the general aspects of the flight test programs now being conducted by the Boeing Airplane Company under contract with the Air Force. Furthermore, it should be mentioned that application of the test program philosophy results in the determination of both static and dynamic repeated loads. In fact it can be seen that for some conditions it is impractical to distinguish between the two.

TEST PHILOSOPHY

The test program philosophy is to collect samples of flight loads data which are representative of the repeated load experience being accumulated on first line military aircraft during operational use. Since the repeated load experience of the fleet airplanes depend strongly upon such things as mission profiles flown, weather conditions encountered and weight distributions maintained, it is evident that the accumulation of representative statistical samples of loads experienced in all flight conditions is required in order to adequately estimate fleet service life. From the standpoint of program length, it is not feasible to acquire this sample with a single airplane. Moreover it is economically infeasible to completely instrument a large group of airplanes in order to obtain the required statistical coverage in a reasonable interval of time.

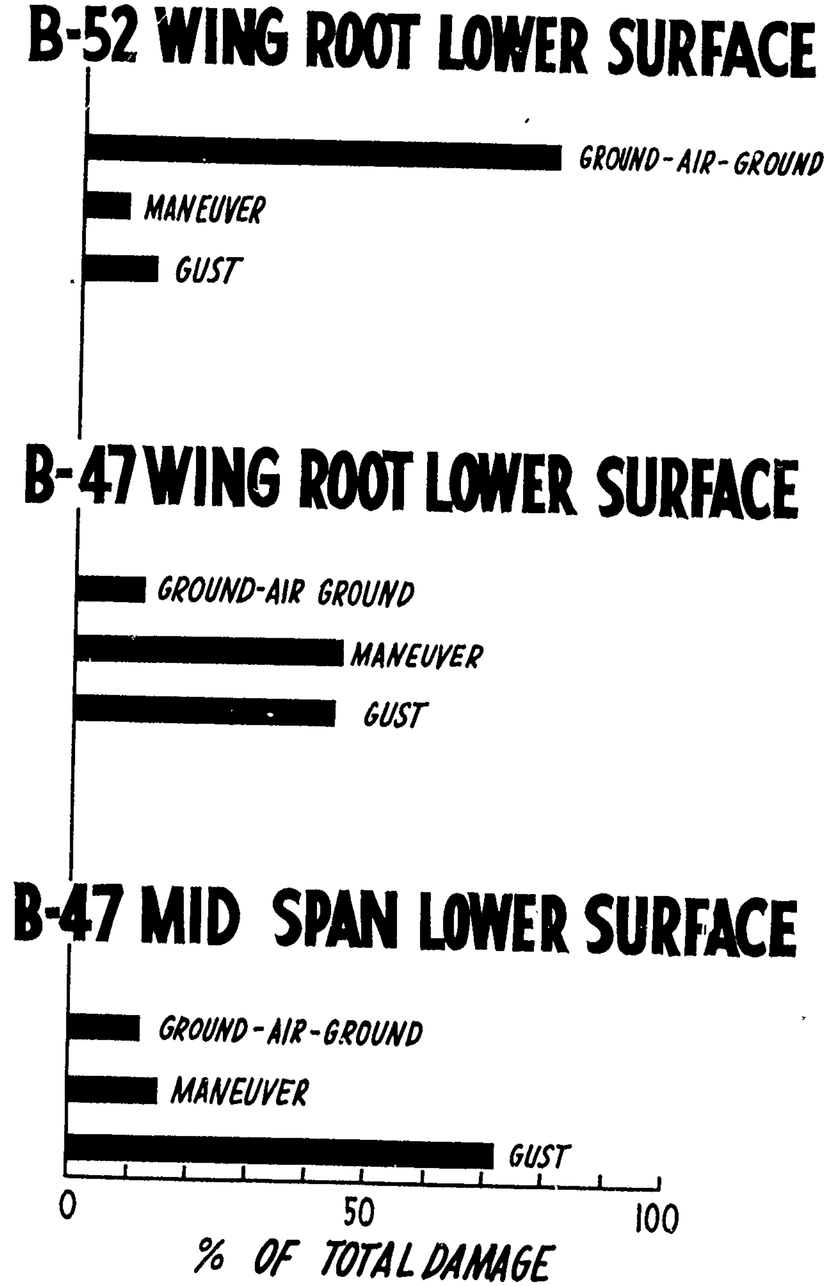


FIGURE 1

Relative Effects of Spectrum Loads on B-47 and B-52 Wing Lower Surface Fatigue Damage

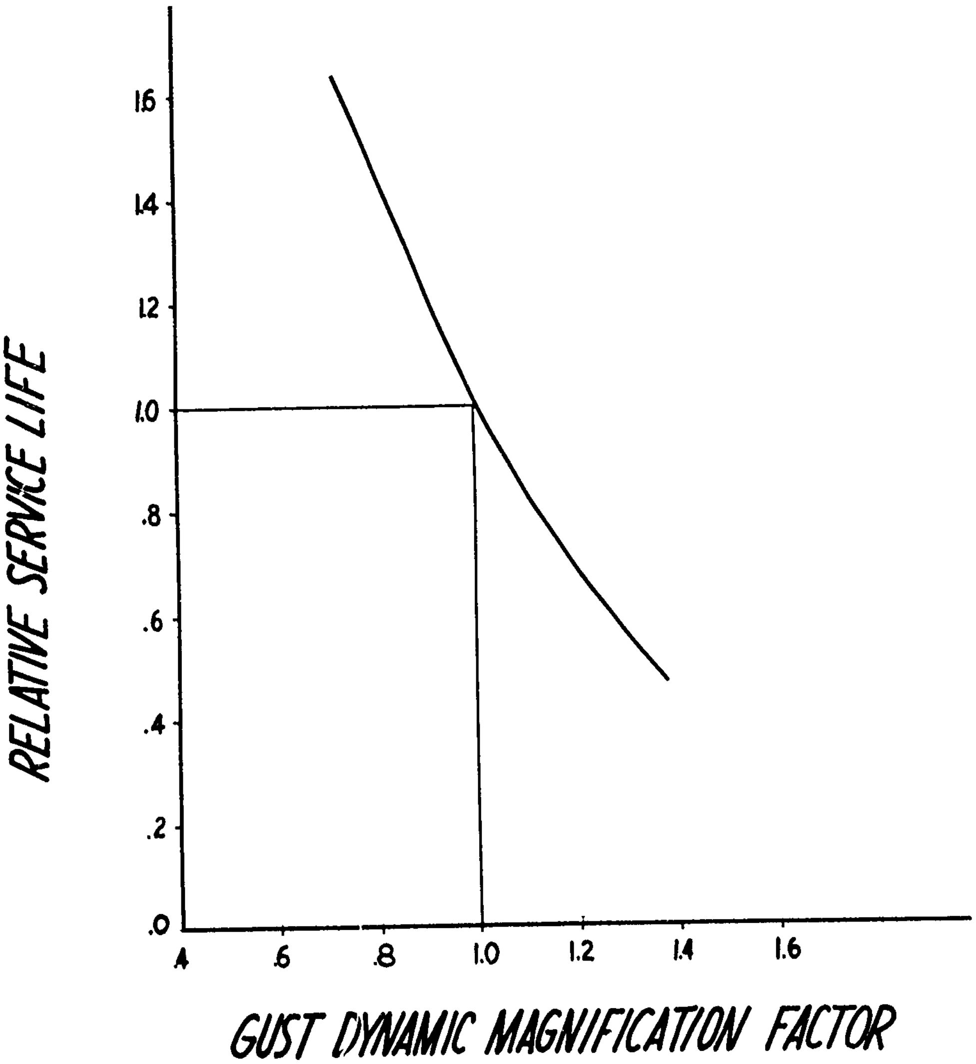


FIGURE 2

Effect of Gust Dynamic Magnification Factor on Relative Service Life of B-47 Wing Root Lower Surface

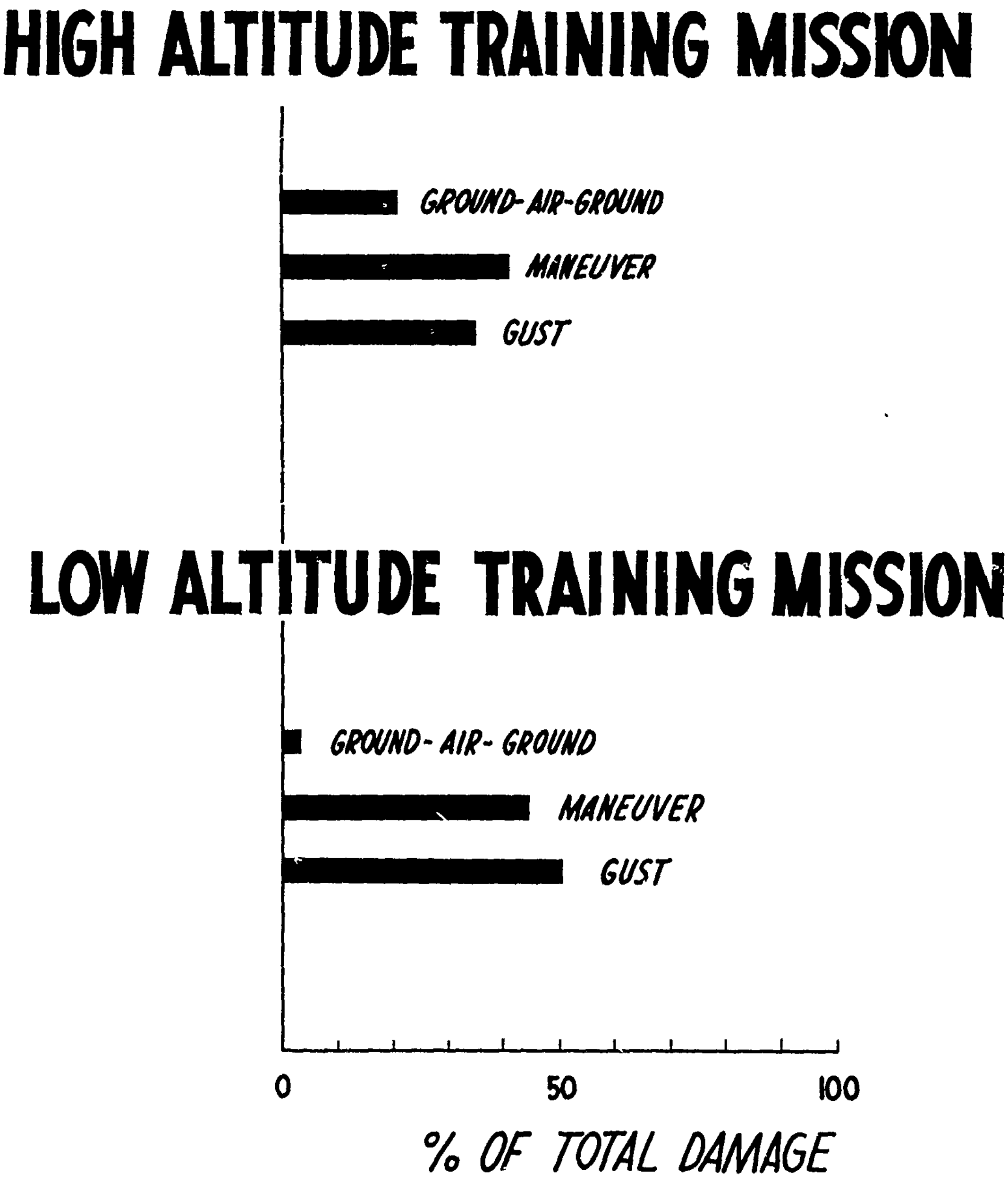


FIGURE 3

Relative Effects of Spectrum Loads on B-47 Wing Root Lower Surface Fatigue Damage for Two Mission Profiles

The general plan of the present test programs for obtaining the required load experience data on each type of aircraft (B-47, B-52 and KC-135) involves the use of one extensively instrumented control airplane and several fleet airplanes with minimum instrumentation installed. The control airplanes are instrumented to record, on FM tape, continuous time histories of internal structural loads on the wing, fuselage and empennage, gust velocities, accelerations at the c.g. and other points of interest on the aircraft. The following objectives were established for the control airplane test programs.

1. Provide transfer functions between vertical c.g. accelerations and the load frequency distribution at various points on the airplane for the range of gross weights, speeds and altitudes being flown by the fleet airplanes.
2. Provide a significant statistical sample of loads for specific flight conditions (low altitude for example).
3. Provide transfer functions between the internal structural loads and accelerations at various locations throughout the airframe and the turbulence encountered.

Using the transfer functions mentioned in objective 1, the large amount of VGH data being obtained from fleet airplanes of the same type as the control airplane can be converted to load frequency distributions. Since these airplanes are operated by Air Force crews for a large number of hours under a wide range of service conditions, the loads information resulting from this conversion yields the large statistically accurate sample required for the determination of the service life.

Accomplishment of objective 2 results in attainment of the most accurate load frequency distributions under the flight conditions which produce the most severe fatigue loads. Transfer functions for gust inputs obtained in accord with objective 3 make it possible to use gust data from other sources to increase the statistical accuracy of the expected gust loads.

DATA PROCESSING

In programs of this type much emphasis is placed on collecting and processing a large amount of data in a relatively short period of time. The calculation of peak count and load cycle frequency distributions from random load time histories and the determination of airplane transfer functions from gust velocity and airplane response time histories requires the use of high speed computers. Efficient and timely reduction of the large quantity of data involved to the desired form, requires rather elaborate plans for analysis of the data. The processes for handling the control airplane data from the present program have been condensed and outlined by the flow chart shown in Figure 4. Reduction of the fleet VGH data to c.g. acceleration peak count distributions is being accomplished elsewhere. It may be noted here that the data recording and data reduction methods used on the fleet airplanes will also be duplicated on the control airplane in order to account for possible differences in data results due to different data handling techniques.

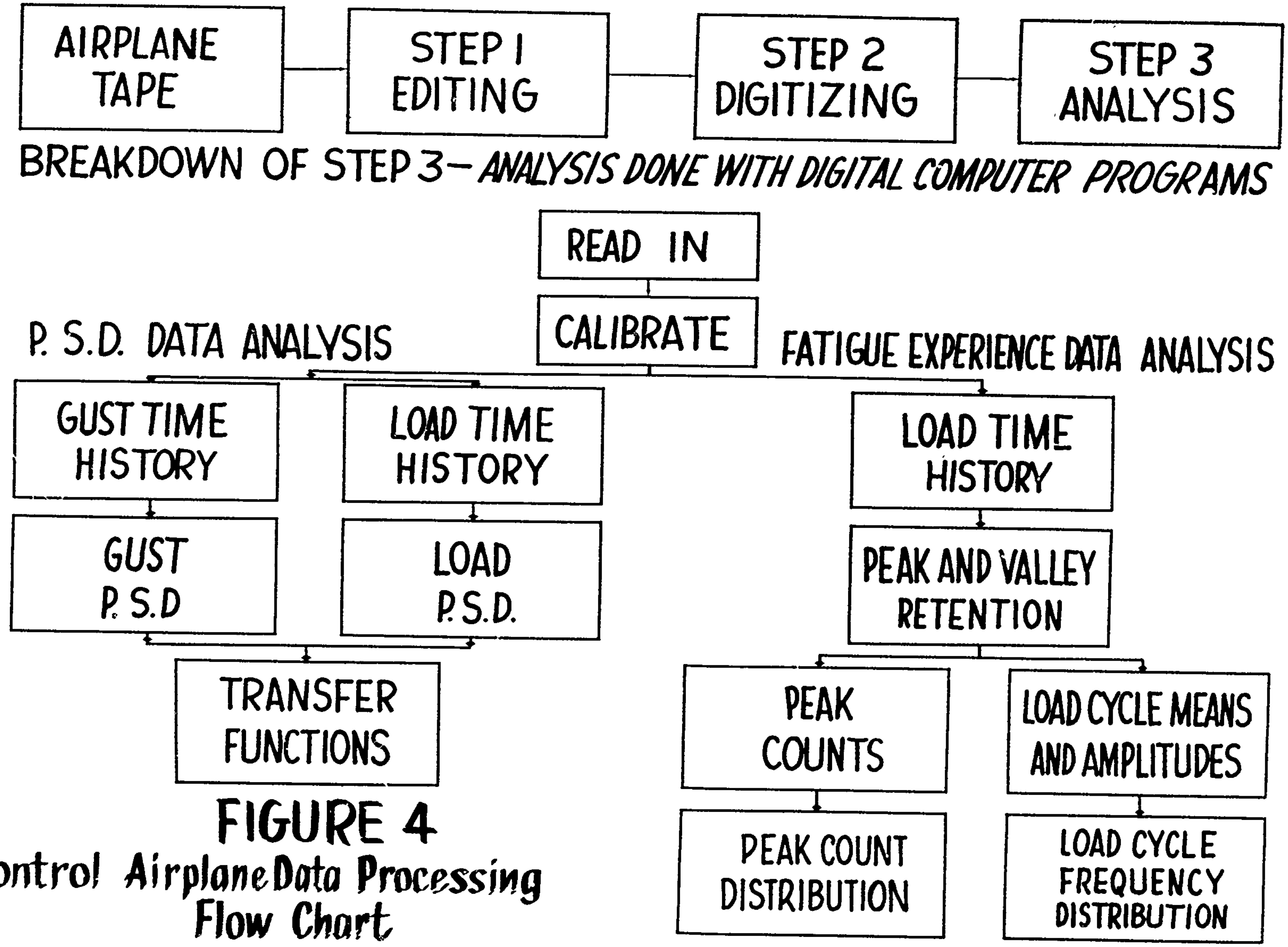


FIGURE 4
Control Airplane Data Processing Flow Chart

The first step in the reduction of the control airplane data includes a scan for discrepancies. Past experience has shown that it is highly desirable to complete the preliminary scan for one flight prior to running additional flights in order to insure against continuing record errors. The data are then categorized with respect to airplane configuration and flight condition. The data recorded on FM tape are identified with respect to airplane configuration and flight condition, beginning and end of record, calibration information, etc., by a tone signal which is recorded on a separate data track during the editing process. The second step of the data reduction process consists of converting the FM tape data to digital form. The previously mentioned tone signals are used to trigger an analog to digital converter at the beginning and end of each record. A sampling rate of 112.5 times per second is used in the conversion process for analysis of the load experience data. The digital data is then stored on metal tape in a form compatible with the Remington-Rand 1103A computer. The third and final step of the digital data reduction process consists of the actual calculations required to determine the peak count and load cycle frequency distributions and the airplane transfer functions. These calculations will be described in following sections of the paper.

POWER SPECTRAL DENSITY DATA ANALYSIS

In order to use the generalized harmonic analysis techniques for the prediction of airplane loads due to random turbulence, a knowledge of the airplane transfer function must be available. Therefore, the control airplane instrumentation, data recording devices, and data reduction methods are designed to yield simultaneous recordings of the gust velocity and airplane response time histories. The gust velocity and airplane response power spectra are computed using Tukey's procedures as outlined in References 1 and 2. These calculations are made for frequencies ranging from .2 cps to 10 cps which include the natural frequencies of all degrees of freedom considered to be significant in the definition of the primary structural load transfer functions. The length of the data sample used in the spectral density analyses is approximately five minutes. The sampling rate and number of power spectral estimates for these calculations may be adjusted as required. Knowledge of the airplane transfer functions allows both the conversion of other load experience to the program airplanes, and the conversion of the program airplanes load experience to a form useful in other fatigue analyses.

ANALYSIS OF LOAD EXPERIENCE DATA

Analysis of the load experience data consists primarily of reducing the random load time history to an idealized form that is more readily used in conventional fatigue testing or structural life calculations. It has been the practice to reduce random load history data to peak count or load cycles per unit time or distance flown and to tacitly assume that either interpretation of the data produces the same fatigue experience as the actual random load history.

The peak count idealization of random loads is obtained by measuring the greatest excursion between adjacent crossings of the load history mean

(see Figure 5). The amplitudes are computed by taking the difference between the values of these peaks or valleys and the load history mean, and these are then accumulated into predetermined amplitude bands. Each peak count is subsequently idealized as one half of a complete load cycle about the load history mean as shown by Figure 5. The peak count is then condensed to a tabulation of load frequency versus load amplitude. The peak count idealization is a highly simplified approximation of the actual load history.

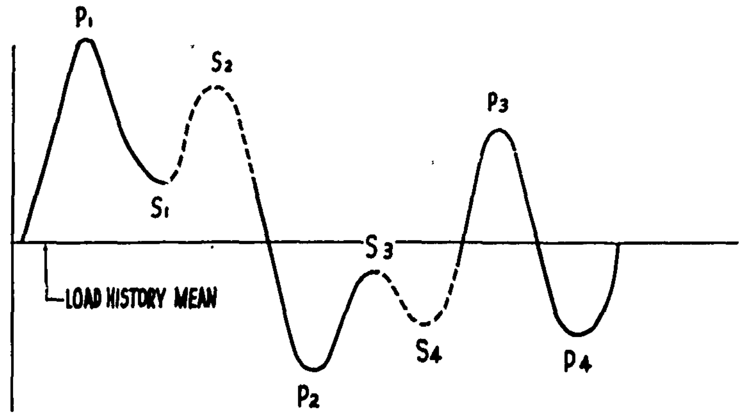
A more rational idealization is obtained by counting the primary and secondary cycles making up the random load history, as defined in Figure 6. The load cycle amplitude is computed by taking half of the difference between the algebraic values of the peak and valley making up the cycle. The mean load level for each cycle is obtained by taking half of the sum of the algebraic values. It is important to note that all significant parts of the actual load history are retained by the load cycle concept. The load cycle equivalent of a given random history may be conveniently presented as a tabulation of load cycle frequency versus amplitude and load cycle mean.

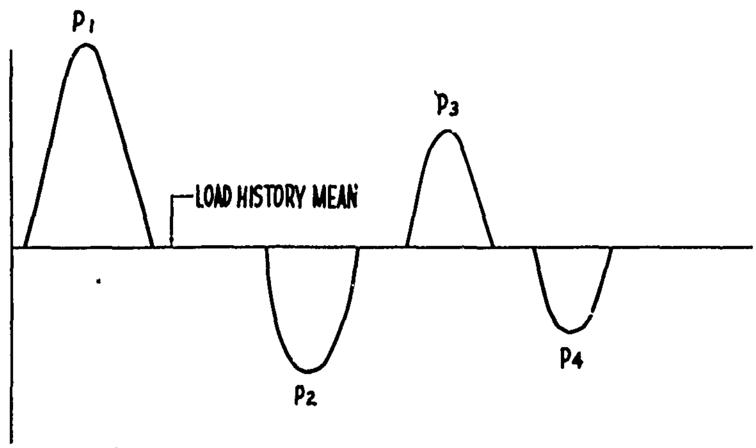
The program used to analyze the random time histories for the control airplanes is designed to provide both peak count and load cycle idealizations of the actual flight experience. The output of this program consists of a tabulation in the form of peak count and load cycle frequency distributions for each airplane configuration and flight condition. The frequency distributions given in Tables 1 and 2 are for a typical sample of low altitude data recorded at Wing Station 315 on the B-47 control airplane. Corresponding plots of the load cumulative frequency versus load level are shown in Figure 7. In order to compare these two interpretations, the sum of the load cycles without regard to their means has also been plotted versus amplitude. It can be seen that this sum is nearly equal to the load cycle cumulative frequency obtained from the peak count interpretation in the low amplitude range, but is substantially different in the high amplitude range. The fact that the plot of the total summation of load cycles lies below the plot of peak counts for large amplitudes is evidently due to the low probability of encountering a large negative disturbance immediately following a large positive one, or vice versa. The peak count idealization does not retain this characteristic of the data.

From Figure 7 it can be concluded that the two idealizations of the random load history may result in substantial differences in service life estimates. At present, comparative results are not available.

CONVERSION OF FLEET PEAK COUNT TO LAOD CYCLE FREQUENCY DISTRIBUTION

Of particular interest is the technique for the conversion of the large sample of c.g. acceleration peak count data obtained from the fleet airplanes instrumented with VGH recorders to peak count distributions and load cycle frequency distributions. As previously mentioned, one of the primary objectives of the control airplane test program is to obtain the necessary transfer relations to make this conversion. It may be noted that this technique can be applied to either lateral or vertical loads and all of the flight, landing, and taxi condi-

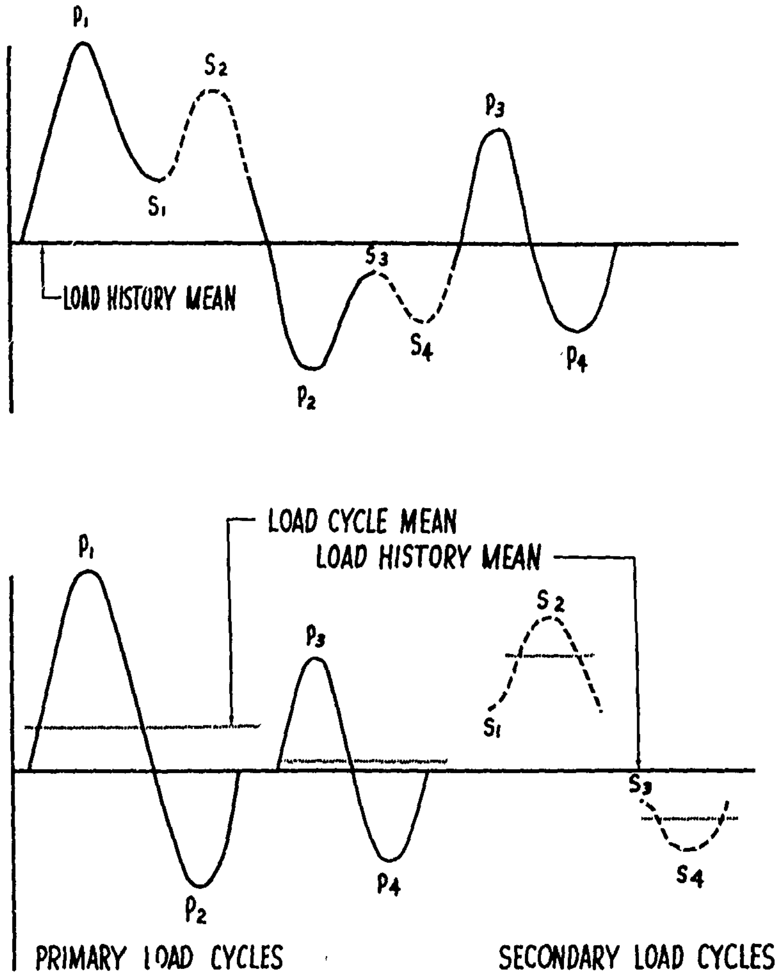




P - *PRIMARY PEAK OR VALLEY*
S - *SECONDARY PEAK OR VALLEY*

FIGURE 5

Random Load Time History and Corresponding Peak Count Idealization



P- PRIMARY PEAK OR VALLEY
S- SECONDARY PEAK OR VALLEY

FIGURE 6

Random Load Time History and Corresponding Load Cycle Idealization

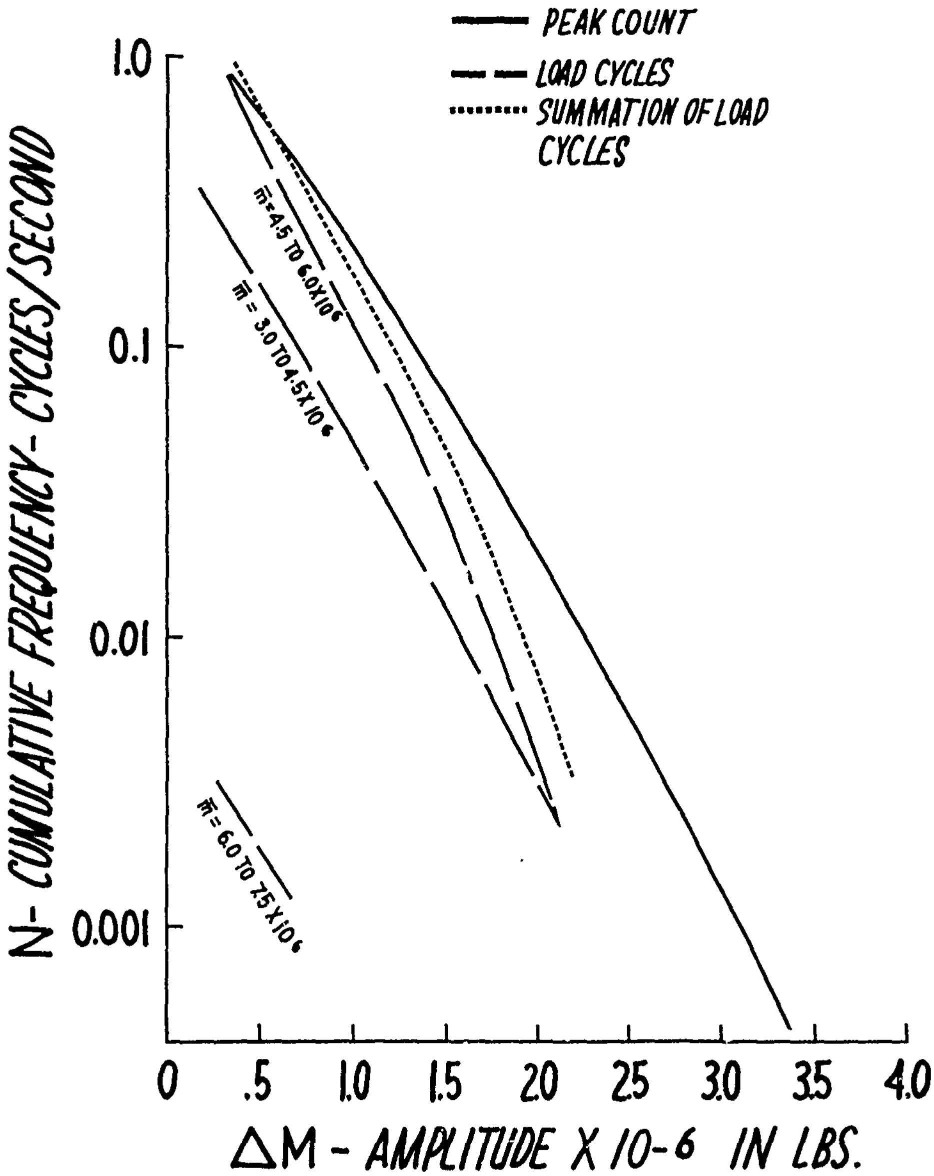


FIGURE 7

Comparison of Load Cycle and Peak Count Data

tions. The control airplane load experience data are collected under standard fleet operational conditions in order to minimize load experience differences due to pilot technique, fuel sequencing, weather, geographical location, and others. It is felt that the accuracy of the conversion of fleet c.g. acceleration peak count to load cycle frequency distributions depends strongly on the similarity of operating conditions for the fleet and the control airplane.

Since the fleet VGH data may differ from that recorded on the control airplane, due to different data resolution and reduction techniques, typical samples of control airplane c.g. acceleration data are reduced to peak counts by the same techniques used to process the fleet VGH data. The difference in the peak counts for the typical control airplane data obtained by using the automatic digital program used to reduce the control airplane data, and the process used to reduce the fleet VGH data, represents the reading correction to be applied to fleet c.g. acceleration peak counts. It is estimated that only a relatively small sample of control airplane data is required to define the reading correction. Figure 8 illustrates typical differences in c.g. acceleration peak counts and a corresponding reading correction obtained from control airplane data when reduced to peak counts by the two methods.

Denoting the VGH peak count distribution by $f_v(\Delta n)$ and the corresponding machine peak count distribution by $f_m(\Delta n)$, a reading correction may be defined

$$C(\Delta n) = \frac{f_m(\Delta n)}{f_v(\Delta n)} \tag{1}$$

Having defined the reading correction, the corrected VGH peak count for the fleet may then be computed by

$$f_F(\Delta n) = C(\Delta n) f_{FU}(\Delta n) \tag{2}$$

where $f_F(\Delta n)$ is the corrected fleet c.g. acceleration peak count distribution

$f_{FU}(\Delta n)$ is the uncorrected fleet c.g. acceleration peak count distribution

One method of estimating the fleet repeated load experience from fleet c.g. acceleration peak counts is to scale the control airplane load peak counts with the ratio of the corrected c.g. acceleration to the control airplane c.g. acceleration. This ratio may be written as:

$$R(f) = \frac{\Delta n_F(f)}{\Delta n_c(f)} \tag{3}$$

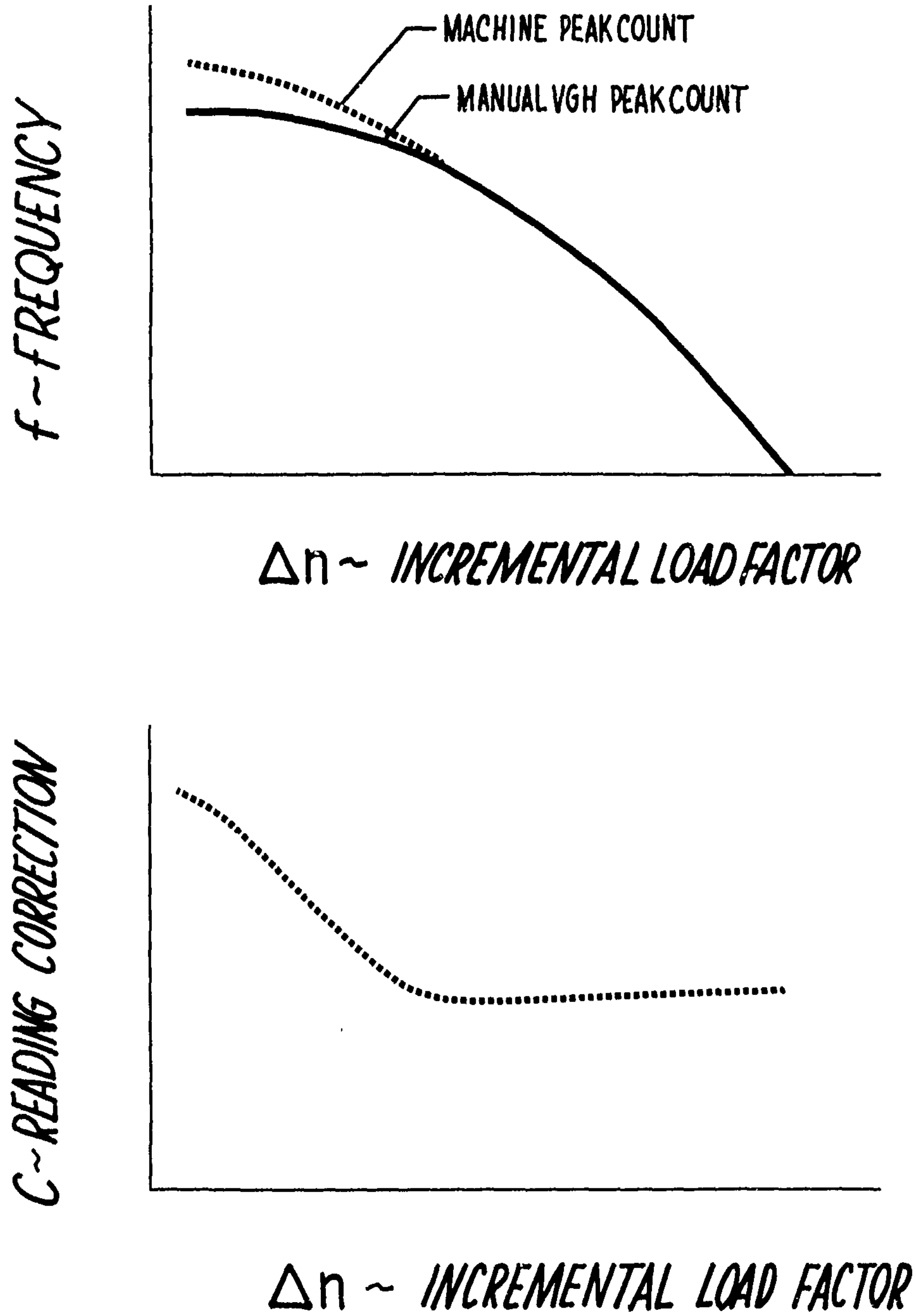


FIGURE 8

Typical Machine and Manual Peak Counts and the Corresponding Reading Correction

where $\Delta n_F(f)$ and $\Delta n_C(f)$ are the incremental c.g. accelerations for the fleet and control airplanes as functions of the frequency of occurrence

Typical plots of control airplane and corrected fleet c.g. acceleration peak count and ratio defined by equation (3) are shown in Figure 9. Denoting the peak count frequency distribution for the control airplane by $F_C(f, \Delta M_C)$ the corresponding fleet peak count frequency can be written as $F_F(f, \Delta M_F)$ (see tables 1 and 2)

where $$\Delta M_F(f) = R(f)\Delta M_C(f) \tag{4}$$

A commonly used concept in converting from vertical c.g. accelerations to wing loads is that of a dynamic magnification factor, D. A definition of D which may be useful for fatigue load analyses is,

$$D(f) = \frac{\Delta M_M(f)}{\Delta M_S(f)} \tag{5}$$

where $\Delta M_M(f)$ is the measured incremental loads as a function of the frequency of occurrence

and $\Delta M_S(f)$ is the incremental load computed from the incremental c.g. accelerations, $\Delta n(f)$ as a function of the frequency of occurrence

The expression $\Delta M_S(f)$ can be computed by

$$\Delta M_S(f) = Q\Delta n(f) \tag{6}$$

where Q is the load per unit c.g. acceleration under steady state conditions

The c.g. acceleration, $\Delta n(f)$, and the loads, $\Delta M_M(f)$, can be computed from data recorded simultaneously on the control airplane. Furthermore the static load per unit c.g. acceleration can be obtained either from flight tests or calculations. This permits the computation of the dynamic magnification factor, D, for the control airplane loads.

It is reasonable to assume that the D for the fleet airplanes is the same as for the control airplane at the same gross weight and flight condition. Thus the loads on the fleet airplanes can be computed as

$$\Delta M_F(f) = D(f)\Delta M_{FS}(f) \tag{7}$$

TABLE 1

PEAK COUNT DISTRIBUTION

AMPLITUDE x 10^-6 IN. LBS.		PLUS	MINUS
	0.3 to 0.9	585	602
	0.9 to 1.5	204	198
	1.5 to 2.1	69	66
	2.1 to 2.7	17	10
	2.7 to 3.3	3	3
	3.3 to 3.9	1	0

DATA TIME- 1151 SECONDS

TABLE 2

LOAD CYCLE FREQUENCY DISTRIBUTION

MEAN x 10^-6 IN-LBS.		AMPLITUDE X10^-6 IN.-LBS.				
		0.3 to 0.9	0.9 to 1.5	1.5 to 2.1	2.1 to 2.7	2.7 to 3.3
	1.5 to 3.0	0	0	0	0	0
	3.0 to 4.5	3	0	0	0	0
	4.5 to 6.0	664	154	27	3	0
	6.0 to 7.5	200	61	12	3	0
	7.5 to 9.0	0	0	0	0	0
	SUMMATION	867	215	39	6	0

DATA TIME-1151 SECONDS

Substituting into equation (7) from equations (5) and (6) combining terms and recognizing that $\Delta M_{M}(f) = \Delta M_{c}(f)$ results in an expression identical to equation (4). It can be seen therefore that the two concepts are equivalent. However the ratio system given in equations (3) and (4) can be applied to loads for which Q can not be readily determined, and its application required less work so that this approach is desirable unless specific knowledge of the dynamic magnification factor is required.

Conversion of fleet c.g. acceleration peak count to a load cycle frequency distribution requires more detailed treatment. Beginning with fleet c.g. acceleration peak count, corrected for the reading correction defined by Equation (1), it is then necessary to account for differences in the amount of time spent in turbulence of a given intensity for the control and fleet airplanes, and to adjust the load cycle amplitudes and mean values accordingly. A method of accomplishing this conversion is suggested by Press, Meadows and Hadlock in Reference 3, by their semi-graphical approach to the calculation of turbulence intensities encountered during flight from measured incremental c.g. accelerations.

It is assumed that the turbulence encountered varies only in intensity, and while the over all distribution of gust velocities is non-Gaussian, the actual distribution can be closely approximated by the summation of a number of properly weighted Gaussian components. The weighting factor associated with each component is proportioned to the fraction of the total time spent in that component having a root-mean-square gust velocity of σ_{v_i}. The number of exceedances of given values of Δn per second can be expressed

$$N(\Delta n) = N_0 \sum_{i=1}^{n} P_i \, e^{-(\Delta n)^2/2\sigma_i^2} \tag{8}$$

where Δn is the given response level

σ_i is the rms value of the i^{th} Gaussian component of response

P_i is the fraction of the total time spent in the Gaussian component having an rms value of σ_i

N_0 is the number of crossings per second of the zero response level with positive slope. (a constant for a given airplane configuration and flight condition)

$N(\Delta n)$ is the number of times per second that the given response level, Δn is crossed with positive slope.

It should be noted that $N(\Delta n)$ is also the cumulative frequency of reaching or exceeding given values of Δn and may be approximated by cumulative summation of the Δn peak count.

Experimental data collected to date on the control and fleet airplanes indicate that the distributions of c.g. accelerations resulting from turbulence can be closely approximated by two Gaussian components. For identical configurations and flight conditions the number of times per second that the c.g. acceleration exceeds given values of Δn for the control airplane is therefore

$$N_C(\Delta n) = N_{1c}(\Delta n) + N_{2c}(\Delta n) \tag{9}$$

where

$$N_{1c}(\Delta n) = N_o P_{1c} e^{-(\Delta n)^2/2\sigma_{1c}^2} \tag{10}$$

$$N_{2c}(\Delta n) = N_o P_{2c} e^{-(\Delta n)^2/2\sigma_{2c}^2} \tag{11}$$

while for the fleet airplanes

$$N_F(\Delta n) = N_{1F}(\Delta n) + N_{2F}(\Delta n) \tag{12}$$

where

$$N_{1F} = N_o P_{1F} e^{-(\Delta n)^2/2\sigma_{1F}^2} \tag{13}$$

$$N_{2F} = N_o P_{2F} e^{-(\Delta n)^2/2\sigma_{2F}^2} \tag{14}$$

For each of the above Gaussian components, the plot of $\ln N(\Delta n)$ versus $(\Delta n)^2$ is a straight line having a slope $-1/2\sigma^2$ and vertical axis intercept $\ln N(\Delta n) = N_o P$ at $\Delta n = 0$. The slopes of the lines are implicitly an indication of the turbulence intensity, assuming linearity between disturbance and response. The vertical axis intercepts are proportional to the fraction of total time spent in each of the two turbulence intensity levels, since N_o is a constant for a given airplane configuration and flight condition.

Using the two line approximation for the plot of $\ln N(\Delta n)$ versus $(\Delta n)^2$ for experimental data recorded on the control and fleet airplanes, the preliminary steps for obtaining fleet load cycle distributions from fleet c.g. acceleration peak count, control airplane c.g. acceleration peak count and load cycle data, are carried out in the following manner:

1. For each airplane configuration and flight condition, separate plots are made of the cumulative frequency of the c.g. accelerations for the control and fleet airplanes in the form $\ln N(\Delta n)$ versus $(\Delta n)^2$ as indicated in Figure 10.
2. Each curve is approximated by two tangents adjusted such that each pair of tangents intersect at the same value of $(\Delta n)^2$. This abscissa should correspond to a boundary between two load cycle amplitudes in the desired load cycle frequency distribution (based on incremental moment per g for example).
3. The portions of the curves to the left and right of $(\Delta n_I)^2$ are considered separately. The ratios σ_{1F}/σ_{1C} and σ_{2F}/σ_{2C} necessary to account for differences in turbulence intensity encountered by the control and fleet airplanes are obtained from the ratio of the slopes of the tangents to the control and fleet curves. The ratios P_{1F}/P_{1C} and P_{2F}/P_{2C}, necessary to account for the difference in time spent in disturbed flight between the control and fleet airplanes are obtained by the ratio of the vertical axis intercepts of the tangents to the control and fleet curves.
4. The load cycle frequency distribution for the control airplane is then divided into two distributions so that load cycle amplitudes below Δn_I are contained in one distribution and those load cycle amplitudes above Δn_I are contained in the other.

Each of the control airplane load cycle frequency distributions is then used to obtain a corresponding fleet load cycle frequency distribution. If the load time history for the control airplane is given by

$$M_c(t) = \Delta M_c(t) + m \tag{15}$$

where $\Delta M_c(t)$ is the time variant incremental load and m is the 1 g straight and level flight mean load.

The fleet airplane load time history for a different turbulence intensity may be written

$$M_F(t) = \frac{\sigma_F}{\sigma_c}\left[\Delta M_c(t)\right] + m \tag{16}$$

The load cycle amplitude and mean for the fleet airplanes may now be computed by the following expressions.

Load cycle amplitude, $$\Delta M = \frac{M(t_1) - M(t_2)}{2} \tag{17}$$

Load cycle mean, $$\bar{m} = \frac{M(t_1) + M(t_2)}{2} \tag{18}$$

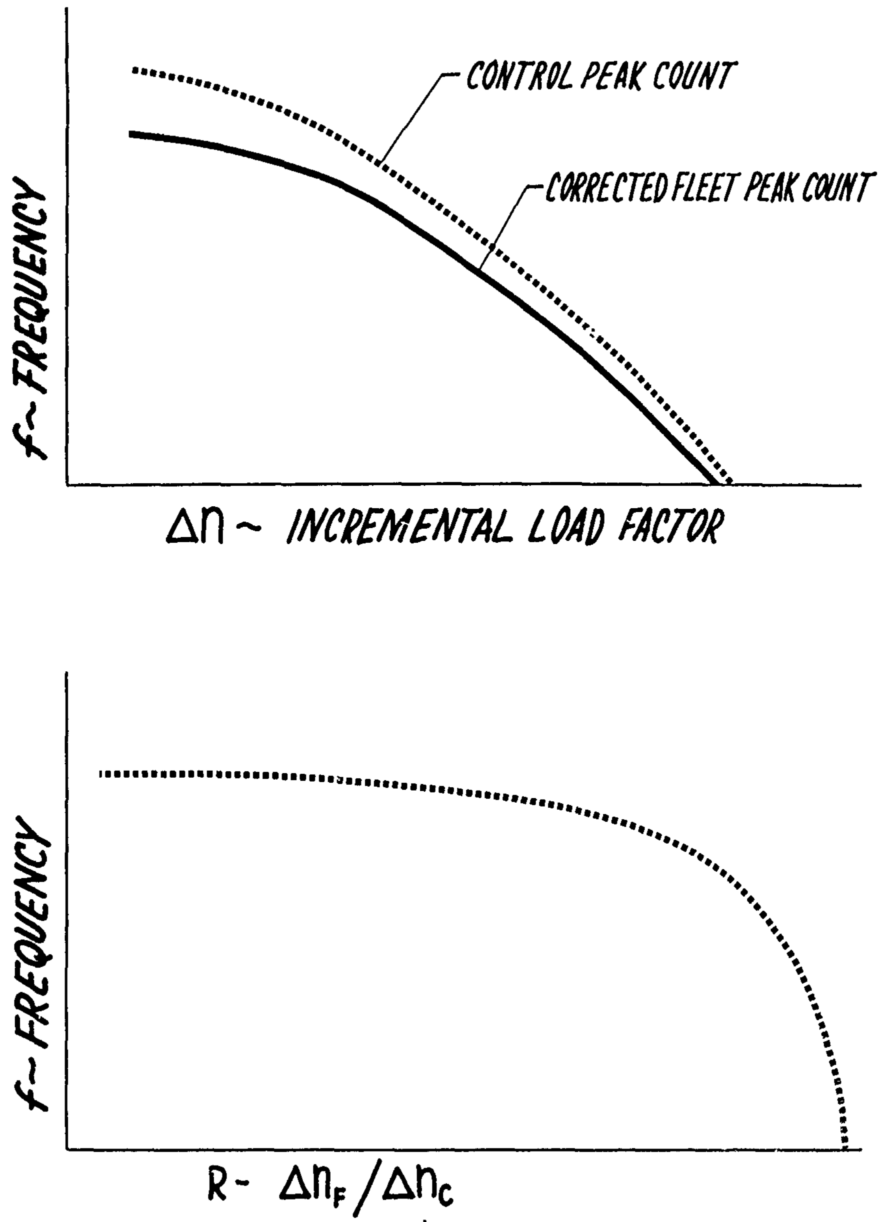


FIGURE 9

Control and Fleet Center of Gravity Acceleration Peak Count Data and the Corresponding Ratio

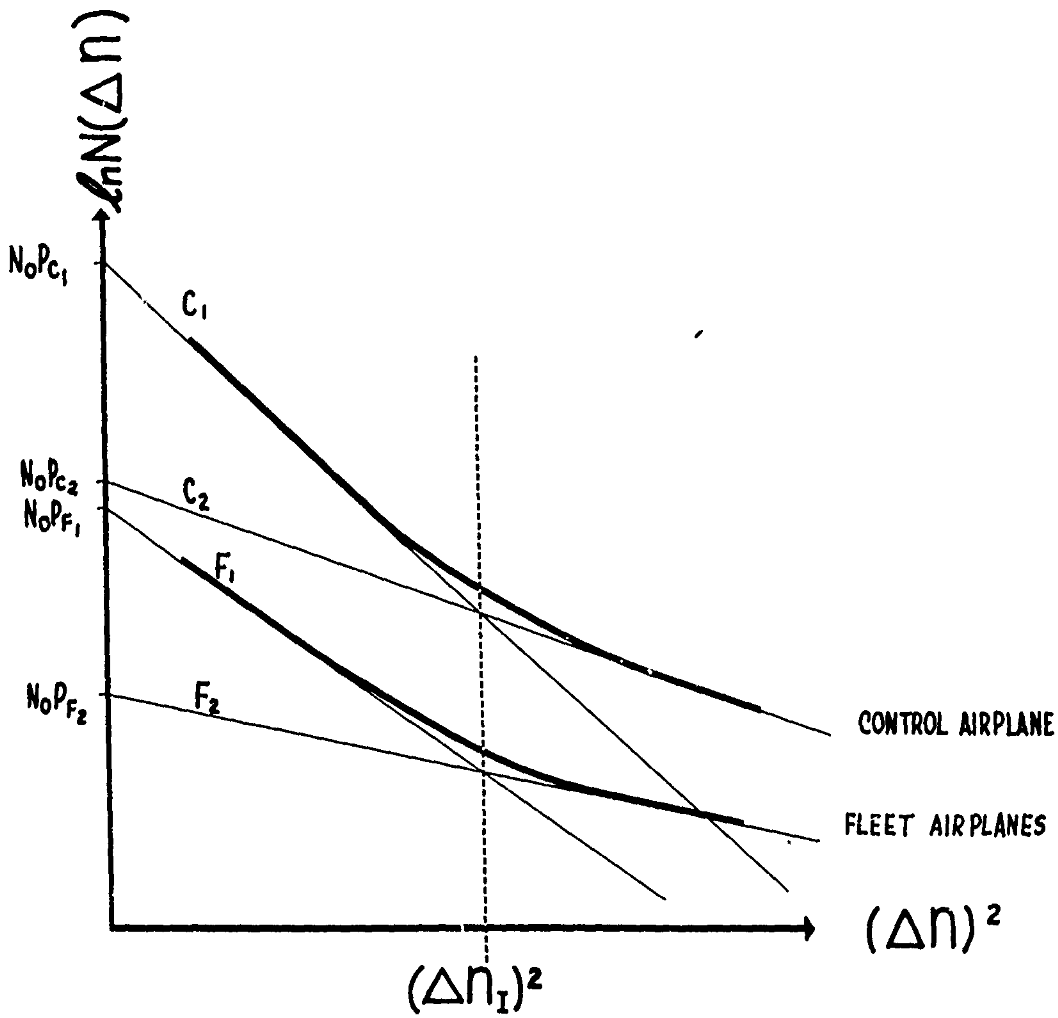


FIGURE 10

Approximation of C.G. Acceleration Cumulative Frequency Distribution by Two Gaussian Components

Substituting equations (15) and (16) into (17) and (18) and dividing the fleet load cycle amplitude and mean by the control airplane load cycle amplitude and mean respectively we obtain

$$\frac{\Delta M_F}{\Delta M_c} = \frac{\sigma_F}{\sigma_c} \tag{19}$$

and

$$\frac{\overline{m}_F}{\overline{m}_c} = \frac{\sigma_F}{\sigma_c} + \frac{m}{\overline{m}_c}\left(1 - \frac{\sigma_F}{\sigma_c}\right) \tag{20}$$

Finally, these ratios when applied to each of the control airplane load cycle frequency distributions, $F(f_{1c}, \Delta M_{1c}, \overline{m}_{1c})$ for $\Delta n < \Delta n_I$ and $F(f_{2c}, \Delta M_{2c}, \overline{m}_{2c})$ for $\Delta n > \Delta n_I$ yields the corresponding fleet load cycle frequency distributions

$$F(f_{1F}, \Delta M_{1F}, \overline{m}_{1F}) \quad \text{for} \quad \Delta n < \Delta n_I$$

and

$$F(f_{2F}, \Delta M_{2F}, \overline{m}_{2F}) \quad \text{for} \quad \Delta n > \Delta n_I$$

where for $\Delta n < \Delta n_I$

$$f_{1F} = \frac{P_{1F}}{P_{1c}} f_{1c} \tag{21}$$

$$\Delta M_{1F} = \frac{\sigma_{1F}}{\sigma_{1c}} \Delta M_{1c} \tag{22}$$

$$\overline{m}_{1F} = \frac{\sigma_{1F}}{\sigma_{1c}} \overline{m}_c + m\left(1 - \frac{\sigma_{1F}}{\sigma_{1c}}\right) \tag{23}$$

and for $\Delta n > \Delta n_I$

$$f_{2F} = \frac{P_{2F}}{P_{2c}} f_{2c} \tag{24}$$

$$\Delta M_{2F} = \frac{\sigma_{2F}}{\sigma_{2c}} \Delta M_{2c} \tag{25}$$

$$\overline{m}_{2F} = \frac{\sigma_{2F}}{\sigma_{2c}} \overline{m}_c + m\left(1 - \frac{\sigma_{2F}}{\sigma_{2c}}\right) \tag{26}$$

It should be pointed out the resulting mean bands and amplitude bands of the fleet data will generally be different than those selected for the control airplane data categorization. It will often be desirable to realign the fleet data bands so that they are the same as those of the control airplane. This can be

easily accomplished graphically. The procedure will not be covered in this paper.

Examination of typical samples of data indicate that discrepancies between the two-line approximation of $N(\Delta n)$ and the correct value are less than 10 percent. Discrepancies between a single straight line approximation of $N(\Delta n)$ may be as large as 100 percent for some values of $(\Delta n)^2$. The extension of the two segment approximation of the plot of $\ln N(\Delta n)$ versus $(\Delta n)^2$ to three or more segments is straight forward.

OTHER DATA FROM THE TEST PROGRAM

The primary purposes of the present program, are to obtain the airplane fleet repeated load experience data in terms of load cycle frequency distributions and to obtain experimental transfer functions relating atmospheric turbulence and airplane response. However, additional information of general interest may be obtained as by-products of the program. Some of these are:

1. Derived gust velocity experience,
2. Distribution of atmospheric turbulence intensity,
3. Power spectra of atmospheric turbulence.

Some data on the above items have already been collected and are of sufficient general interest to be presented here.

The derived gust velocity experience may be determined by converting the airplane center of gravity c.g. acceleration occurrences to gust encounters through the well known discrete gust loads formula Reference 4. The derived gust velocity experience for the B-47 and B-52 control airplanes during a portion of the total low altitude flight time is shown in Figure 11. Shown for comparison are low altitude flight data obtained on the Meteor, Mk 7, Reference (5), and preliminary unpublished data obtained by the National Aeronautics and Space Administration on an F9F in flight at 300 feet absolute altitude. The flight programs were conducted under similar flight conditions. It can be seen that the general agreement of the data collected is good.

A method for determining the distribution of the atmospheric turbulence intensity from airplane response data is outlined in Reference 3. Using this method, a probability density distribution of root-mean-square gust velocities encountered by the B-47 control airplane during low altitude flight was computed, and is shown in Figure 12. The distribution was derived from beam bending moment peak count data obtained for Wing Stations 120 and 315. Theoretical values of the airplane response to random turbulence (A and N_0) were used in the calculations. Probability density distributions of gust intensities for three altitude ranges published in Reference 3 are also shown for comparison. Data taken from the c.g. were not used because of an apparent discrepancy in the theoretical N_0. It is believed that this discrepancy is due to over estimating the effect of a body vertical bending mode at a frequency of about 4.5 cps. With this effect supressed, good agreement between the σ_u distributions computed from the measured wing bending moments c.g. accelerations is expected. Figure 13 shows a comparison of the B-47 low altitude probability

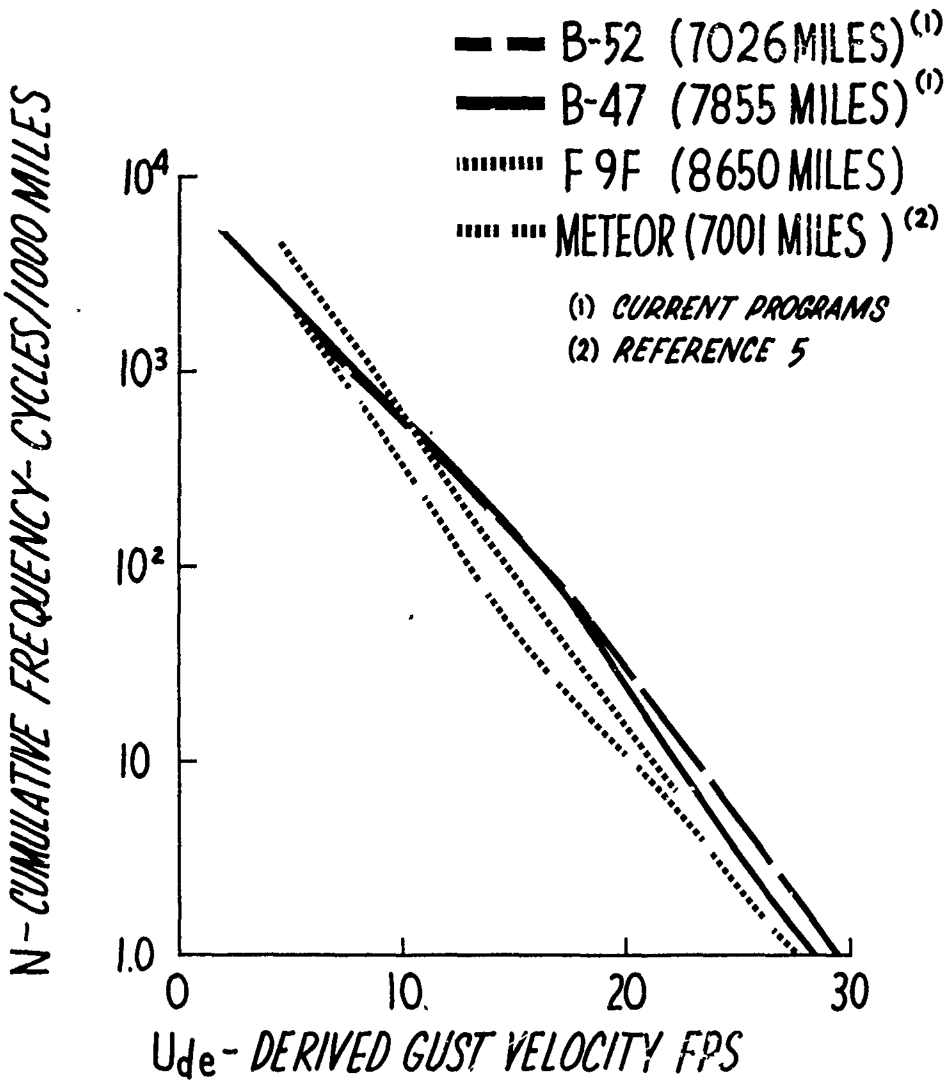


FIGURE 11

Low Altitude Derived Gust Velocity Cumulative Frequency Distribution

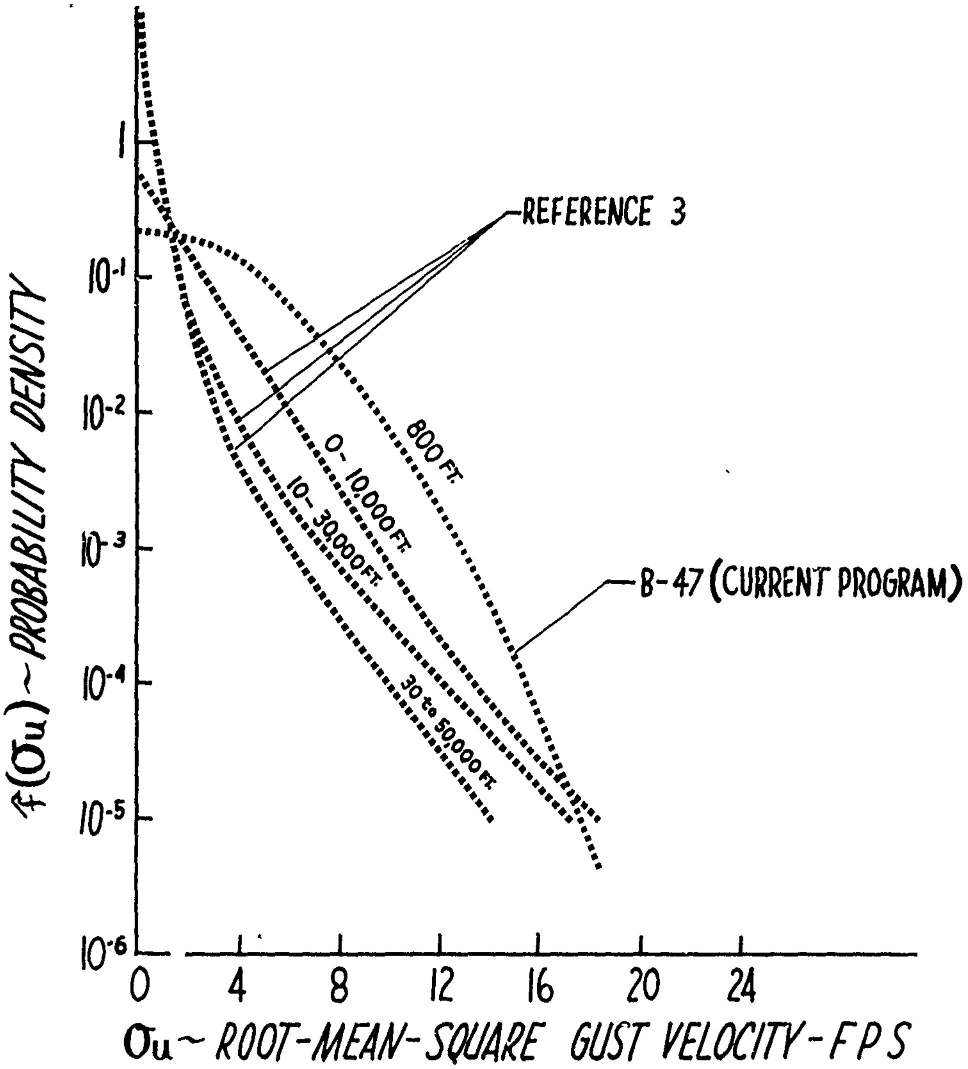


FIGURE 12

Probability Density Distribution of the Root-Mean-Square Gust Velocity for Various Altitudes

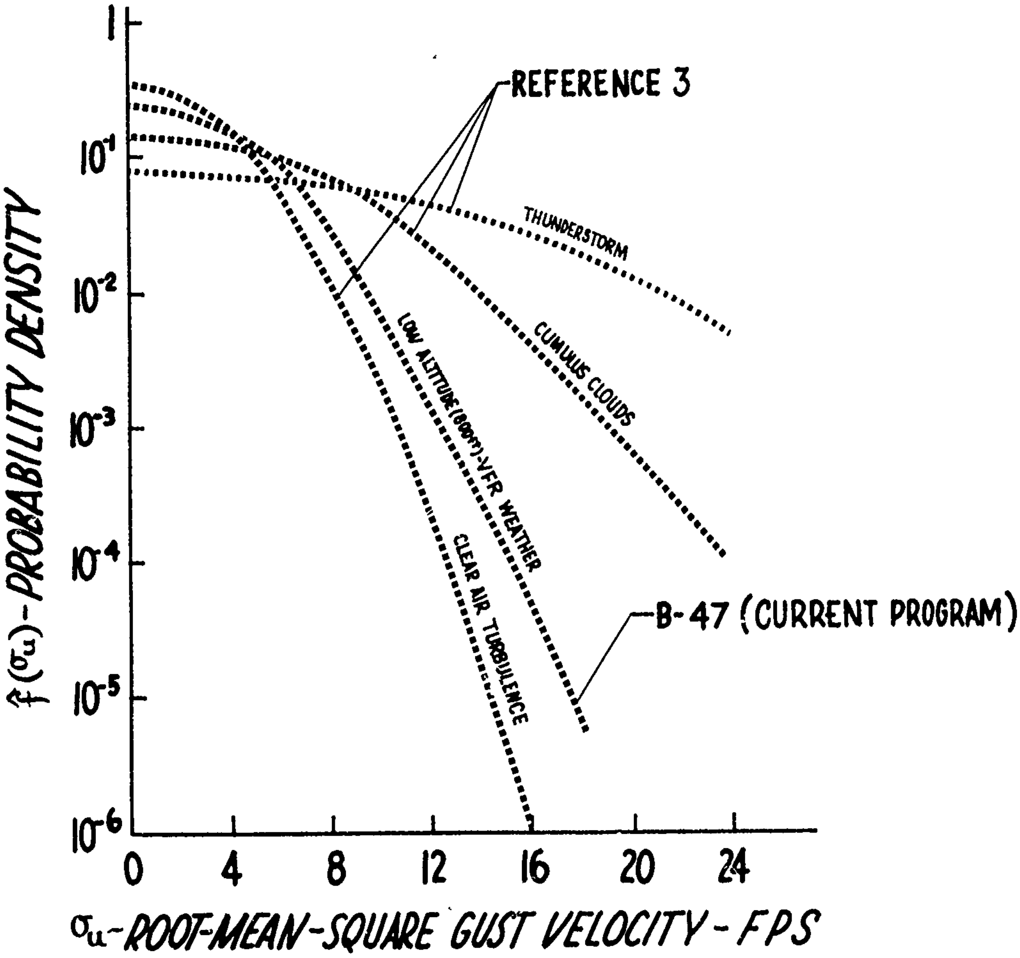


FIGURE 13

Probability Density Distribution of Root-Mean-Square Gust Velocity for Various Weather Conditions

density distribution of gust intensity to those presented in the above reference for clear air turbulence, cumulus cloud turbulence and thunderstorm turbulence. The B-47 control airplane low altitude flights were conducted under VFR flight conditions only.

At this time, the reduction of data collected during this program has not progressed far enough to provide specific information regarding the nature of the turbulence spectrum encountered.

AREAS FOR FUTURE RESEARCH

The results of programs undertaken to date have provided valuable data for use in fatigue prevention studies. The work has also pointed up areas which may require future research. Two areas in particular stand out.

The data comparisons made so far show a substantial difference in load occurrences obtained from the peak count and load cycle representations. This difference has been noticeable without exception in the loads data analyzed to date. These data are from all parts of the airplane and for all speed and weight ranges. It is reasonable to expect that these differences will also be displayed in other aircraft. The current theory permits the calculation of exceedances (only approximately the peak values). It may be desirable therefore to find methods of extending the theory to provide a more refined interpretation of the random load experience.

Examination of the data available from the current programs discloses that the zero load intercept in cumulative frequency plots varies from flight to flight for the same weight and flight condition. Two possible explanations are evident. A variation in the shape of the atmospheric input spectrum could account for this discrepancy. Another possible explanation, as suggested earlier in this paper, is that the amount of time spent in disturbed flight is a variable. Programs now underway will shed some light on the cause of these differences. Reviews of data available might help to determine the extent of the phenomena with regard to airplane types, flight and atmospheric conditions.

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DATA PROCESSING FOR THE USAF-USN-NASA STATISTICAL LOADS PROGRAM

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ABSTRACT

A study has been made of the equipment and procedures required to process automatically the data from special magnetic tape recorders mounted in a large number of service aircraft. The data consist of continuous recordings of accelerations along, and angular velocities about, the three body axes of the aircraft, together with impact and static air pressures. At least 30,000 flight hours per year will be recorded. The purpose of this statistical program, as originally established by the NASA, is to provide realistic maneuver-loads criteria for use in the design of new aircraft as well as service life limits for existing aircraft. A single computing facility may serve the needs of both the U. S. Air Force and Navy.

Processing includes preliminary editing, correction of measured accelerations to the center of gravity of the aircraft, and derivation of additional parameters. Descriptive statistics are required for 19 quantities, as histograms of

(a) distributions of peaks classified for correlation
(b) time distributions
(c) time-of-dwell distributions

and envelopes of extreme values. Accumulated statistics are kept separated for each aircraft type and mission category. Each flight run results in a statistical file of about 14,000 entries.

OBJECTIVES OF THE PROGRAM

Aircraft must be designed to withstand the loads imposed on their structures during maneuvers and while flying through turbulent air. The need for adequate design criteria based on actual experience has become increasingly acute with the accelerated demands for higher performance in modern aircraft, particularly in terms of speed, increasing weight, and load of accessory equipment. Statistics are required for the entire spectrum of loads encountered by aircraft in regular service, ranging from extreme loads, which can cause immediate damage, through lesser loads which may occur frequently enough to cause fatigue failure in an

unacceptably short time. The distribution of various types of loads of various magnitudes will depend on many factors, including the human factor - the response of the pilot to the aircraft and the situation. The military services are particularly concerned with the group of operating conditions defining a mission for a given type of aircraft. Thus, the aircraft should be designed for the mission or group of missions it will fly. Statistics are to be kept separated by "category", a category consisting of all flights alike as to aircraft type, mission, and other factors expected to be significant statistically. A given aircraft type may operate in several classes of missions, and hence in several categories.

The U. S. Navy and Air Force* have for some years maintained programs for acquiring such statistics by means of airborne recording equipment. Statistical accelerometers have been employed, to count the occasions when the normal load factor has exceeded specified values, and to separate these counts according to several ranges of air speed and altitude. Oscillographs have been employed to obtain continuous time-history recordings of "VGH" data, for airspeed, normal load factor in g-units, and altitude, respectively. More recently, oscillograph records have been taken of all three load factors n_x, n_y, n_z along the body axes of the aircraft, together with the angular velocities or accelerations about these axes. Most statistics have been taken from VGH data. The continuous recordings have been in the form of visual records - strip charts - and laborious manual operations are required to transcribe the data, or selected items, to punched cards or other media suitable for automatic processing by computing machines.

A special panel of the NASA Subcommittee on Aircraft Loads was set up in 1954 to make recommendations for an expanded statistical program. Main emphasis was placed on maneuver loads, on recording of more aircraft motion parameters, and on the need for automatic data processing to handle the necessarily large amount of data. Subsequently, basic studies were performed by the NASA regarding the choice of loading parameters to be recorded, the statistics to be developed, and an approach to utilizing the statistics in design criteria. The National Bureau of Standards has assisted in one phase of this effort by studying the necessary computing procedures and possible equipment configurations for a single computing facility to serve both the Air Force and the Navy.

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*In the Navy, responsibility for this data reduction has rested with the Naval Air Materiel Center, Philadelphia; in the Air Force, with the Structures Branch of the Aircraft Laboratories, Wright Air Development Center.

The joint USAF-USN-NASA program calls for airborne recorders developing continuous data on at least eight parameters;

q_c	-	Impact pressure
p_h	-	Static pressure
n_x, n_y, n_z	-	Load factors in g-units along the three body axes of the aircraft
p, q, r	-	Angular velocities; roll, pitch and yaw rates.

From these measured data, one can derive the various parameters necessary for calculations of the component loads on major portions of the aircraft structure, such as the horizontal tail assembly. Probability curves for these parameters can be obtained from their time distributions and other statistics resulting from processing the flight data. These statistics can, of course, be used in considerations of the safe service life of the present aircraft type (or of the individual aircraft). However, this indirect statistical approach is of particular value when the probabilities for the various parameters are used with the proper coefficients in load equations specifically for contemplated designs of future aircraft of the same class intended to fly similar missions. It is necessary, of course, to assume that aircraft-with-pilot characteristics will result in a similar history of load factors (3-axis accelerations) and angular velocities.

Primary data are to be obtained from special airborne magnetic tape recorders. Hence, the proposed computing facility, operating directly from these recordings, can provide an enormous increase in the number of flight-hours that can be processed per year, and will deal with enough parameters to provide greatly improved design criteria. There are approximations, of course. However, the recording of the above six motion parameters, instead of n_z only as in the past, will permit computation of probabilities of loads on various components. Furthermore, this complement of data permits correcting the accelerations for displacement of the accelerometers from the c.g. of the aircraft.

The main task of the proposed processing facility will be to obtain the descriptive statistics for the various measured and derived quantities separated according to aircraft-mission category. The <u>evaluation</u> of these statistics, by suitable calculation of probabilities, and application to load equations, so as to establish design criteria, will probably be done by the individual services. These computations are much more sophisticated, but comprise a relatively slight machine load.

DESCRIPTION OF THE REQUIRED PROCESSING

The measured quantities have already been listed as q_c, p_h, n_{xm}, n_{ym}, n_{zm}, p, q, and r. Here we use the second subscript "m" to denote measured, as opposed to "corrected" quantities.

The special airborne recorder was developed by the Emerson Research Laboratories. Very severe requirements were placed upon the design of this unit. The complete unit (less the 3-axis accelerometer for n_x, n_y, n_z) is contained in a package of about 325 cubic inches, weighing about 20 pounds, including transducers and a full reel of magnetic tape to record 50 hours. About half the space is taken up by the pressure transducers, so that the remaining space is scant even for housing the 50-hour reels. Eight parallel recording tracks are employed, and the frequency range extends from about 6 cycles per second down to d-c. A ninth channel provides timing pulses and calibration levels. Accuracy has been specified at $\pm 2.5\%$ of the peak-to-peak range, including playback. The unit starts and stops automatically with successive flights of the aircraft. The small size is required to permit simple installation on a variety of aircraft. Also, to meet a wide range of installation situations, operation at temperatures, ranging up to almost 200°C have been specified; incidentally, the unit contains no "electronics such as tubes or transistors. Vicalloy metal tape is used, to permit meeting the temperature requirement. Direct recording is employed, with magnetization perpendicular to the surface of the tape, which is threaded through a tight slot in the head. Individual reels are not removed from the machine, but the sealed magazine containing both reels and the magnetic head assembly is removed for shipment to the computation center. Capstan drive is not employed. The tape is propelled by the take-up reel, and the speed varies during recording from approximately 0.03 inches per second at the start to about 0.06 ips at the end of the reel. The tape is rewound before playback. During playback, the takeup reel is again driven at constant speed, so that the relative speed between playback and recording remains fixed. A change in the recording on the ninth channel permits recognition of the beginning of each flight. Preceding each flight record, full-scale and zero-level magnetizing currents are recorded on all channels for calibration.

The direct playback unit, also developed by Emerson Research Laboratories, accepts the airborne magazine, and drives the tape at a much higher speed than was used during recording. A speed ratio of 50:1 or 100:1 will probably be used. A flux-sensitive technique is used in reading the recorded signals, using the same heads employed for recording. Signal level is independent of tape speed - readings may be taken with the tape stationary. The playback unit must be stopped for manual adjustments of zero level and gain following each run of data for one flight.

The load factors are to be corrected to refer these measurements to the center-of-gravity of the aircraft. The corrections are kept as small as possible by mounting the 3-axis accelerometer close to the c.g. If its location with respect to the c.g. has coordinates x, y, z, correction equations are:

$$n_x = n_{xm} + \frac{x}{g}(q^2 + r^2) + \frac{y}{g}(\dot{r} - pq) - \frac{z}{g}(\dot{q} + rp)$$

$$n_y = n_{ym} + \frac{y}{g}(r^2 + p^2) + \frac{z}{g}(\dot{p} - qr) - \frac{x}{g}(\dot{r} + pq)$$

$$n_z = n_{zm} + \frac{z}{g}(p^2 + q^2) + \frac{x}{g}(\dot{q} - pr) - \frac{y}{g}(\dot{p} + qr)$$

Derived quantities must be developed before the statistical phase of computations. These are:

$$n_{ze} = \frac{W_i}{W_d} \quad n_z$$

The effective normal load factor, allowing for instantaneous weight W_i differing from the aircraft design weight W_d.

$\dot{p} = dp/dt$

$\dot{q} = dq/dt$

$\dot{r} = dr/dt$

$pq, qr, rp, p^2, q^2, r^2$

ϕ = Roll angle — Obtained by integrating observed roll rate. Integration is started whenever the variable p exceeds its threshold band, and is reset to zero whenever a non-maneuvering status signal is resumed, or when ϕ exceeds plus or minus 360°.

$H_p = f(p_h)$	Pressure altitude
$M = f\left(\frac{q_c}{p_h}\right)$	Mach number
$V_e = f(p_h, q_c)$	Equivalent airspeed. V_T is true airspeed, and "a" is the speed of sound.
$V_T = aM = f(p_h, q_c)$	$a = f(p_h)$
$H_e = pb/2V_T$ Helix angle	b = wingspan

Another quantity expected to be included for statistical treatment is the stagnation temperature $T_s = T_a(1 + 0.2\ M^2)$, for which $T_a = f(p_h)$ is the ambient temperature for a standard atmosphere. However, stagnation temperature was not considered in detail in the present study and will not be discussed further.

Logged Data and Hand Computations comprise essential data required for automatic computation but not originally recorded by the airborne equipment. These are:

Aircraft design weight	W_d
Half wingspan	$b/2$
Aircraft-mission category	Q
Gross weight at takeoff	W_o
Rate of decrease of weight	W_n
Coordinates of accelerometer relative to mean center of gravity of the aircraft	x, y, z

The first two quantities are the same for all data in a magazine. The others must be specified for each flight. The category Q represents the minimum identification for statistical purposes. Additional information may be desired for troubleshooting. The quantities W_o and W_n are chosen to give the best approximation in the simple formula

$$Wi = W_o - W_n t$$

where t is the elapsed time after takeoff. This is a rough approximation, particularly when heavy external stores are dropped during a flight. Maximum errors will range from about 3 to 8%. Some consideration was given to simulation of engine characteristics to allow for weight changes due to expenditure of fuel, to corrections for in-flight refueling, and also to semi-automatic schemes to allow for dropping external stores, but these were given up for the present as being impractical. Likewise, consideration was given to the migration of the aircraft c.g. during flight, and hence variations in the accelerometer coordinates x, y, z. This would be fairly awkward in any event, and entirely impractical for the case of analog computer processing, when the speed of processing is so high that automated changes in coefficients are required for as many as 30 flights per magazine of data. Accordingly, no allowance was made for migration of the c.g. during flight. The correction equations are therefore of little value when the accelerometer is located close to the mean c.g., and the migration is the main source of error.

Elimination of data is to be effected whenever all the quantities n_x, n_y, n_z, p, q, r, and possibly $\dot{p}$, $\dot{q}$, and $\dot{r}$ fall within threshold bands specified for each. It has been estimated that this general threshold condition will be exceeded only about 20% of the time. However, even during the non-maneuvering conditions, time distribution statistics are required for the quantities M, V_e, and H_p, so that these must be derived for statistical computations. In addition, a record of the total elapsed time must be kept.

Output statistics are to be developed for the 18 quantities (19 if stagnation temperature is included) listed in the extreme left column of Table 1. For three of the quantities, V_e, M, H_p, simple time distributions are to be computed for all the originally recorded data, as listed in the top three entries in column 8. For example, tallies are to be made of the accumulated time spent by the variable H_p in each of 10 specified height intervals, such as zero to 5000 feet, 5000 to 10,000 feet, etc. The variable is inspected at intervals of one second of flight time.

For the other quantities, statistics are computed only during maneuvering conditions, but the sampling rate is 5 per second (or 10 per second if this is chosen as the basic sampling rate). The time distributions listed in column 8 are similar to those for H_p. Column 9 calls for a number of distributions, classed in 3 ways according to dwell time spent by the variable within a given

TABLE I REQUIRED OUTPUT STATISTICS

	CORRELATION TABLES (Histograms for Peaks)							TIME DISTRIBUTIONS		ENVELOPES OF PEAKS	
Qty.	(1) 1	(2) x V_e x H_p	(3) x n_{zc}	(4) x n_{zec}	(5) x q	(6) x p	(7) x n_y	(8) 1	(9) x t x M	(10) x V_e x H_p	(11) x p
V_e								20			
H_p								10			
M								20			
n_{xc}	20							20		2x10x5	
n_{yc}		20x10x5	20x5	20x5				20			
n_{zc}		20x10x5						20	10x20x5		
n_{zec}		20x10x5						20	10x20x5		
p		20x10x5	20x5	20x5	20x5		20x5	20			
q		20x10x5	20x5	20x5				20			
r		20x10x5	20x5	20x5		20x5		20			
$\dot{p}$		20x10x5	20x5	20x5				20			2x20
$\dot{q}$		20x10x5	20x5	20x5				20			
$\dot{r}$		20x10x5					20x5	20			
pq	20									2x10x5	
qr	20									2x10x5	
pr	20									2x10x5	
H_e										2x10x5	
ϕ	20										
TOTALS	100	9,000	600	600	100	100	200	250	2,000	500	40
									COMBINED TOTALS		13,490

interval during each excursion into that interval. Ten classes are specified for the primary quantity n_{ze}. The dwell time is specified as one of twenty classes, zero to 1 second, 1 to 2 seconds, etc., with further classification according to five classes of Mach number M. Thus, after each excursion into an interval of n_{ze}, a count is registered in one of 10x20x5 = 1000 tally cells. The records of extreme values (envelopes of peaks) listed in column 11 as 2x20 represent upper and lower extremes (2 classes) observed for the variable $\dot{p}$ for 20 interval classes of another variable p. Column 10 calls for similar tables of 2 extreme values, except for further classification according to five interval classes of a third variable.

The statistics of columns 1 through 7 are histograms of the values of various quantities, noted only at instances when "peaks" occur. The concept of a peak is actually associated with a cycle of fatiguing stress applied to a metal structure such as an aircraft wing root, but is applied here to various parameters. In estimating the computation load we have defined a peak according to a set of criteria developed by the NASA. If the variable S goes through a minimum S_a to a maximum S_{max} and to another minimum S_c, the maximum is considered to be a peak if it meets all of the four requirements:

$$S_{max} - S_a > k_1$$

$$S_{max} - S_c > k_1$$

$$S_{max} - S_a > k_2 S_{max}$$

$$S_{max} - S_c > k_2 S_{max}$$

where k_1 is some fixed magnitude, say 10% of full-scale, and k_2 is a specified fraction, say 25%. The peak is then characterized only by the value S_{max}. The criteria for recognition of a peak imply something about the characterization of peaks, but no further discussion will be attempted here. In column 1 of Table 1, peaks are classified according to 20 intervals of the variable, i.e., a tally is registered in one of twenty cells whenever a peak occurs within that interval. In columns 2 through 7 the peaks must be classified in correlation with secondary variables. Thus, in the fifth line of column 2 we find that peaks in n_y are to be correlated with concurrent values of V_e and H_p, the notation 20x10x5 meaning there are 20 class intervals of n_y, 10 of V_e and 5 of H_p. A peak in n_y might fall in the interval 3.5 to 4.0 g-units, while equivalent airspeed is in the

interval 400 to 500 knots, and pressure altitude 20,000 to 30,000 feet. A count will be tallied in one of 20x10x5 = 1000 memory cells. In terms of digital computer memory requirements, Table 1 calls for 13,490 memory locations for tally purposes.

NATURE OF THE DATA PROCESSING FACILITY

It is desirable that conventional equipment be used if possible. With this approach, the processing is characterized by three phases:

(a) Transcription of data
(b) Editing and preliminary computations
(c) Statistical processing requiring a digital computer

A simplified flow chart is provided in Figure 1.

The rapid transcription of the data from the original magazine to an analog working tape is desired in order to release the magazine for further airborne service. The "quick-look" oscillograph is operated intermittently, providing a strip chart of the eight recorded quantities and time marks (at intervals of 10 seconds of flight time). It is used only for rough visual checks of equipment malfunction and grossly mislabeled data. The transcription process if rapid, and develops a backlog of working magnetic tapes.

In the second phase of processing, the data are converted to the necessary format and recorded for entry to the digital computer. Simultaneously, certain editing processes may be carried out, based on the general-threshold criteria for eliminating most of the required processing, and on criteria for recognition of a rough-air condition in case this has not been achieved by instrumentation on board the aircraft. In addition, the required computations on continuous functions in time - the correction equations for the accelerations, and the derivation of additional parameters - can be achieved with a conventional analog computer during this operation. Since coefficients used in the analog computer must be changed with each new reel of data, and with each flight on the reel, an automated system is desired for these changes. Hence, a perforated control tape is prepared on the basis of manual computations. Some of this information is also required for setting up the digital computer. It may be feasible to enter these coefficients and sufficient identification information through the Digital Tape Preparation Unit 6 as indicated at C. For security of identification, it would seem desirable to enter this digital information directly on the analog transcription tape at point B. However, this would introduce a number of complications and require very closely controlled system engineering, particularly if it is intended to control coefficient changes in an analog computer automatically.

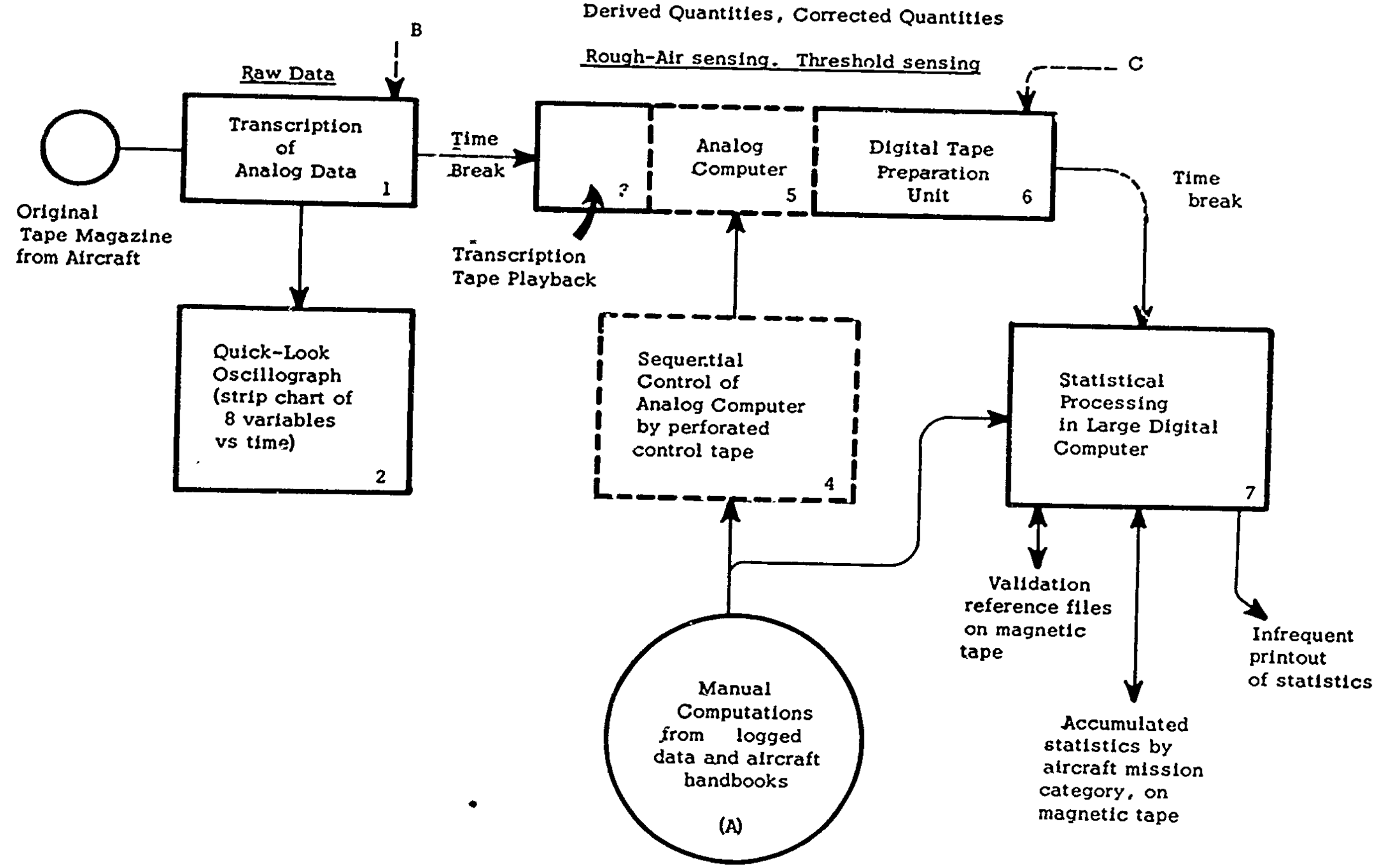

Figure 1 - Simplified Schematic of the Data Processing System

In the third phase of processing, a conventional digital computer of adequate speed and capacity is used for the purely statistical operations, and for preparation of printed reports or digital magnetic tapes for evaluation processing elsewhere. A process of validation is called for after processing data for each individual flight, in an effort to detect mislabeling by determining whether the data are reasonably consistent statistically with other data in the same category or with specified criteria. Valid output statistics for each flight are then merged with the statistical file for that category, together with a record of the elapsed time. These accumulated statistics, nominally 13,490 entries for each category together with identification, are kept on digital magnetic tapes. Each tape could store from 100 to 500 such files, depending on the choice of computer. Approximately 200 categories may be required, and the number could be doubled if rough-air statistics are to be retained. Since the facility is expected to process about 600 flight hours of data per week, and a sample size of several hundred hours of data will be required for each category* the call for printouts of final statistics would be infrequent - on the average. However, completion schedules may tend to group, and monthly reports will probably be desired, so that a fairly high speed line printer is needed.

The off-line rather than direct operation of the digital computer as indicated by the time break after unit 6 seems desirable since the units 3, 5, and 6 will operate at a steady rate, whereas the digital computation load - and hence the required speed of operation - will fluctuate with changes in the character of the recorded data. Thus, the digital computer is required to keep ahead of the preliminary processing equipment only on a long-term average basis.

The possibilities of using highly specialized computing components will be discussed later.

AN ALL-DIGITAL SYSTEM USING A LARGE GENERAL-PURPOSE COMPUTER

This is the simplest approach from the viewpoint of equipment procurement and was the first to be considered. It provides the greatest flexibility and is simplest to specify. Due to the large computation load, a very fast machine is required, with a rather large high speed internal memory - though not notably large by today's standards.

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*The Naval Air Development Center (NADC) has the responsibility for providing criteria for judging sufficiency of sample size.

Our first estimates indicated that the processing load would tax the capabilities of a Univac 1103A or IBM 709 computer, so that two-shift operation would be needed. It seemed inadvisable to plan for three-shift operation, since full value could not be obtained from the third shift, and no margin of performance would remain. We have assumed the use of modern reliable equipment and a schedule allowing at least one hour a day preventive maintenance and one full day per month. Nonetheless, an allowance was made for 25% loss of schedule due to the combined effects of the following factors:

(a) Unscheduled maintenance - breakdown of equipment
(b) False starts - from faulty setups
(c) Repeated runs - for checking or identification
(d) Routine loss of time due to hand operations

The loss will be much greater before fully routine operation is established. All this means about 3000 hours of available machine time on a two-shift, five-day work week basis. Thus, if 60,000 flight hours are to be processed each year, the machines, while in bona fide operation, must process data in 3,000/60,000 = 1/20 flight time represented by the data. If the computer is capable of a speed ratio of 20:1 it can process 30,000 flight hours operating on a single shift.

The equipment required can be seen in Figure 2, if we ignore the units numbered 23, 24, 25, and 27. The magazine of tape from the airborne recorder is played back at 100:1 speed ratio in the playback unit 3, the process taking 1/2 hour for a 50-hour magazine exclusive of time taken for calibration between flights. The analog transcription is by means of conventional FM recording technique at 15 inches per second tape speed. There is appreciable degradation in accuracy of the data during this process, but the errors are considered small compared to those previously introduced.

The transcription, or working, tape can be played back subsequently in unit 6 at a standard tape speed of 3-3/4 inches per second corresponding to a 25:1 speed ratio. The Digital Tape Preparation Unit (DTPU) accepts these data continuously, sampling all eight channels of analog information every 1/125 second, corresponding to 1/5 second of flight time. Each sample is called a frame of data. It may also contain a repetitive pattern of digital data defining the category and flight number, changed by hand for each flight (9) or in an automatic sequence from stored data in unit 10. The DTPU includes a fast analog-digital converter, a multipole analog switch, fairly large buffer storage, and control circuits to permit recording the digital data in "blocks" or "records" (which may contain several standard machine blocks) separated by gaps as required by the computer. A more detailed block diagram of the DTPU is given

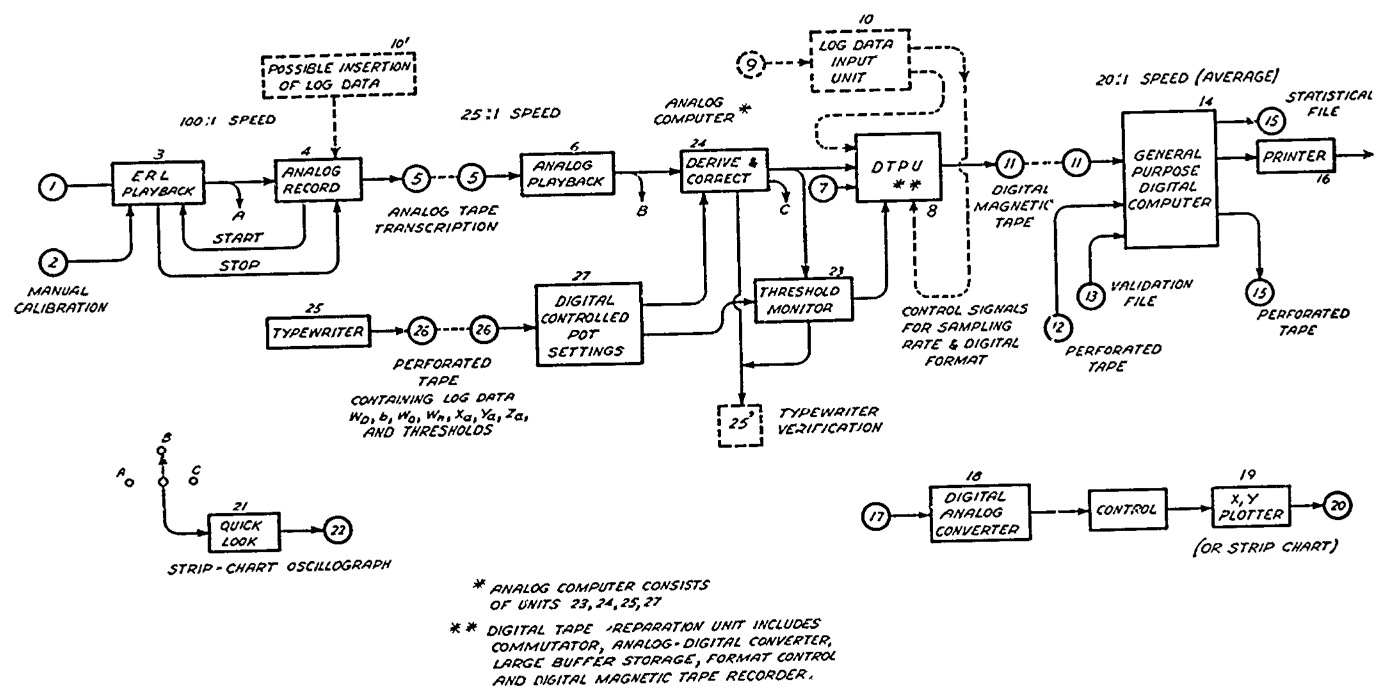

Figure 2 - The Analog-Aided Hybrid System

in Figure 6. The associated tape drive may be stopped and started rapidly to produce these gaps, or a continuous running technique may be employed. The standard input data record consists of about 1000 to 2000 6-binary-bit characters, depending on the choice of computer. The length of the working Input Record is determined by the duration of flight time (about 20 seconds) required for each scan in the search for peaks. The duration selected is not critical, since a data and status storage subroutine is employed for partially confirmed peaks.

The specified sampling rate of 5 per second of real time is subject to change. It is about the minimum permissible value, limiting the useful spectrum to about one cycle per second. If the sampling rate is doubled, however, the processing capability of the digital computer is effectively halved. It is to be noted that the analog signals must pass through a filter to exclude frequency components more than half the sampling frequency, f_o, before the sampling process takes place. This is to prevent "aliasing", or the generation of spurious components at less than 1/2 f_o in the sampled (digital) data.[2]

The accuracy required in the analog-digital conversion is not great. Seven binary digits (1 part in 256) plus sign would suffice, and 9 bits plus sign have been specified. Actually, two standard characters of 6 digits each have been assigned for each quantity expressed in digital form. This leaves two bits for tagging the data as "rough-air", if desired. For a variety of reasons, 12-bit binary representation has been adopted for representing practically all quantities manipulated within the digital computer, and stored on its output and auxiliary tapes (for "validation" and "statistical files").

The digitized data (11) from one magazine will consist of 2 to 7 reels of computer tape, depending on the computer. A threshold sensing and editing circuit operating ahead of the digital converter might be expected to reduce the required tape space by 80% and notably increase the effective speed of the computer. Actually, it does not appear to be feasible to save much tape space in this manner. Furthermore, since it turns out that the computing speed is limited by internal operations in any event, and not by the tape data transfer rate, there is little gain in average computing speed - the threshold-sensing task being a trivial one for the digital computer.

The computer input tape (11) is placed on one of the two input tape drives of the computer. When the computer is available, all necessary auxiliary data and identification not included on the working tape by some special process (10) can be inserted by conventional means such as the perforated tape reader (12).

An auxiliary plotting unit is shown at (19). If only envelopes are to be plotted, as points, the required speed and accuracy are quite low, and a simple, though specialized, arrangement of a stripchart recorder operating from punched tape (or cards) prepared from the computer might suffice. However, a precision x, y plotter and control system to produce straight line segments may be needed.

The digital computer required is in a class including the IBM 709 and 7090, Univac M-460, Philco S-2000, Control Data 1604, and Datamatic 800. These differ widely in speed. However, all are binary machines, have large random access magnetic core memories of high speed, and provide interlaced (or "simultaneous") input and output operations using fast magnetic tape drives and other devices while computing. Detailed estimates of requirements have been made, but will not be discussed here. Eight high speed tape drives are indicated, with multiplexing for simultaneous input and output on any two of them. The memory size mainly depends on the number of entries in the Statistical Record (i.e., the 13,490 entries of Table 1), the number of binary digits required for each entry, and decisions whether data need to be stored in the high speed memory or on tapes. It was considered that a nominal memory size of 8,000 words would meet present estimates if the computer word length is 48 bits with two instructions per word, or 12,000 words, if 36-bit with one instruction per word. It is, of course, advisable for the memory to be larger to meet contingencies.

VARIATIONS OF THE ALL-DIGITAL SYSTEM

The large general-purpose digital computer will have a variety of input registers, and also "interrupt" programming features, so that it can automatically share its time in several assigned computation tasks. Its nature is such that it can hold a field of digital information, rearrange it, and transfer it at any desired time sequence. Provided the machine is kept busy, it does this very economically. Thus, depending on rather detailed considerations, it may be feasible to use the computer itself to accomplish the specialized buffering and format-control operations required in the DTPU. A direct connection ("on-line") is established between the analog converter (and multipole analog sampling switch) and a high speed register in the computer, which then accepts data at a steady rate (i.e., a fixed number of frames per second), stores them in a portion of its high speed memory, and reads out on magnetic tape in spurts - providing the necessary gaps in recording a working magnetic tape. This process can be carried on concurrently with statistical computations using a digital tape previously prepared by the computer itself. The tape preparation function would have priority. The advantages listed for off-line operation would be retained. We can designate this arrangement as "computer-buffer tape preparation".

As mentioned in the last section, an external threshold monitoring unit can be used to reduce the input data rate - nominally by 80% - during non-maneuvering flight conditions. During such periods the sampling rate would be cut to an equivalent one-per-second flight time, and only M, V_e, H_p, are required. Study shows that negligible reduction in processing time would be achieved, and that the reduction in digital tape requirements would be much less than 5:1. It is not expected that these input digital tapes will be kept for archive purposes, ordinarily. The analog working tapes will be kept for a period of time, perhaps 6 months.

Thoght was given to a special time-distribution statistical computing unit operating only on M, V_e, and H_p, so that these quantities would not be sent to the digital computer at all. This scheme would really save tapes, but appeared impractical after extensive consideration.

ALTERNATIVE USE OF HIGHLY SPECIALIZED COMPUTING COMPONENTS

Sinde the program of statistical data reduction will be uniform for all Navy and Air Force aircraft, and must persist without much modification for a long period - two years or more - to attain adequate sample sizes, it seemed possible that an assemblage of specialized components, worthless for general computing applications, might be desirable. The increase in machine efficiency, or in sheer processing speed, might offset inflexibility and the inconvenience of the special engineering development required.

Special designs were considered for all the basic routine processes; simple time distributions, dwell-time distributions, and peak recognition units. It is possible to devise a scheme in which an assemblage of such units does a major share of the computation load assigned to a general-purpose digital computer in the flow chart of Figure 1. In principle, this can permit use of a far slower digital computer, of the type using a drum memory, since the computer deals with a relatively small input of partially classified data, and is primarily engaged in replacing items or rearranging these for printout. A conventional analog computer would be included for deriving and correcting quantities.

For example, analog differentiating and holding circuits can be employed for detecting peaks and valleys, and noting when the logical criteria for a true "peak" have been fulfilled.

A scheme using the most direct logic for recognition of peaks is shown in Figure 3. No differentiating circuits are employed. The input variable S goes through a buffer amplifier M, and thence to a group of voltage comparators

VC, having all their inputs in parallel. Each comparator in the series S_{n-1}, S_n, etc., is set to operate at a higher signal voltage than the one below. The coincidence gates A_1 and A_2 produce a positive output if, and only if, all their inputs are positive. When S is at a voltage below S_{n-1}, the three inputs to A_2 (reading upward) are + + -, since FF has previously been "off". If then, S rises to a maximum $S_n < S < S_{n+1}$ and descends, the inputs to A_2 are

- + - when $S_{n-1} < S < S_n$

- - + when $S > S_n$

- + + when $S_{n-1} < S < S_n$

+ + + when $S < S_{n-1}$

changing to (+ + -) as the
output of A_2 resets FF

Note that the flip flop FF is set when S first exceeds S_n..This whole cycle of events results in counting a peak of class S_n. The connection from the next higher voltage comparator into 0_1 makes it impossible to get an S_n count in case the variable went on to an S_{n+1} or higher class of peak. Figure 4 gives some indication of how interconnections are made to include the criterion that the excursion should be greater than 25% of the value at the peak.

By use of gating circuits, the output counts could be classified according to concurrent values of other variables as needed for correlation purposes, as listed in Table 1. This scheme becomes excessively complicated when such correlation is required, since the peak is not confirmed until the necessary minimum is reached, and this may be many seconds of flight time after the peak.

In general, a complete specialized processing system made up of an intricate arrangement of such devices does not appear to be advantageous, at least for the main task at hand.

ROLE OF A CONVENTIONAL ANALOG COMPUTER

The solution of correction equations and computation of various derived quantities can be performed by a conventional analog computing facility. The inputs and outputs are continuous functions of time. By nature, the analog computer is more efficient in such a case, since the digital computer

operates with whole quantities instead of increments. Therefore, high speed may be possible at low cost. Furthermore, the accuracy requirements are not high and are easily met with well-known components. On the debit side, the analog computer has a quite limited range of application, cannot be multiplexed in operation like a digital computer, and is awkward when a great many changes in coefficients or modes of operation are required during a single problem run, particularly if continuous processing at very high speed is required.

Our first estimates of the computing load imposed on an all-digital system indicated that an IBM 709 computer - the most powerful of the standard machines then existing - would be quite incapable of processing 30,000 flight hours per year, even operating two shifts per day. These estimates were based on a basic sampling rate of 10 per second in real (flight) time. With the 5 per second sampling rate, the computation load is effectively halved. This estimate proved to be considerably in error; an over-estimation of the computation load. However, it appeared that the addition of an analog computer might be advantageous to

(a) increase processing speed by about 2:1
(b) reduce cost by performing about half the task with equipment costing much less than half the digital computer cost.

The analog computer is fitted into the system as shown in Figure 2, where it is represented by units 23, 24, 25, 27. As a result, input to the DTPU consists of 19 channels of analog information (including stagnation temperature T_s) instead of 8. The standard Input Record for input to the computer (comprising about 20 seconds of flight time) therefore requires more space on the digital tape, and it is pretty definitely advantageous to employ an analog Threshold Monitor unit, number 23, which acts on the DTPU to cut the sampling rate by a factor of five during non-maneuvering flight conditions.

Figure 5 shows the overall arrangement of the analog system alone, with the input and output quantities. Two banks of servo-set potentiometers are shown, one bank being connected in the computing circuits while the other is being set up. An electric typewriter 25', types out a verification of the settings actually made in the analog computer in response to the punched paper control tape 26.

For a slower average processing rate, time could be taken between "flights" to position the pots; or it could be done by hand, since most of the potentiometers would be ganged. In the Threshold Monitor Unit, instead of arbitrary settings of servo pots, it is likely that a simple switch

between two sets of values will be sufficient - putting all aircraft in two broad classes.

The bandwidth requirements are fairly severe for some of the analog computing components, such as multipliers. Moderately accurate performance is required at frequencies corresponding to 5 cycles per second referred to flight time. For a speed ratio of 25:1, required to process 60,000 flight hours per year (assuming a 16-hour work day) the corresponding frequency is 125 cps. This can be achieved with standard components; to go higher would require reconsideration.

The digital computer, and complement of peripheral digital equipment, would have the same specifications as for the all-digital system, but the computation speed required would be about half as great.

The Digital Tape Preparation Unit (DTPU) would be somewhat more complex. However, for any large data processing facility, even though its reason for existence is a specialized task, some flexibility in the input equipment should be provided. Hence, the DTPU should be capable of operating from a lesser or greater number of analog channels, changing the sampling rate, and adjusting output format to effectively fill the digital tape despite these variations. Figure 6 indicates the general arrangement of the DTPU. There are several system design approaches to the objective, of course, and the actual system will vary with the choice of equipment supplier. A speed ratio of 12.5:1 is mentioned in the drawing, but 25:1 seems to be an optimum figure at present.

In particular, to cover all contingencies, the DTPU is specified to be capable of accepting the original 8-channel data, so that all computations can be performed by the digital computer if desired.

PROCESSING RATES ACHIEVABLE WITH DIGITAL COMPUTERS

To estimate the capabilities of available general-purpose computers, fairly detailed programs for computation were prepared, but not in such detail as to be coded for any specific machine. It was assumed that the computer would be of the single-address type, as represented by the IBM 709 and 7090, the Philco S-2000, Control Data Corporation 1604, and Univac M-460.

Figure 7 indicates roughly the sequence of digital routines during statistical phase of computations only, and for above-threshold conditions. The cycle beginning with data input is repeated with each new input load

of about 20 seconds of flight data. In each cycle, the various time distributions are updated, for each frame of data, then a search is made for peaks in various parameters, and the peaks are classified according to concurrent values of specified parameters. The envelopes (tables of extreme values) for these peaks need not be developed until the end of each flight, before the validation and merge operations. However, envelopes for helix angle, $pb/2V_T$, must be updated for every frame, since peaks are not determined. Not shown is a conditional data-storage subroutine, which permits use of a shorter scan in flight time (e.g., 5 seconds) without loss of slowly developing peaks that require more than this interval for recognition.

To process each input load of data for 20 seconds of flight time requires for average data 370 "long" operations such as division, 97, 800 "short" operations such as addition (with associated memory reference for the operand), and 88, 100 memory references for instructions. As mentioned, these numbers are not optimized for any one machine. Now, for illustration, consider a hypothetical computer having a performance representative of those listed. Assume that 100 microseconds are required for a long operation, and 10 microseconds for a memory reference, or 5 microseconds per instruction if two are contained in each computer word. Assume that it takes 12 microseconds to pick up an operand from the memory and execute a short operation. Such a machine would take 1.65 seconds to process 20 seconds of flight time. Since about 80% of the flight data will be sub-threshold, the hypothetical computer would easily exceed the average speed ratio of 20:1 required to process 30,000 flight hours per year, operating on an 8-hour work day. No allowance is made here for magnetic tape operations, or for merging, validation, and the occasional printouts. However, these operations contribute very little to the average processing load. In this example, it is assumed that the preliminary computations, comprising about half the task, are performed by an analog computer.

DEVELOPMENTS THAT COULD AID THIS PROGRAM

The U. S. Air Force and Navy are proceeding with the establishment of this computing facility and the necessary engineering to combine a fast general-purpose digital computer and a conventional analog computer in an adequate working system. Certain advances in equipment would be desirable, particularly if a continuing program calls for increases in the amount and variety of flight data.

The compact airborne recorder is basic to the program, and continuing improvements in recording accuracy and speed stability are worth seeking.

The reliability of the processing would be increased if means could be provided for direct recording of identification and working coefficients on the original tape. Some refinements in processing might be incorporated if markers could be recorded to indicate the occurrence of certain events such as refueling, dropping of stores, or critical maneuvers. A reliable airborne instrument for detecting rough-air conditions is desired. Alternatively, criteria are sought for a computing scheme to identify those portions of the data which have been recorded during rough air conditions.

The state of the art seems adequate for achieving with available equipment the present objectives for the computing facility. However, a better margin of recording accuracy is desirable in the analog tape transcription, which uses FM recording on magnetic tape. As in other installations of this kind, data preparation for the digital computer is a fairly complicated process. The maximum in flexibility is desired, without complications in equipment. For processing more than about 60,000 flight hours per year, the conventional analog computer included in the system of Figure 2 may limit the speed unless unusually fast nonlinear components are employed.

Lastly, it may be worthwhile to reconsider the use of specialized computing devices to augment this program. These will tend to be advantageous whenever large quantities of data are to be processed according to a fixed formulation.

ACKNOWLEDGEMENTS

The author wishes to acknowledge the work of his associates in this study, Mr. E. S. Sherrard and Mr. D. C. Friedman, and the guidance provided by Mr. Herman Levy, of the Naval Air Material Center, Philadelphia.

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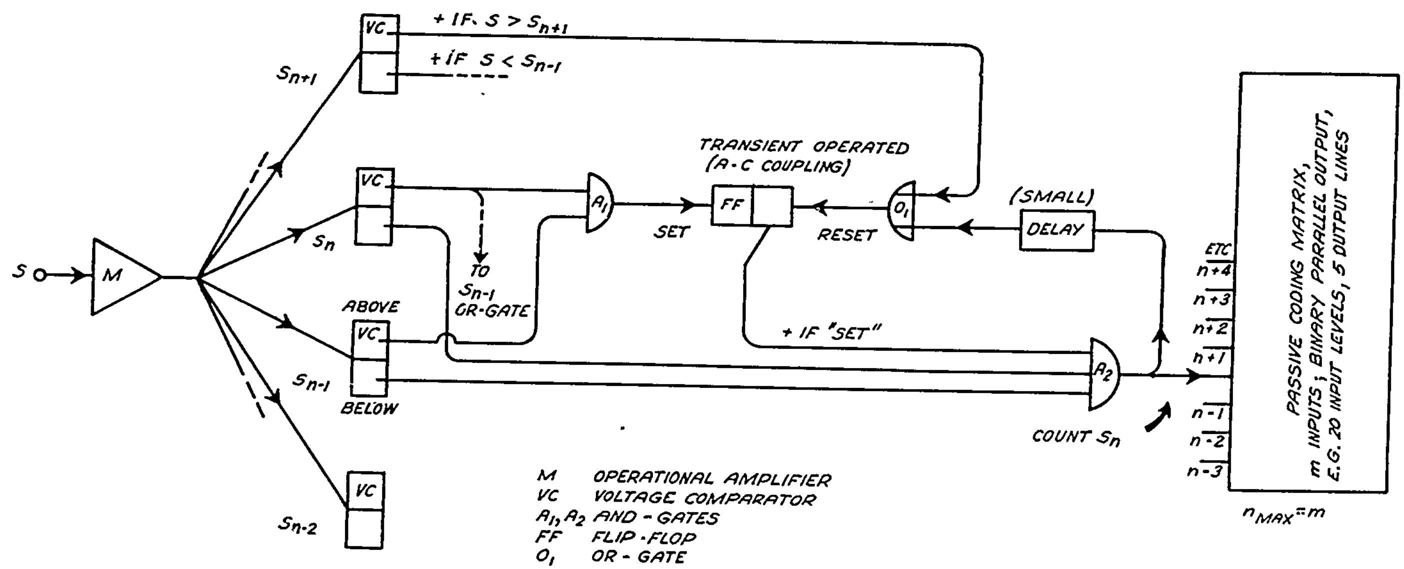
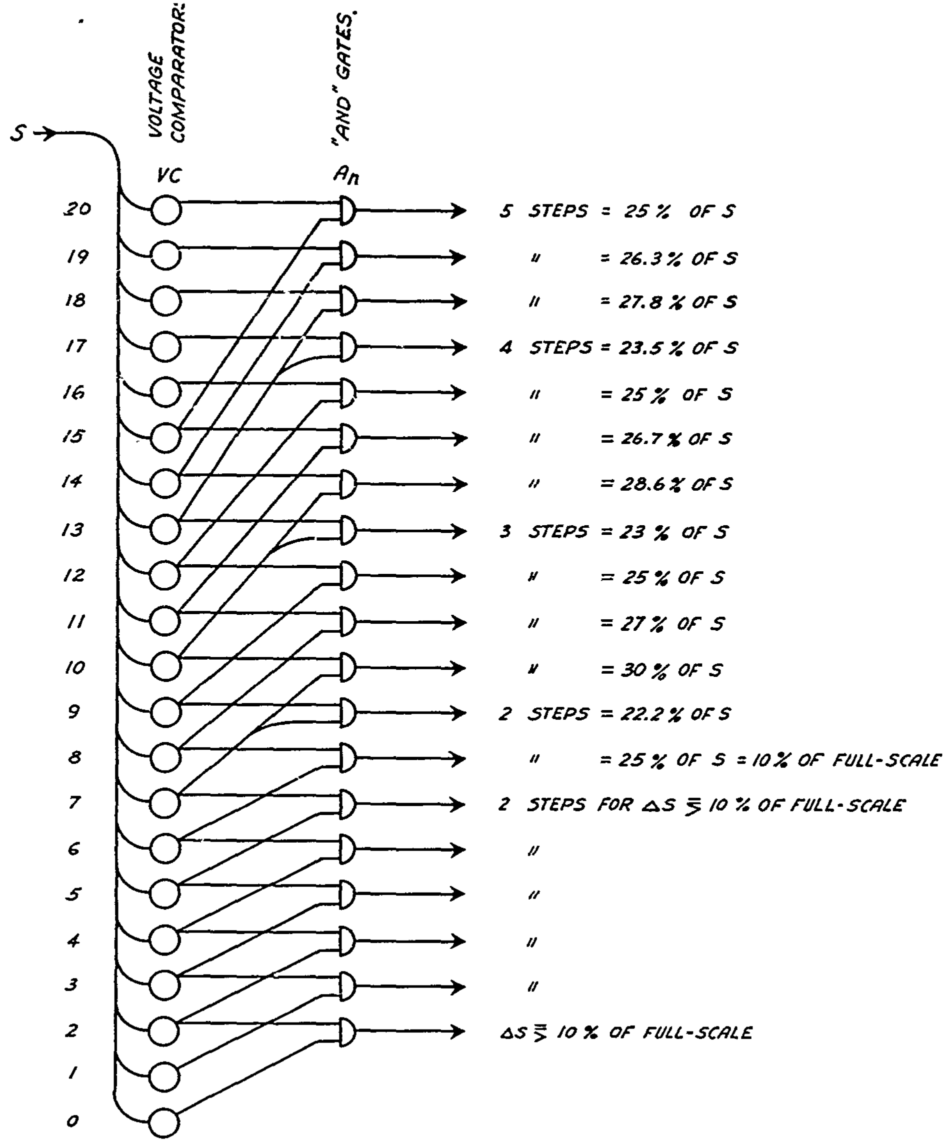


Figure 3 - Special Peak Recognition Module



CRITERIA

$|\Delta S|$ MUST EXCEED 25% OF $|S|$ AND MUST EXCEED 10% OF FULL-SCALE VALUE OF S

NOTES:

FOR THIS EXAMPLE IT IS ASSUMED THAT "S" IS POSITIVE.

THE VOLTAGE COMPARATORS ARE SET FOR 20 EQUAL STEPS IN THE VARIABLE "S".

THE PERCENTAGES INDICATE APPROXIMATIONS OF THE 25% CRITERION.

OUTPUTS SHOWN FROM A_n ETC. ACTUALLY CORRESPOND TO A_2 OF FIGURE 3. INHIBIT LEADS ARE NOT SHOWN, NOR THE CODING AND READOUT CIRCUITS.

Figure 4 - Interconnections of Modules

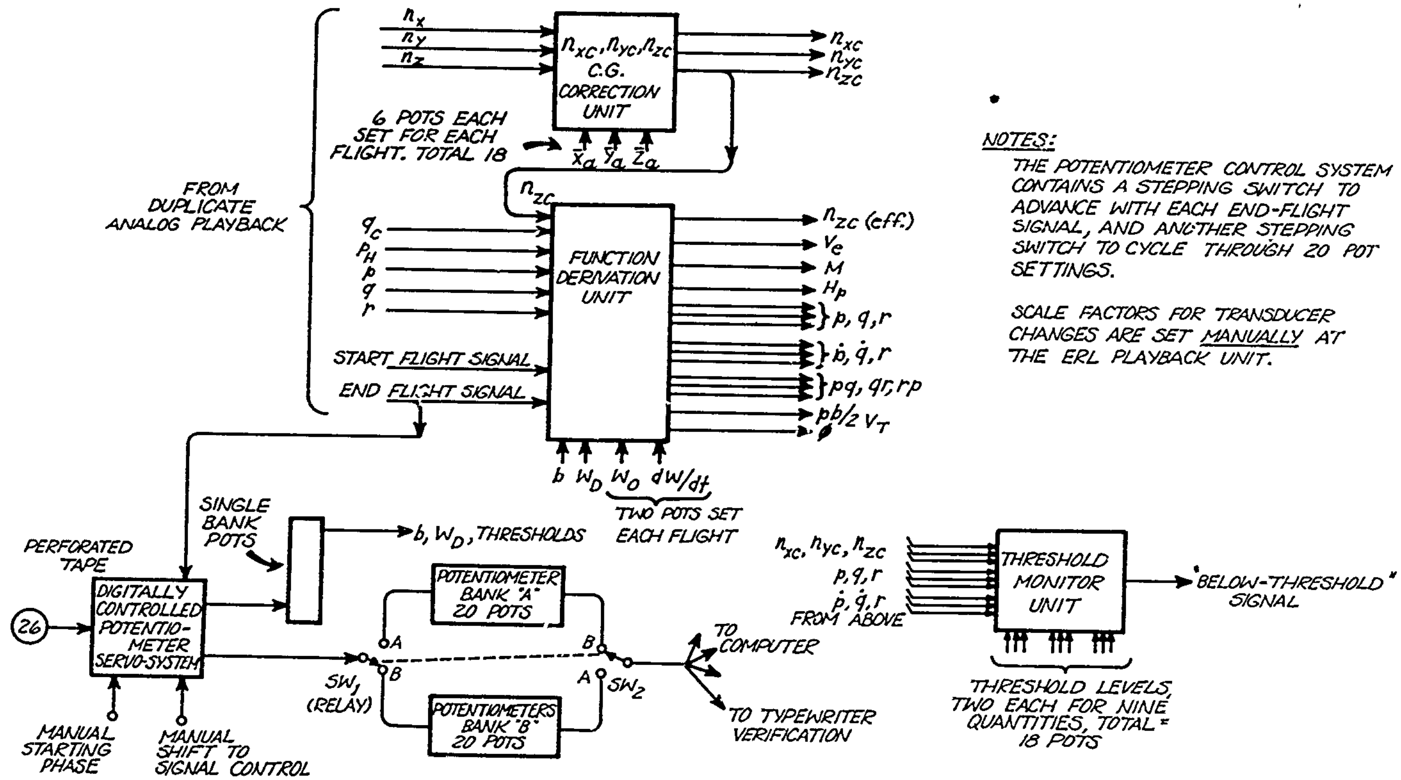


Figure 5 - Analog Computer

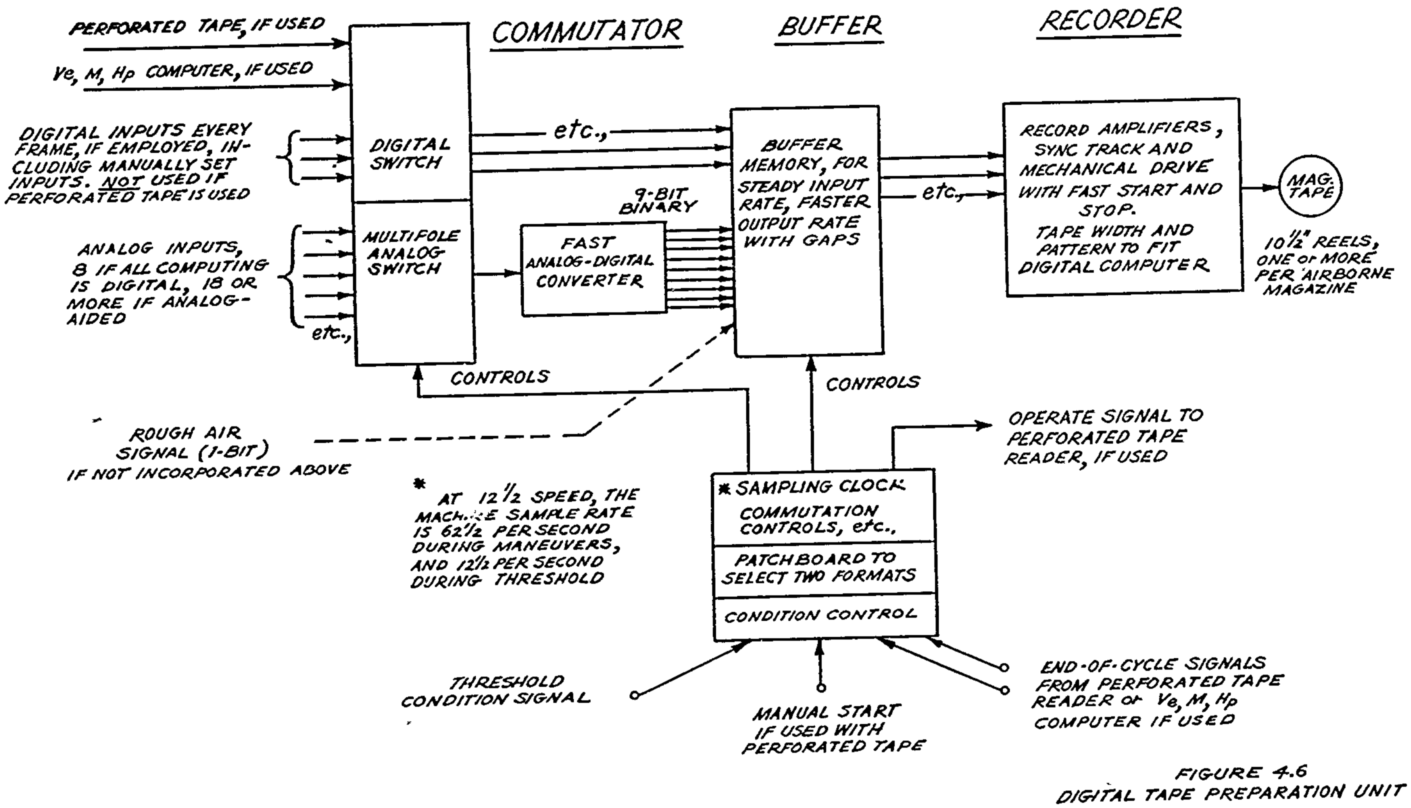


Figure 6 - Digital Tape Preparation Unit

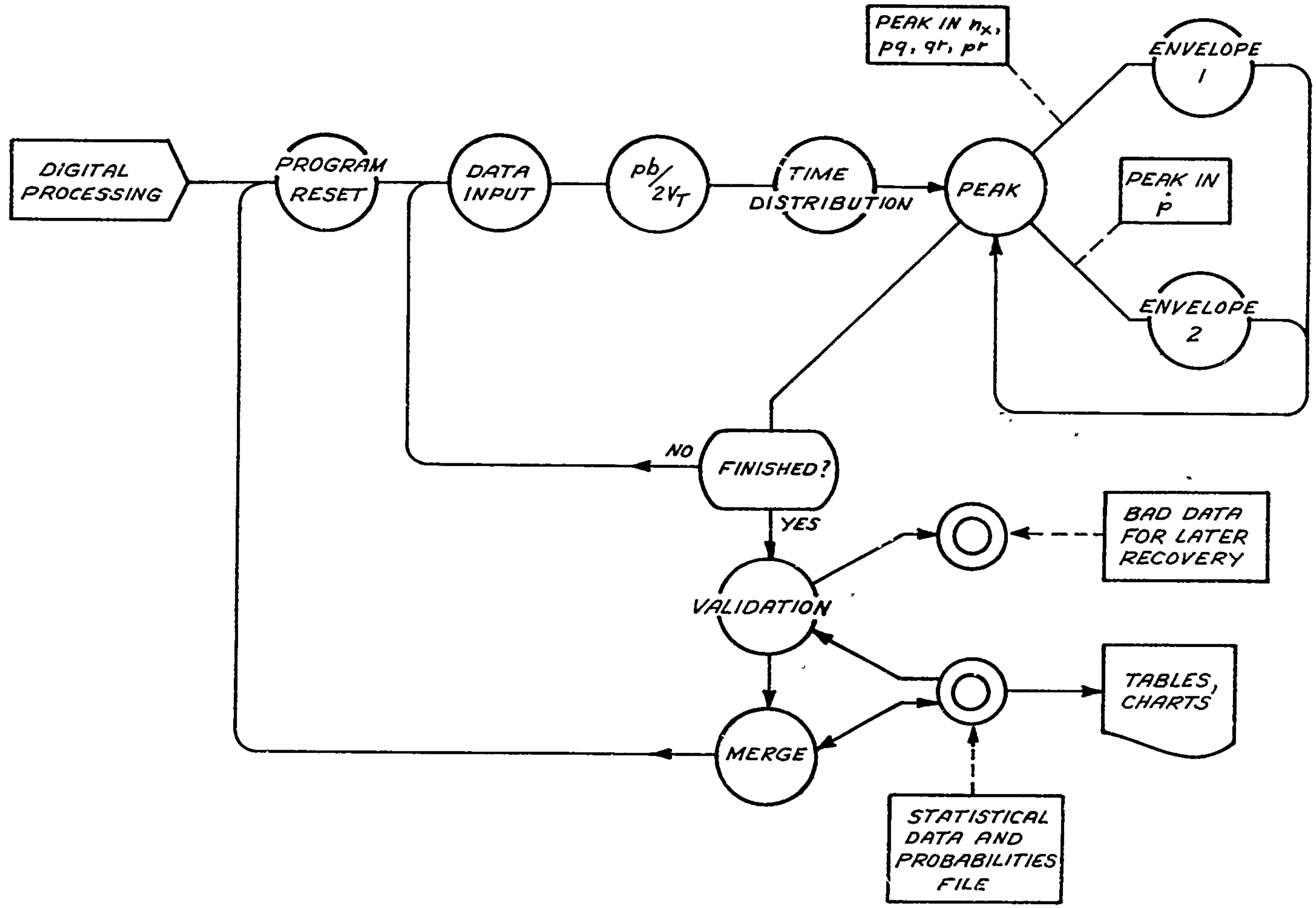


Figure 7 - Overall Flow Chart for Computing Descriptive Statistics

A 2-DIMENSIONAL
ACCELERATION PEAK FREQUENCY MODEL
FOR
PREDICTING FLIGHT ENVELOPES

By

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ABSTRACT

This paper describes a procedure for developing an analytical expression for rate of occurrence of acceleration peaks and constructing prediction envelopes.

The formal aspects of the problem of fitting bivariate frequency data for prediction of extreme maneuver loads, are described in Section I. Section II contains the history of our efforts to develop a solution to the problem and a brief outline of related efforts by other workers. A procedure for constructing prediction envelopes which meets the requirements of flexibility, simplicity, and general applicability, desired of the solution, is described and illustrated in Section III.

LIST OF SYMBOLS

H	total number of flight hours on the records
E	number of hours for which prediction is made
V	airspeed
g	acceleration
Λ_E	probability of a peak exceeding g over any 20 mph airspeed interval
$L_E(V, g)$	flight envelope for E hours, $P_E(x>g, V) = L_E(V, g)$
$\hat{P}(x>g\|V)$	relative frequency with which g is exceeded in airspeed band, $V \pm 10$
$F(x>g, V)$	joint frequency distribution of acceleration and airspeed
N	total number of positive peaks observed in H hours of flight
$\sqrt{\beta}_1$	standardized third moment
β_2	standardized fourth moment
$B(a, b)$	Beta function
$I_y(a, b)$	Incomplete Beta function

SECTION I

INTRODUCTION

The problem is to find an analytical expression for any model of aircraft which gives the rate of occurrence of acceleration peaks observed in H hours with a view to predicting rates to be expected in E hours, $E > H$. The data on which these predictions are based are in the form of frequency tabulations of acceleration peaks cross-classified by airspeed and sign and magnitude of acceleration, Figure 1. Imposed on the theoretical diagram in Figure 2 is a schematic representation of the V-g frequency tabulation of Figure 1.

The predictions for E hours are to be made in the form of theoretical envelopes, also shown in Figure 2. The expected rate of occurrence of peaks in the area above the envelope is 1 per 20 mph per E hours of flight. Similarly, 1 negative peak per 20 mph is expected below the envelope in E hours. These predicted values are as likely to occur at one airspeed as another.

This work was done in connection with the flight loads study program and was concerned specifically with maneuver loads. The data used in illustration were taken from V-g tabulations of acceleration peaks read on V-g-h charts from the Hathaway recorders installed in F86F jet fighters at Nellis AFB. Peaks greater than 2.0 g were read provided the acceleration trace increased by 1.0 g and decreased by 0.75 g or vice versa. Negative peaks were read when less than or equal to 0.5 g and separated from adjoining negative peaks by at least a plus and minus 0.5 g variation in trace deflection. Whenever the acceleration trace is read, the airspeed and altitude traces are also read.

The envelope for E hours can be formally defined. Suppose the incidence of acceleration peaks by magnitude and airspeed be governed by a true probability law P(xy) for positive (or negative) peaks. Then the positive portion of the envelope for E hours should be determined so that for any 20 mph airspeed interval with center at V

$$\int_{V-10}^{V+10} \int_{g}^{g_{max}} P(xy)\, dx\, dy = \Lambda_E \qquad (1)$$

where Λ_E is the probability of a peak exceeding g over any 20 mph airspeed interval and is equal to the reciprocal of the total number of positive peaks expected in the number of hours E for which the prediction is made. The probability of exceeding g must be the same for all airspeeds so that if the function P(xy) were known (including its upper acceleration limit g_{max}), the envelope could be found by solving the integral equation (1) for its lower limit g in terms of the covariate airspeed, giving the envelope $L_E(V, g)$.

F-86F NELLIS AFB
MANEUVER LOADS
COMPOSITE OF TRAINING OPERATIONS

FLT. TIME 909.2 HRS.
NO. FLTS. 1117

ACCELERATION — g

V_E — M.P.H.	-1.7	-1.2	-0.7	-0.2	0.2	0.7	1.2	1.7	2.2	2.7	3.2	3.7	4.2	4.7	5.2	5.7	6.2	6.7	7.2	7.7	8.2	8.7	TOTAL
110					7																		7
130					9																		9
150					26																		26
170					26																		26
190					55				30														85
210				4	72				164	41													281
230				8	136				304	78	15	1											542
250				8	215				786	426	97	14											1546
270				10	194				944	686	261	52	10										2157
290				7	243				1067	954	527	170	27	8									3003
310			2	16	258				1091	1096	684	293	67	24	1								3532
330			1	17	233				968	1149	857	488	157	52	11	4							3937
350			4	17	209				853	1094	993	685	299	117	32	9	3						4315
370	1	1	5	15	237				842	1053	1194	926	515	231	117	30	3	1					5171
390			2	8	210				615	845	1024	996	673	377	211	65	28	5					5059
410	1		1	15	164				446	522	621	648	551	319	198	92	45	27	2				3652
430				12	135				306	277	361	376	320	232	155	97	50	22	5	3	1		2352
450				12	97				181	197	211	237	230	209	140	106	58	30	12	4	1	1	1726
470			1	3	78				132	122	153	194	160	170	172	150	100	43	22	6			1506
490			2	2	33				72	86	77	108	84	98	106	75	50	17	8	6			824
510				1	24				32	40	48	40	40	38	36	31	16	4	2	1			353
530			1		14				22	13	13	18	13	14	13	10	3	2		1			137
550					5				9	9	6	5	4	3	5	3	1	2					52
570					2				5	4	3	5	3	1		1	2						26
590					2				2	2	3	2	2										13
610									2	1	2	1	1										7
630											1												1
650																							
670																							
690																							
TOTAL	2	1	19	155	2684				8873	8695	7151	5259	3156	1893	1197	673	359	153	51	21	2	1	40345

Figure 1.

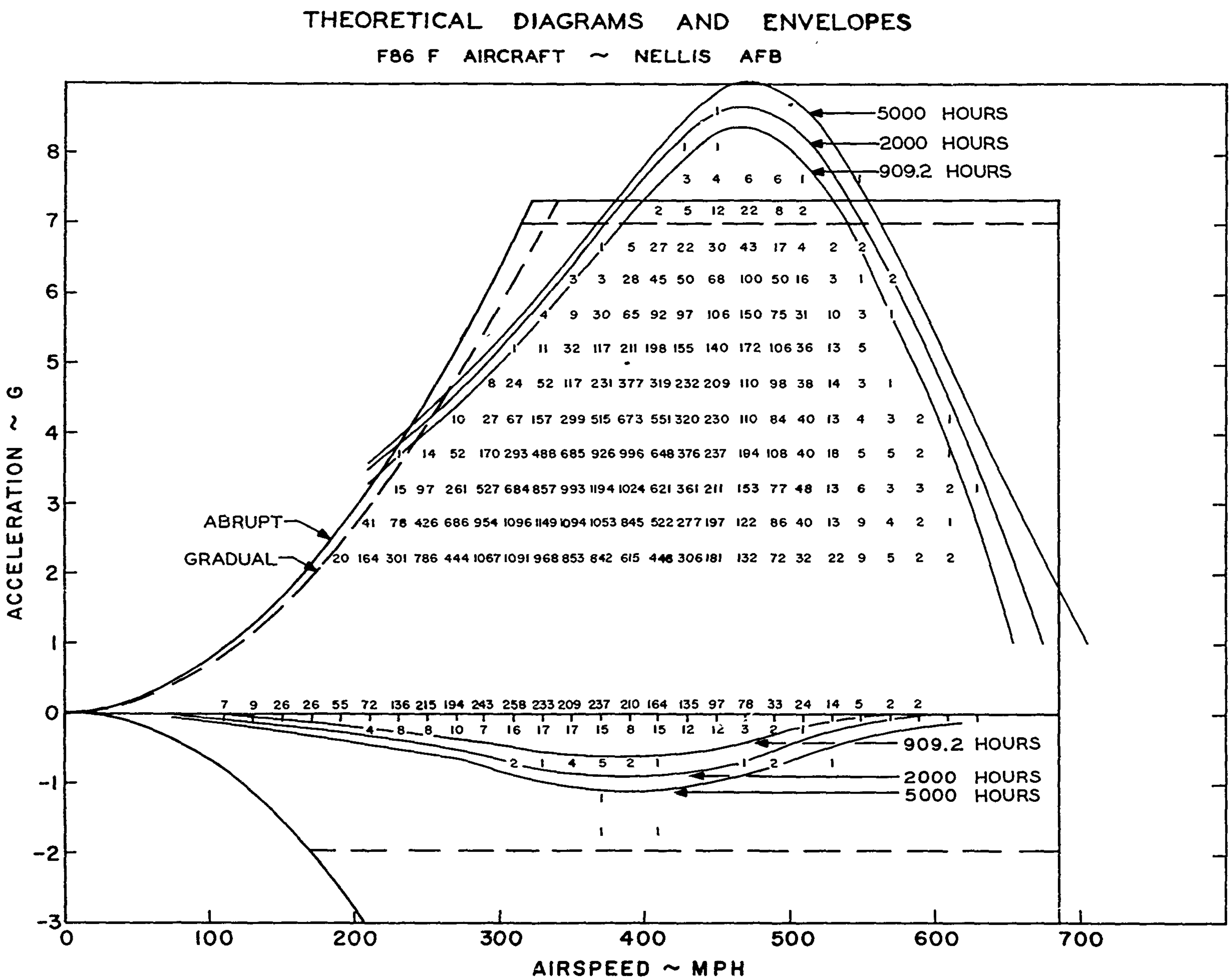

THEORETICAL DIAGRAMS AND ENVELOPES
F86 F AIRCRAFT ~ NELLIS AFB
5000 HOURS
2000 HOURS
909.2 HOURS
ABRUPT
GRADUAL
909.2 HOURS
2000 HOURS
5000 HOURS
ACCELERATION ~ G
AIRSPEED ~ MPH

Figure 2.

Working with experimental data which are grouped in class intervals, the integrals are replaced by summations and the probability law P(xy) must be estimated. The general probability function P(xy) may be written P(x|y) P(y) where

$$P(y) = \int_{\text{Range of } g} P(xy)dx$$

and P(x|y) is the conditional acceleration distribution at airspeed y. For clarity, we give a little more fairly obvious notation here. Since our data are grouped, we are always working with probabilities for intervals so the relative frequency $\hat{P}(V)$ of peaks in the airspeed band V±10 is an estimate of the integral of P(y) over that band

$$\hat{P}(V) \rightarrow \int_{V-10}^{V+10} P(y)\,dy.$$

The relative frequency $\hat{P}(x > g\,|\,V)$ with which acceleration magnitude g is exceeded in airspeed band V±10 is an estimate of the integral of the conditional probability in V±10:

$$P(x > g\,|\,V) = \int_{g}^{g_{max}} \frac{1}{P(V)} \left[\int_{V-10}^{V+10} P(xy)\,dy \right] dx.$$

Customarily, positive and negative accelerations are treated separately. Previous study has shown that a specific magnitude of acceleration does not occur with equal probability at different airspeeds. The "true" probability law P(xy) governing rate of occurrence of acceleration peaks is a model useful in ascertaining the smooth pattern which we suppose to underlie the somewhat erratic distribution of frequencies in a V-g tabulation.

However, maneuver loads are not essentially random phenomena. The severity of the maneuver is pretty much under the pilot's control and study of our V-g tabulations suggests that maneuvers of a given type tend to be performed within a certain airspeed range and to have their own central tendency. When data from a number of maneuvers are grouped in a composite tabulation, the frequency arrays are sometimes bimodal across accelerations and there often seems to be more than one airspeed center of higher frequencies. No standard probability function will fit such an array adequately.

What we would like to find is some method of fitting which does not assume anything about the form of P(xy) or which assumes very little so that it can be used to fit for one class of mission or a composite of missions for any aircraft without giving individual attention to the specific form of each different frequency array involved.

The best solution to the problem would be an explicit expression for the integral (1)

$$\int_{V-10}^{V+10} \int_{g}^{g_{max}} P(xy)dx = P(x > g, V)$$

$$P(x > g, V) = \Lambda_E \tag{2}$$

which could be easily solved for g giving an "Equation for the Envelope,"

$$g = L_E(V). \tag{3}$$

Failing this, we would be happy to settle for an explicit expression for P(xy) such that a high-speed computer could solve the integral equation (1) for the points of the envelope $L_E(V, g)$.

Failing this, we would settle for an explicit expression for $P(x > g|V)$ and a good smooth fit for P(V), enabling us to solve the equations:

$$P(x > g|V) = \frac{\Lambda_E}{P(V)}$$

and obtain a smooth locus of points $L_E(V, g)$ for the envelope.

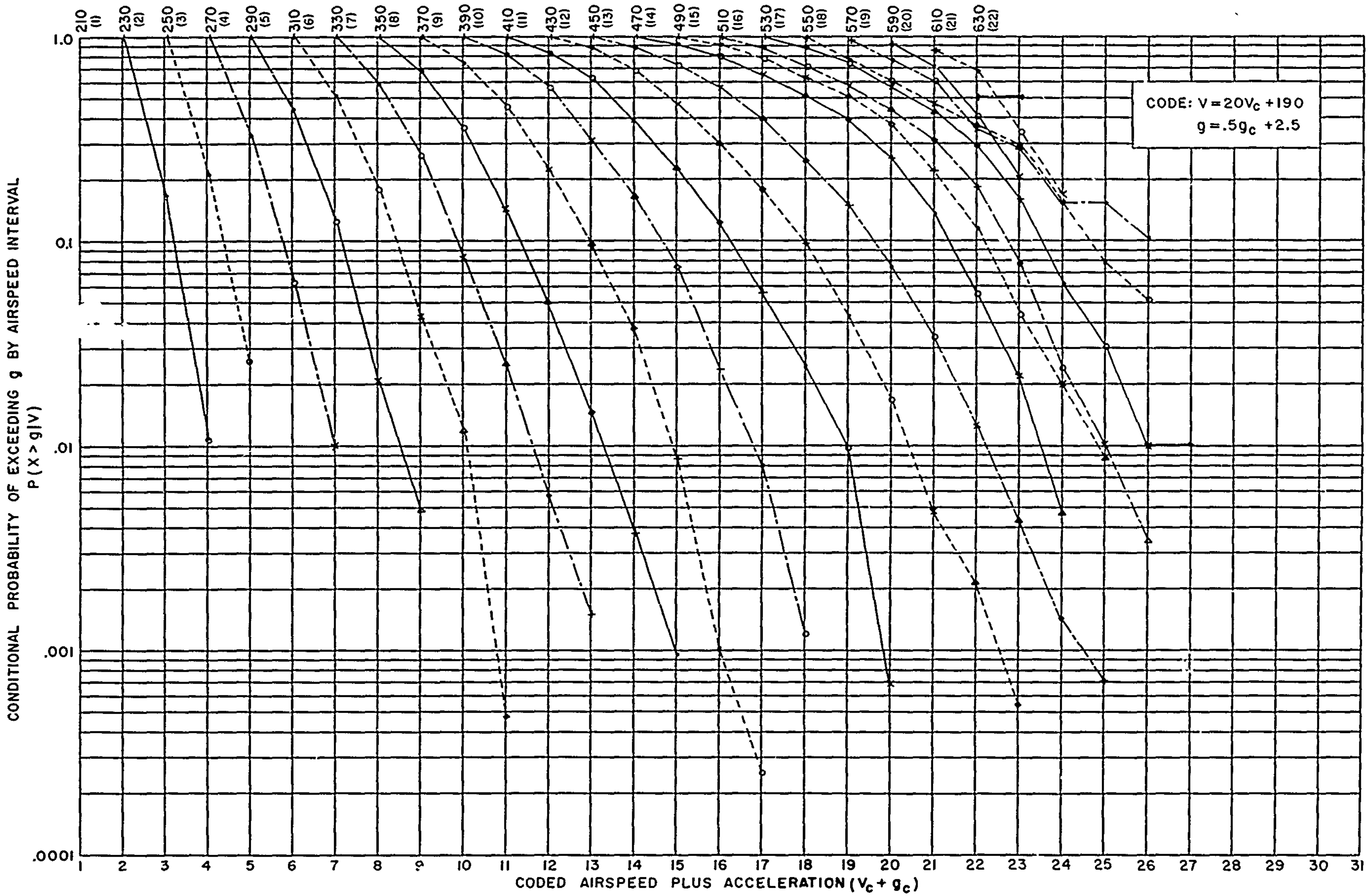
We evolved a method of fitting $P(x > g|V)$ which assumed very little about its form and appeared to give a fair to good fit of the data plotted in probability curves, Figure 3. But we had no luck in finding a smoothing technique for P(V) deriving gracefully from the method of fitting the conditional probability curves, or at least consistent with it. However, a different tack using the same fitting method on joint frequencies, $F(x > g, V)$, seems to work fairly well for extrapolation. The smoothed cumulative frequencies, $\hat{F}$, obtained by fitting a general function to log $F(x > g, V)$, are plotted against their observed counterparts, F, in Figure 4.

Figure 5 contains a plot of the frequencies of exceeding g in each airspeed band, V,

$$F(x > g, V) = N\,P(x > g, V).$$

These curves are plotted in two sections for ease of reading as they would otherwise overlap. Above the F = 1 line are the observed frequencies corresponding to the conditional probabilities plotted in Figure 3. Below the F = 1 line are the extrapolated curves obtained from the fitted function.

Of course, many different methods of fitting the $P(x > g|V)$ individually are available; virtually all assume some basic probability law and the "method of fitting" consists in calculating estimated values for the parameters of the law



CODE: V = 20V_c + 190
g = .5g_c + 2.5
CONDITIONAL PROBABILITY OF EXCEEDING g BY AIRSPEED INTERVAL
P (X > g|V)
CODED AIRSPEED PLUS ACCELERATION (V_c + g_c)
1.0
0.1
.01
.001
.0001
210 (1)
230 (2)
250 (3)
270 (4)
290 (5)
310 (6)
330 (7)
350 (8)
370 (9)
390 (10)
410 (11)
430 (12)
450 (13)
470 (14)
490 (15)
510 (16)
530 (17)
550 (18)
570 (19)
590 (20)
610 (21)
630 (22)
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20 21 22 23 24 25 26 27 28 29 30 31

Figure 3.

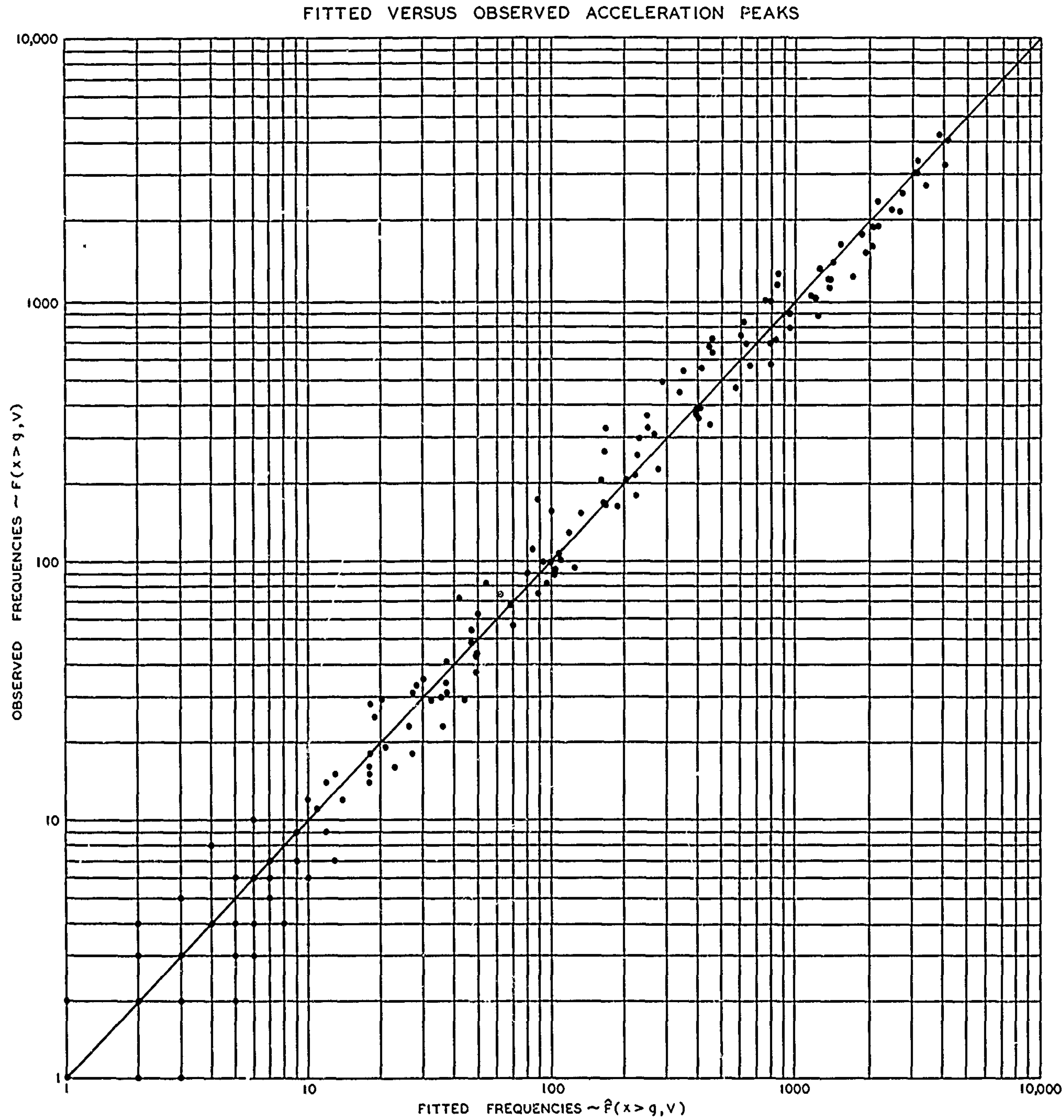

FITTED VERSUS OBSERVED ACCELERATION PEAKS
10,000
1000
100
10
1
OBSERVED FREQUENCIES ~ F(x > g, V)
10
100
1000
10,000
FITTED FREQUENCIES ~ $\hat{F}$(x > g, V)

Figure 4.

EXCEEDANCE FREQUENCY CURVES FOR THE F86F

(BASED ON 909 HOURS OF FLIGHT RECORDS)

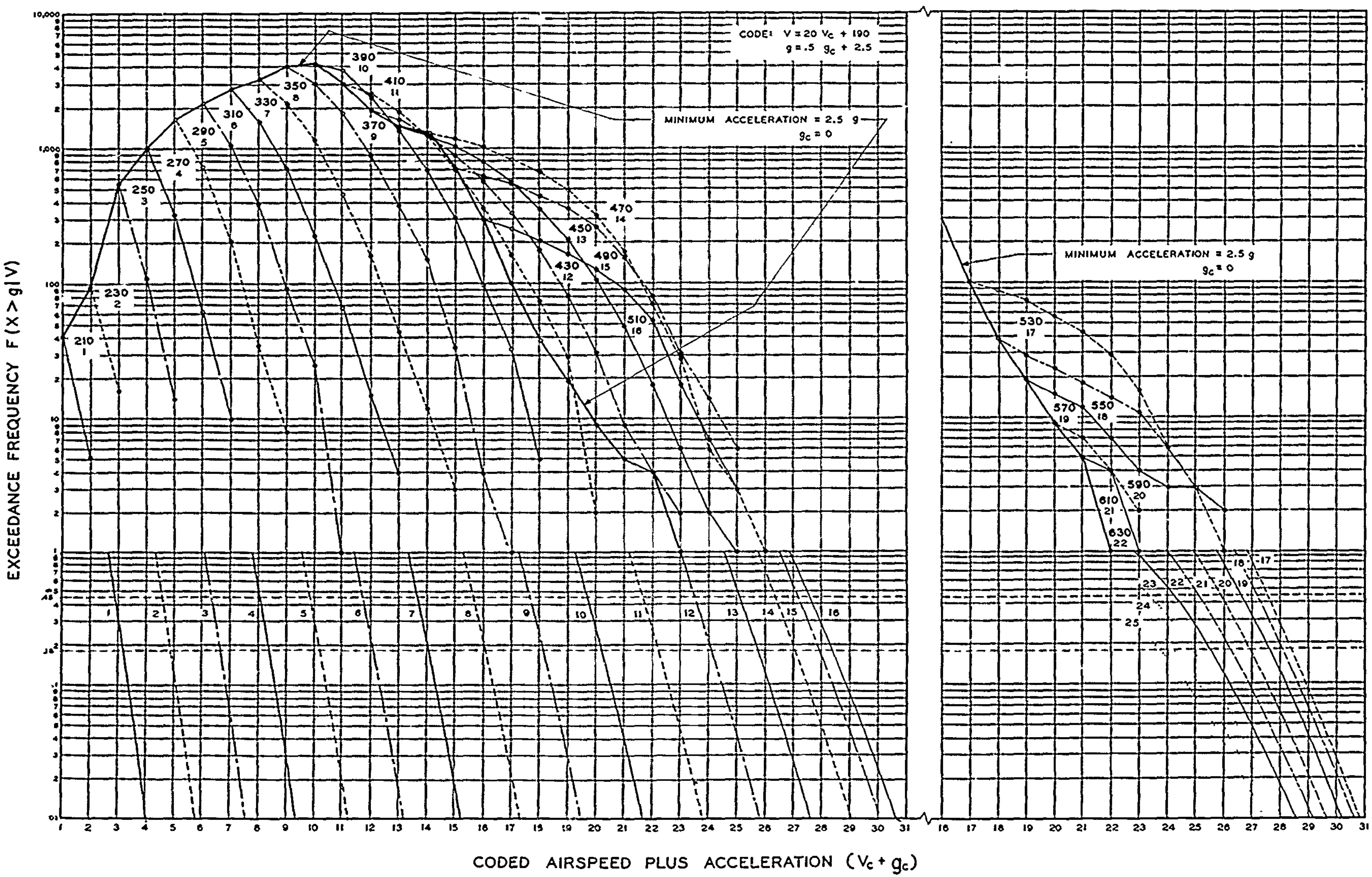


Figure 5.

(function) assumed. But we are trying to find a method which makes few enough formal demands that it can be applied wholesale to the fitting of maneuver load distributions.

Different methods, assuming different probability laws, have sometimes been found to give comparable and good fits to the same data. On the other hand, a method which appears and tests quite adequate for data from one type of mission may not be suitable at all for the same plane on a different mission. In fact, using standard fitting methods, some adjacent airspeed intervals may seem to require wholly different types of probability distributions and methods of fitting, which does not seem reasonable.

Section II describes the standard and not so standard fitting techniques which we have tried and rejected. While we cannot yet report a general or even generally satisfactory method for fitting an unpresumptuous model for P(x,y), we think we have made some progress, if only that of having acquired solid doubt that a completely satisfactory solution exists or can be evolved for the problem as here defined. The procedure we have worked out to date is described in Section III and its limitations and possibilities for further generalization in Section IV.

SECTION II

HISTORY

The flight loads program at WADC has made vast amounts of data available for the prediction of extreme loads. Where a relatively small amount of data is available, the prediction problem degenerates to one of fitting a univariate probability function to the marginal acceleration frequencies.

It has been the custom among American and British agencies concerned with flight loads to assume a Pearson Type III curve (5) (6) (8) (10) (11) (12). Possibly with sufficiently little data this probability function may appear to give a decent fit; however, in our large sample experience it has not looked very good.

Aside from our efforts, Starkey (12), working with gusts, also smoothed the data by airspeed bands in estimating probabilities. He used the P-III approach to fit points in the airspeed bands independently smoothing the moments across airspeed bands in an effort to obtain a smooth gust envelope.

OAR sponsored a paper by E. J. Gumbel (3) on return periods. This paper presents a nice schematic method of fitting and representing time required to reach or exceed curves assuming a simple exponential probability function. It is concerned solely with the univariate case.

An interesting deviation from wholesale fitting of Type III's is Mayer's paper (7) on the spectral density method of fitting maneuver loads. Mayer's study of marginal distributions of five flights of two fighters persuaded him that

they were normal curves truncated at the 50% line. Fitting half normal curves to his data, he then tested goodness of fit and obtained no significant values of chi-square. He does not describe his method of fitting the "half-normal." We obtained fantastically large values of chi-square for the goodness of fit tests applied to truncated normal fits of our fighter data. However, the amount of time covered by our data was very much greater than that used by Mayer.

We first investigated the form of the distributions of acceleration peaks, using Pearson's Method of Moments. Using this method, the standardized third and fourth moments, $\sqrt{\beta_1}$ and β_2 are calculated and plotted on Pearson's graph (see Figure 6), relating values of β_1 and β_2 to the various probability density functions of Pearson's system.

This was done for two fighters, a bomber, and a cargo ship. The distributions of peaks of constant magnitude ($g \pm \Delta g$) across airspeed bands were studied, as well as the magnitude distributions by airspeed band. Also the data from the fighters was broken down by mission and the two perpendicular sets of distributions were examined by mission. Figure 6 contains an example of the graphical results of this procedure. The great preponderance of points are in the Type I region. This was true regardless of the type of plane—bomber, fighter, or cargo—or of fighter mission.

As the sampling distributions of Pearson's distribution criterion are not worked out, a standard statistical test of the hypothesis of a Type I distribution based on the criterion is impossible. Some few of the points for each set do fall on the Type III or V line or in the Type IV region, but we can make no probability statement about the extent of their deviation from the Type I region, although sampling variation in the moments should cause some strays among a sizable number of points.

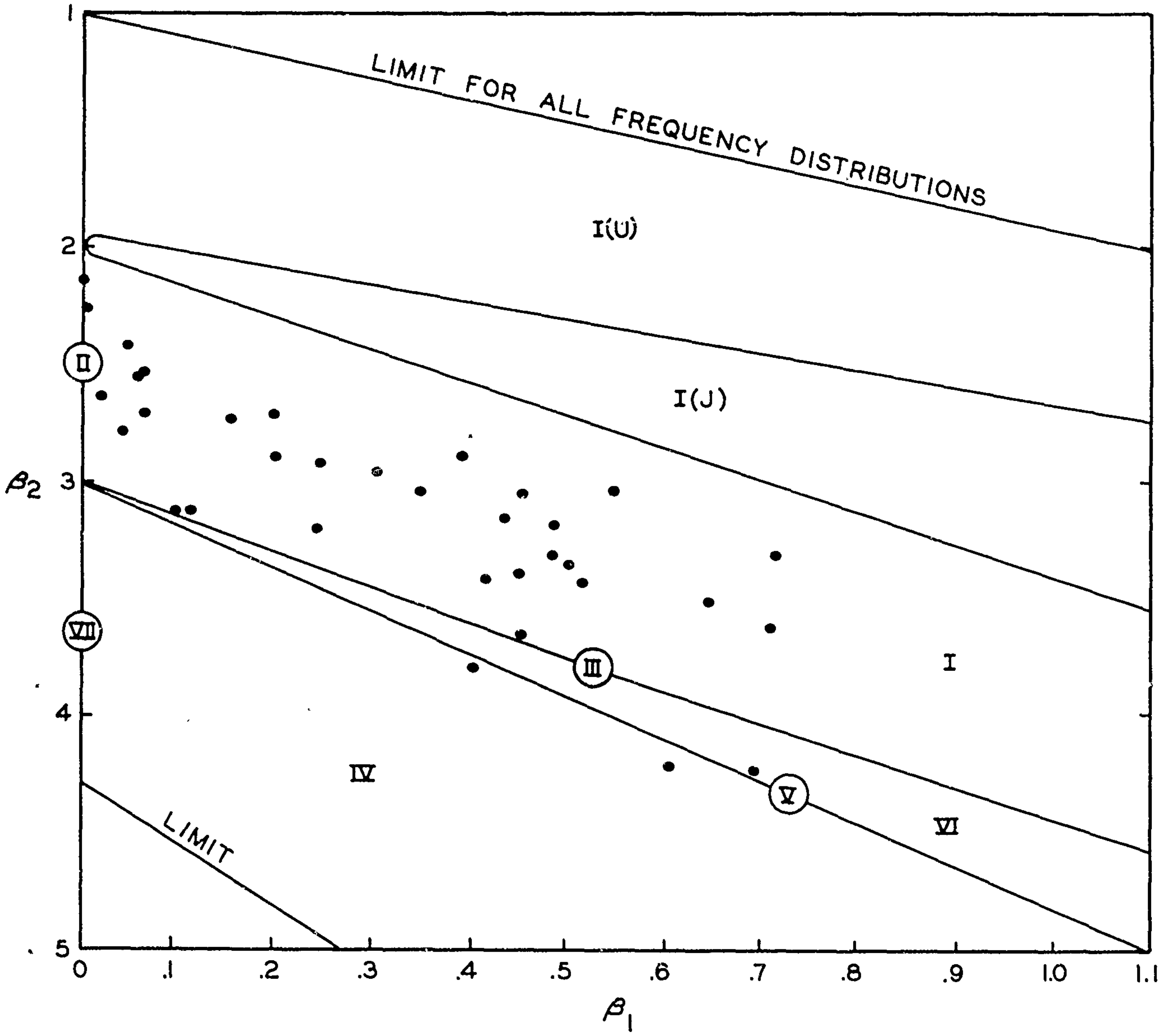
The heavy concentration of points in the Type I region for all of the distributions studied indicates that V-g frequency arrays may well be approximated by a bivariate Type I surface. The Type I function may assume a J shape (or even a U shape) or a bell shape, although it is usually skew, and it has an upper and lower limit, that is, the range of a Type I variable is limited on both sides:

$$P(x)dx = \frac{1}{B(a,b)}\left[\frac{x-x_o}{R}\right]^{a-1}\left[1-\frac{x-x_o}{R}\right]^{b-1} dx,$$

$$x_o \leq x \leq x_o + R.$$

The cumulative Type I probability is the incomplete Beta function, $I_y(a, b)$. The upper and lower limits on magnitude of acceleration peak and on airspeed band of occurrence make good physical sense, are in conformity with the physical facts of C_{n_a} curves, maximum airspeed, graph reading truncation at 2.0 g, and so on. Most of the observed g|V curves tend toward the J shape and are so indicated on

CHART RELATING THE TYPE OF PEARSON FREQUENCY CURVE TO VALUES OF β_1 AND β_2



STANDARDIZED THIRD AND FOURTH MOMENTS OF THE CONDITIONAL AIRSPEED DISTRIBUTIONS OF POSITIVE PEAKS EXPERIENCED BY FIGHTER AIRCRAFT IN COMBAT AND TRAINING AND BY BOMBERS IN REGULAR OPERATIONS.

Figure 6.

the Pearson graph, while the more symmetrical V|g distributions fall in the regular Type I region.

If a satisfactory bivariate generalization of the Type I surface existed, it would appear that we were all set. The Pearson system of curves has not lent itself nicely to bivariate generalization; however, Narumi (9) was successful in evolving a bivariate Type I distribution function for the case of complete "similarity" in the geometric sense, of the $P(x|y)$ curves and the $P(y|x)$; that is, when the moments of the conditional distributions in either direction are linearly related. Plots of the moments of our conditional distribution indicated a relation which is certainly not linear so we cannot use Narumi's function and hence cannot fit a bivariate Type I function.

By way of consolation, however, we observe that in estimating upper and lower limits for our conditional distributions independently, we obtained several estimated upper limits well within the observed range of variation. Such estimates are clearly useless for prediction purposes, particularly since we cannot put confidence limits on them. Probably, if it were feasible to fit a bivariate Type I, part of its estimated perimeter would fall within the observed range, making it useless for prediction. We conclude that while a Type I is clearly indicated for interpolation purposes, for prediction we need a function without an upper limit or else one whose upper limit is surely higher than the observed maximum peaks.

Another area we explored was that of transformations of the data to get a normal or truncated normal surface. The bivariate normal distribution function, of course, has many agreeable features. For combat data the logarithmic transformation appeared to bring us closest to a truncated normal surface. Training data from the same types of aircraft, however, were better fitted by the truncated normal without transformation. Plots of both types of data on probability paper indicated an upper, as well as lower, truncation of the conditional acceleration distributions; this is in agreement with the Pearson Type I indication given by the method of moments.

These effects are briefly illustrated using combat and training data collected from the F86F. The two marginal acceleration probability distributions are plotted on normal probability paper versus an arithmetic scale in Figure 7 and versus a logarithmic scale in Figure 8.

Normal probability graph paper has cumulative normal probability distances on one scale and logarithmic cycles or arithmetic distances on the other. In using this paper with grouped data, one simply computes the cumulative relative frequencies and plots them against the upper (lower) limits of the class intervals.

If the data is normally distributed, the plot is about a straight line on arithmetic paper. If the logarithm of the variable is normally distributed, the

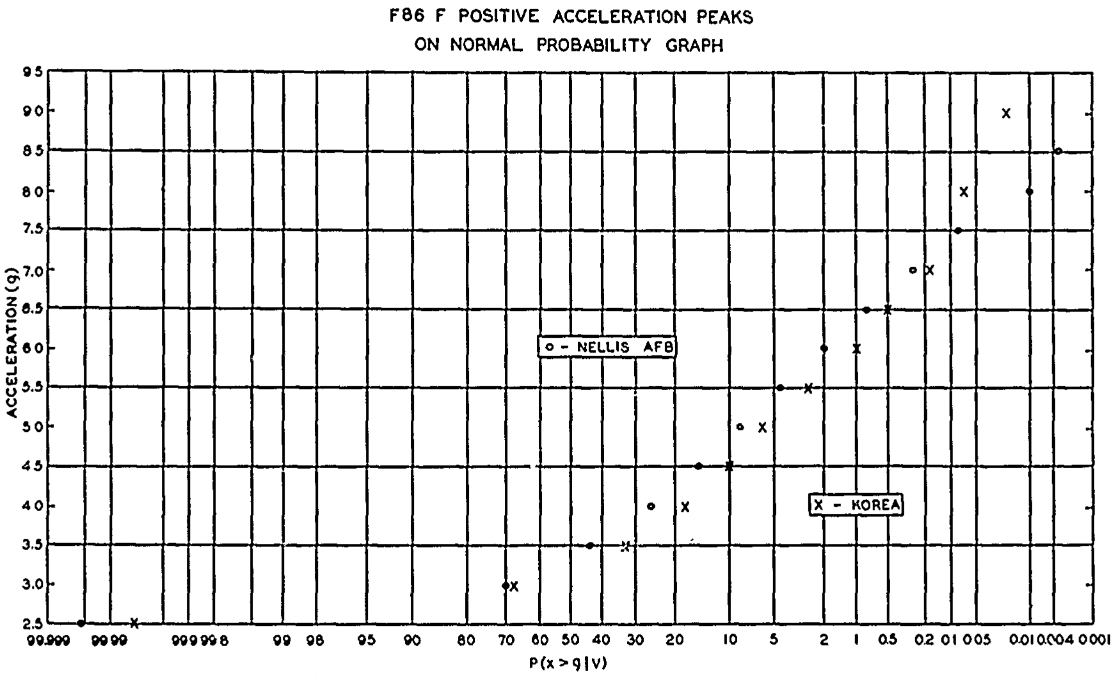


Figure 7.

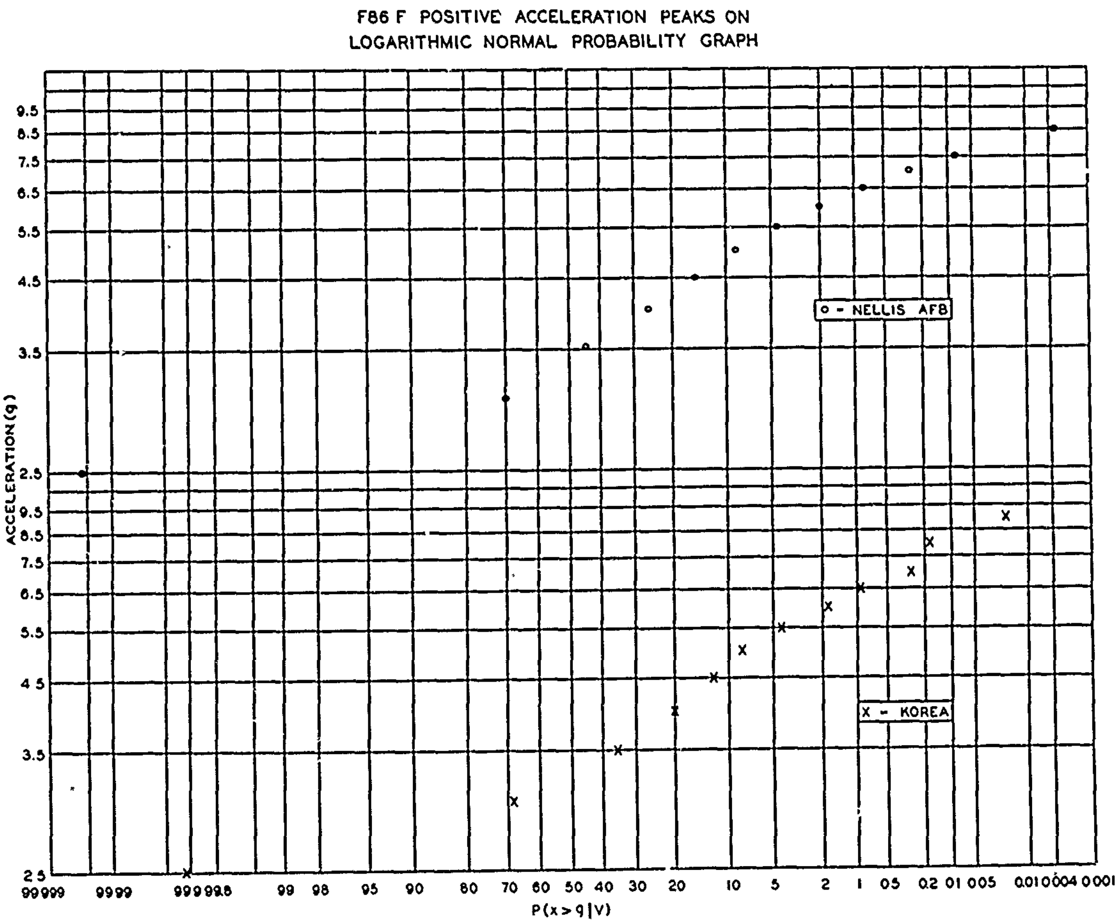


Figure 8.

plot is concave upwards on arithmetic paper, but about a straight line on log-normal paper. A bending toward horizontal at the extremes with a good long straight line segment in the middle suggests truncation where the bending occurs.

While a one-sided truncation of normal data is easily handled (4), the two-sided truncation, with upper truncation point unknown, is not (2). Even if upper truncation points could be estimated, they would not be dependably useful for prediction purposes, coming as they sometimes might within the observed range of variation.

SECTION III

DESCRIPTION OF THE PROCEDURE

Using the conceptualization described in the introduction, the V-g frequency array in Figure 1 is a sample from an infinite population of peak magnitudes governed by a probability law P(x,y). Plots of the conditional distributions, $\hat{P}(x > g|V)$, on semi-logarithmic paper (Fig. 3) indicate that this function is probably exponential.

Assuming the general form

$$P(x > g, y) = e^{X(gy)}$$

where X(gy) is an analytic function in the region of interest and expandable in a Taylor series, the coefficients of as many terms, $g^j y^i$, as necessary or justified would be fit by the method of least squares to the logarithms of the relative frequencies.

Our empirical model then is

$$\log \hat{P}(x > g, V) = \sum_{i,j=0}^{l,n} a_{ij} V^i g^j + \text{ERROR}$$

or working with joint frequencies (which differ only by an additive constant in the model)

$$\log \hat{F}(x > g, V) = \sum_{i,j=0}^{l,n} a_{ij} V^i g^j . \tag{4}$$

In Figure 3 the conditional probabilities of exceeding g are plotted against a sliding acceleration scale. Successive curves correspond to successive airspeed bands. The g scale begins farther to the right for each airspeed curve, for example, the vertical line for 5.0 g at an airspeed in 250 ± 10 mph is the 4.5 g line for V in 270 ± 10, the 4.0 g line for V in 290 ± 10, and so on.

This same scheme is followed in the graph of frequencies in Figure 5. Dashed, dotted, and solid lines are used for the different airspeeds to make reading easier. As mentioned earlier, the top portion of the graph contains the observed exceedance frequencies and points are connected by straight line segments there. Beneath the F = 1 lines are the extrapolations of the smooth curves obtained by fitting eleven terms in the expansion of (4):

$$\log \hat{F}(x > g, V) = a_0 + a_1 g + a_2 V + a_3 g^2 + a_4 gV + a_5 V^2 + a_6 g^2 V + a_7 gV^2 + a_8 V^3 + a_9 g^2 V^2 + a_{10} gV^3 + a_{11} g^2 V^3. \tag{5}$$

Because of errors of measurement, it would not seem worthwhile to fit terms of higher degree, particularly since our purpose is extrapolation.

Table 1. Analysis of Regression of log F (x > g, V)

Source of Variation	D. F.	Sum of Squares	Mean Square
Total	160	177.938789	
Regression	11	174.641994	
g	1		54.234794
V	1		14.949918
g^2	1		1.799250
gV	1		2.091355
V^2	1		90.557892
g^2V	1		.246509
gV^2	1		8.810058
V^3	1		1.412795
g^2V^2	1		.446969
gV^3	1		.072236
g^2V^3	1		.020218
Residual	149	3.296985	.022127
R^2 = 99.15%.	Std. error = .14875		

The variation in logarithms of the frequencies accounted for by each term in (5) is shown in table 1. 99.2% of the total variation in the logarithms was accounted for by this model which seems quite good by usual standards. A standard error of estimate, based on residual variation, of .14875 corresponds to a healthy error potential when translated back into frequencies, however. Bearing in mind the fact of a highly irregular array to begin with, having very low frequencies at the extremes of airspeed, this may be as good as we can hope to do. It is the best overall fit we have obtained so far in our endeavors.

The Envelopes

The expected number of peaks exceeding g in E hours is P(x > g, V) EN/H, where N is the number of positive peaks observed in H hours. According to the

definition of the envelope and equation (2), $L_E(gV)$ is the set of points for which

$$\frac{EN}{H}\left[P(x > g, V) \cdot \Lambda_E\right] = 1$$

or

$$P(x > g, V) = \frac{H}{EN} \ .$$

Since $F(x > g, V) = N\,P(x > g, V)$, the points on the envelope for E hours are given by setting

$$\hat{F}(x > g, V) = \frac{H}{E}$$

and solving for g corresponding to each V.

This is easily accomplished using the graph of Figure 5. The values of g for each airspeed are obtained from the intersections of the $F = H/E$ line with the airspeed curves. The $F = 1$ line gives the fitted points for the hours of observation H.

Table 2. Arithmetic for the Envelope

		909.2 Hours			2000 Hours			5000 Hours		
Airspeed	V_c	$g_c + V_c$	g_c	g	$g_c + V_c$	g_c	g	$g_c + V_c$	g_c	g
210	1	2.6	1.6	3.3	2.9	1.9	3.5	3.2	2.2	3.6
230	2	4.3	2.3	3.7	4.6	2.6	3.8	4.9	2.9	4.0
250	3	6.1	3.1	4.0	6.4	3.4	4.2	6.7	3.7	4.4
270	4	7.8	3.8	4.4	8.1	4.1	4.6	8.4	4.4	4.7
290	5	9.6	4.6	4.8	9.8	4.8	4.9	10.2	5.2	5.1
310	6	11.4	5.4	5.2	11.7	5.7	5.4	12.0	6.0	5.5
330	7	13.3	6.3	5.7	13.7	6.7	5.8	14.1	7.1	6.0
350	8	15.3	7.3	6.2	15.7	7.7	6.4	16.1	8.1	6.6
370	9	17.3	8.3	6.6	17.7	8.7	6.9	18.2	9.2	7.1
390	10	19.2	9.3	7.1	19.7	9.7	7.4	20.2	10.2	7.6
410	11	21.2	10.2	7.6	21.7	10.7	7.9	22.2	11.2	8.1
430	12	23.0	11.0	8.0	23.5	11.5	8.3	24.1	12.1	8.6
450	13	24.6	11.6	8.3	25.1	12.1	8.6	25.8	12.8	8.9
470	14	25.7	11.7	8.4	26.4	12.4	8.7	27.1	13.1	9.1
490	15	26.5	11.5	8.3	27.2	12.2	8.6	27.9	12.9	9.0
510	16	26.8	10.8	7.9	27.6	11.6	8.3	28.4	12.4	8.7
530	17	26.7	9.7	7.4	27.5	10.5	7.8	28.4	11.4	8.2
550	18	26.3	8.3	6.7	27.2	9.2	7.1	28.1	10.1	7.6
570	19	25.7	6.7	5.8	26.6	7.6	6.3	27.6	8.6	6.8
590	20	24.8	4.8	4.9	25.9	5.9	5.5	26.9	6.9	6.0
610	21	23.9	2.9	4.0	25.1	4.1	4.6	26.2	5.2	5.1
630	22	22.7	.7	2.9	24.2	2.2	3.6	25.4	3.4	4.2
650	23	21.0	- 2.0	1.5	23.3	.3	2.6	24.8	1.8	3.4
670	24				22.1	- 1.9	1.5	24.2	.2	2.6
690	25							23.7	- 1.3	1.8

The numbers across the bottom of the graph in Figure 5 are coded values $g_c + V_c$. Airspeeds are coded $V_c = 1, 2, \ldots, n$ and accelerations are coded $g_c = 0, 1, 2, \ldots, k$. To obtain g from the coded scale, one reads the abcissa $g_c + V_c$ corresponding to the intersection of an airspeed frequency curve with the line H/E, subtracts the coded airspeed value, and uncodes g_c. For example, the V = 250 mph ($V_c = 3$) curve crosses $\hat{F} = 1$ at $g_c + V_c \doteq 6.1$. From the code given in the upper right hand corner of the graph, we have $g = 0.5(3.1) + 2.5 = 4.0$. The numerical details of these steps in constructing the envelopes for E = 909.2, 2000, and 5000 hours ($\hat{F} = 1.0$, 0.455, and 0.182, respectively) appear in Table 2.

SECTION IV

CONCLUSIONS

The procedure described here originated in the idea of fitting a general exponential type of joint probability function by expanding the exponent in a Taylor series and estimating the coefficients by the method of least squares. However, there is nothing in the assumptions or method to force the estimated function to yield an exponential function having the properties of a probability law. Rather than solving a distribution problem, this approach reconstructs it within the compass of the powerful methods of general regression theory. The problem becomes one of constructing a flexible regression model to represent exponentially a set of rates of occurrence.

From the standpoint of accounting for the variability of the logarithms, this approach is very good (99.15%) and the estimated exceedance frequencies still look reasonably good. However, working back to the original V-g array, we find fairly large discrepancies with the original data at low acceleration magnitudes. Further, values of the chi-square goodness of fit tests imply very strongly that this procedure does not reproduce the original array adequately, particularly in the region of high frequencies; the lower frequencies are better estimated.

In conclusion, this procedure for smoothing exceedance frequencies is particularly well-suited for simple graphical estimation of predicted occurrence rates. The model used is flexible and its flexibility can be increased further by adding or substituting terms which are functions of V and g in the regression equation (5), for instance, a_{12}/V or $a_{13}\log(c_o - g)$.

The model can be generalized to three or more flight parameters, very simply by adding terms to the regression equation. Adding altitude (h), for example, we would fit

$$\log F(x > g, V, h) = \sum a_{ijk} V^i g^j h^k .$$

The capacity of the computing equipment used to solve for the coefficients imposes the primary limitation on the number of variables (or terms) that can be added to the model and, thereby, on the versatility and generality of the method.

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APPLICATION OF POWER SPECTRAL TECHNIQUES TO DYNAMIC RESPONSE FLIGHT TESTING

By Harry Press and Thomas L. Coleman

ABSTRACT

The effects of airplane flexibility on the structural responses of an airplane are discussed in general terms and the difficulty of interpretation of dynamic response flight test data noted. Particular attention is given to the problem of airplane responses to rough air. The significance of the airplane frequency response functions is noted and their determination chosen as a basic goal of dynamic flight testing. In many cases, the disturbances and the airplane responses involve statistical or random type processes and in these cases the power spectral techniques, with which this paper is concerned, are required. These techniques are also useful in other contexts.

Two basic methods are outlined for frequency response determination from random type data; the power spectrum and cross-spectrum methods. Their relative merits are considered in terms of data requirements, their susceptibility to distortion by extraneous effects or "noises" and to statistical or sampling errors. The concept of a coherency function is defined and the usefulness of this function described. In order to illustrate the application of the power spectral techniques, examples are presented of the results obtained in the study of the responses of a flexible airplane to rough air.

INTRODUCTION

In recent years it has become increasingly clear that the structural loads and stresses resulting from such disturbances as atmospheric turbulence, runway roughness and acoustic noise can be considerably amplified by airplane flexibility. For the case of airplane response to gusts, the magnitudes of the stress amplifications are frequently of the order of 30 percent and in some cases are in excess of 100 percent. Inasmuch as stress amplification of the order of 30 percent can act to reduce fatigue life by an order of magnitude, the effects of flexibility require close attention in fatigue studies. In general, these flexibility effects are complex and the analytic technology for the calculation of the structural response is still being developed. As a consequence, experimental techniques for studying these problems and particularly full-scale flight testing techniques are of wide interest.

During the last ten years a number of flight investigations of the effects of flexibility on the structural response of airplanes to gusts have been made. The primary purpose of these tests was to establish the magnitude and character of these effects. In general, the procedures used involved simple comparisons of measured strains with a reference measurement which was usually based on the airplane center-of-gravity acceleration or acceleration measurements at wing nodal points. These studies have served to indicate the magnitude of the dynamic effects but have had a number of severe limitations. The principal difficulty encountered was in the inability to obtain a suitable reference measurement from which to gauge flexibility effects. In many cases, strong coupling existed between the rigid body and elastic modes and complicated the desired comparisons. A second limitation was the inability to make direct comparisons between measured and calculated results inasmuch as the input was not determined.

During recent years two significant developments have served to make a more satisfactory approach to the gust response flight test problem possible. These were; first, the development of techniques for the direct measurement of turbulence in flight and secondly, the development and application of power spectral techniques for handling random type disturbances such as gusts in dynamic response analyses. These two developments were utilized in a recent NASA study of the response of a flexible airplane, the B-47 airplane, to atmospheric turbulence. The results obtained in this study have served to indicate how gust measurements and power spectral techniques may be applied in the analysis of dynamic flight test data and specifically, for the determination of frequency response functions. These same techniques also appear to have wide applicability to problems involving other types of disturbances.

This paper presents a brief review of some of the significant results obtained in this study. Further details may be found in reference 1. The paper is divided into three sections containing:

(a) A brief discussion of some of the characteristics of flexibility effects

(b) A description of the power spectral methods of frequency response determination and some of the problems involved

(c) Some illustrative examples of the results obtained in the application of these methods to the problem of airplane response to atmospheric turbulence

EFFECTS OF FLEXIBILITY ON STRUCTURAL RESPONSE

The manner in which flexibility affects the structural strain response of airplanes is schematically illustrated in three different ways in Figure 1; in terms of the time history, the power spectrum, and the overall frequency distribution of peak strains. These three representations are complementary and help to make clear the character of the flexibility effects.

As can be seen from the time histories, the effects of flexibility are to complicate the nature of the response. In particular, the superposition of higher frequency vibrations associated with structural modes are evident. These frequency aspects are perhaps more clearly seen in the power spectra on the lower right which reflect the contributions of the various frequencies to the respective time histories. The first peak in each case is associated with the short period mode while the peak at higher frequency for the flexible case reflects the first mode in bending. The principal effect of flexibility consists of the addition of a secondary peak in power associated with, in this case, the first symmetrical bending mode. Although not shown on the sketch, significant secondary peaks at higher frequencies are frequently present and are associated with higher structural modes.

For fatigue considerations the overall frequency distribution of peak strains is generally significant. In this representation, the effects on flexibility can be broken into two parts. One part is essentially a rotation of the distribution curve arising from the increased power or increased root-mean-square strain value for the flexible response. The second part is an upward translation of the curve arising from an increase in the characteristic or average frequency of the oscillations. This increased frequency is associated with the increased radius of gyration about the vertical axis for the

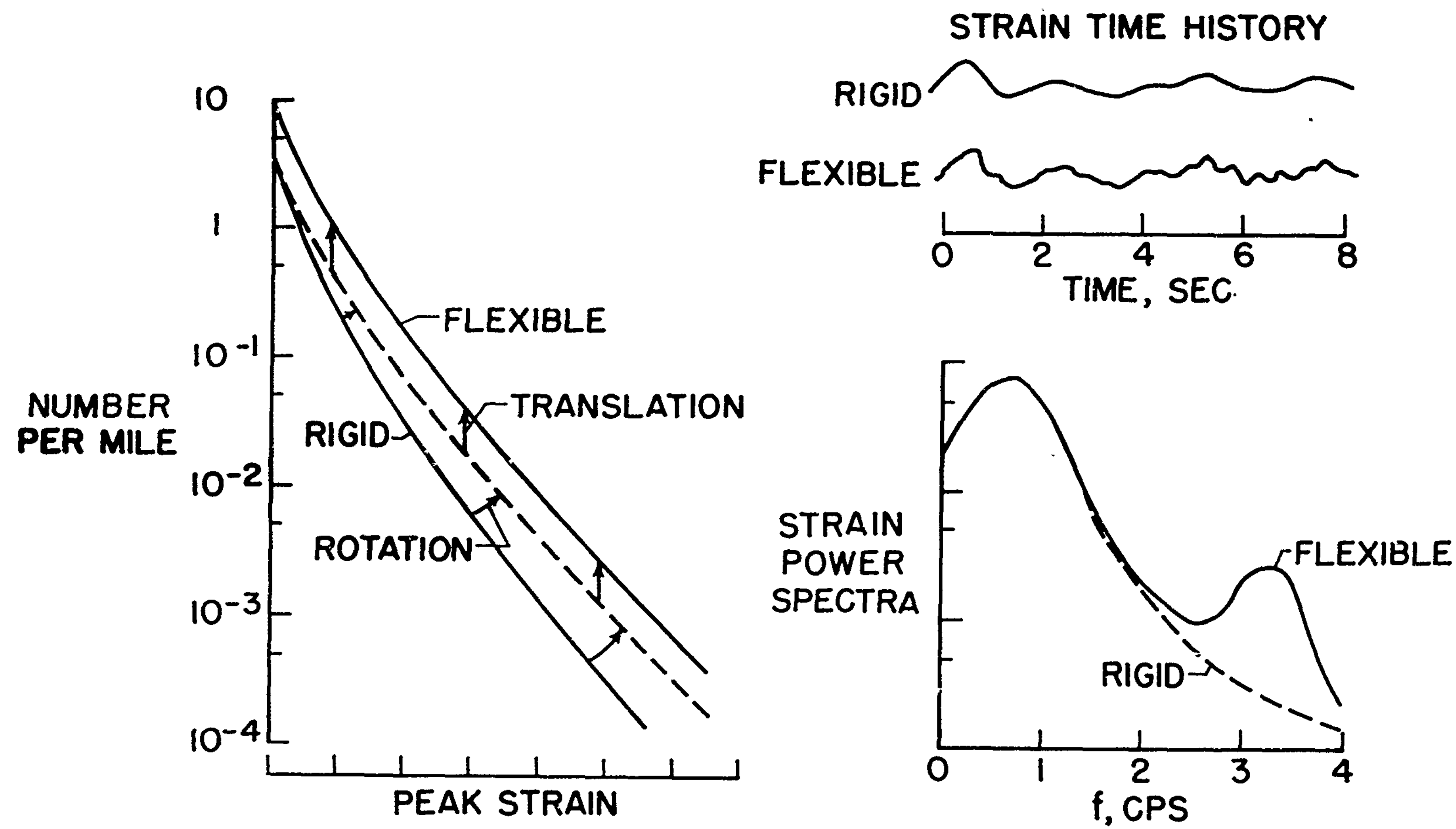


Figure 1.- Effects of wing flexibility on the time history, power spectrum, and peak distribution of wing root bending strain.

power spectrum for the flexible case. These two effects, the rotation and the translation, both act to increase the number of stresses or strains at given levels and thus proportionately reduce the fatigue life. The fatigue life for the flexible airplane is frequently one-tenth, or less, the value for the rigid case.

As mentioned earlier, dynamic response flight testing has been largely used to establish the magnitude and character of the dynamic effects. For this purpose, measurements of the actual flexible response are compared with some reference measurements, normally the center-of-gravity acceleration or acceleration measurements at the airplane nodal points which are taken as measures of a quasi-static airplane response. In some cases the reference measurement is relatively unaffected by flexibility effects and this procedure is reasonably satisfactory. This is the case illustrated in the spectral curves of Figure 1 and applies to relatively stiff structures where the first bending mode is sufficiently far removed frequency-wise from the short period mode in the longitudinal case to preclude much coupling action. Unfortunately, the more flexible nature of many contemporary vehicles prevents this simple approach. In many cases the rigid body and flexible modes are closely coupled and, in addition, particularly for swept wings, significant static aeroelastic effects are present to further complicate such comparisons.

Figure 2 illustrates the separate effects of vertical motion, pitching, static aeroelasticity and dynamic structural flexibility for the center-of-gravity acceleration of the B-47 airplane. The figure shows the calculated frequency response functions for the center-of-gravity acceleration response for various assumptions in the calculations. Four curves are shown. These are for the assumption of freedom in vertical motion only, freedom in vertical motion and pitch, freedom in vertical motion, pitch and a few elastic modes. A fourth reference condition, termed the quasi-static case, is also shown. For this case the airplane is considered free in vertical motion and pitch but with stability derivatives reflecting the static aeroelastic effects.

Comparison of the results shown in Figure 2 for the various conditions reveal a number of points of interest. For present purposes, perhaps the most significant points that can be deduced are the relatively large reflection of static aeroelastic effects indicated by a comparison of the vertical motion and pitch case with the quasi-static case and the large effects of flexibility indicated by the comparison of the flexible case with the quasi-static case. The effects of dynamic flexibility cause a substantial amplification in the response not only in the neighborhood of the first bending mode frequency at about 1.4 cps, but also over the short period mode. Thus, the measured center-of-gravity acceleration is quite different from the rigid or quasi-static airplane case and cannot provide an adequate reference condition for measuring flexibility effects. Similar considerations may also be expected for acceleration measurements at wing nodal points.

In view of the foregoing considerations, it would appear difficult, if at all possible, to obtain reliable measures of structural amplifications directly from flight test data. For this reason it is both desirable and necessary to tailor flight testing techniques to obtain basic airplane response characteristics. The frequency response functions of an airplane is one convenient way of representing the basic response characteristics. The basic role of flight testing may thus be considered to be the determination of such response functions. The frequency response functions have the added merits of providing results that are independent of the gust input and results that are directly

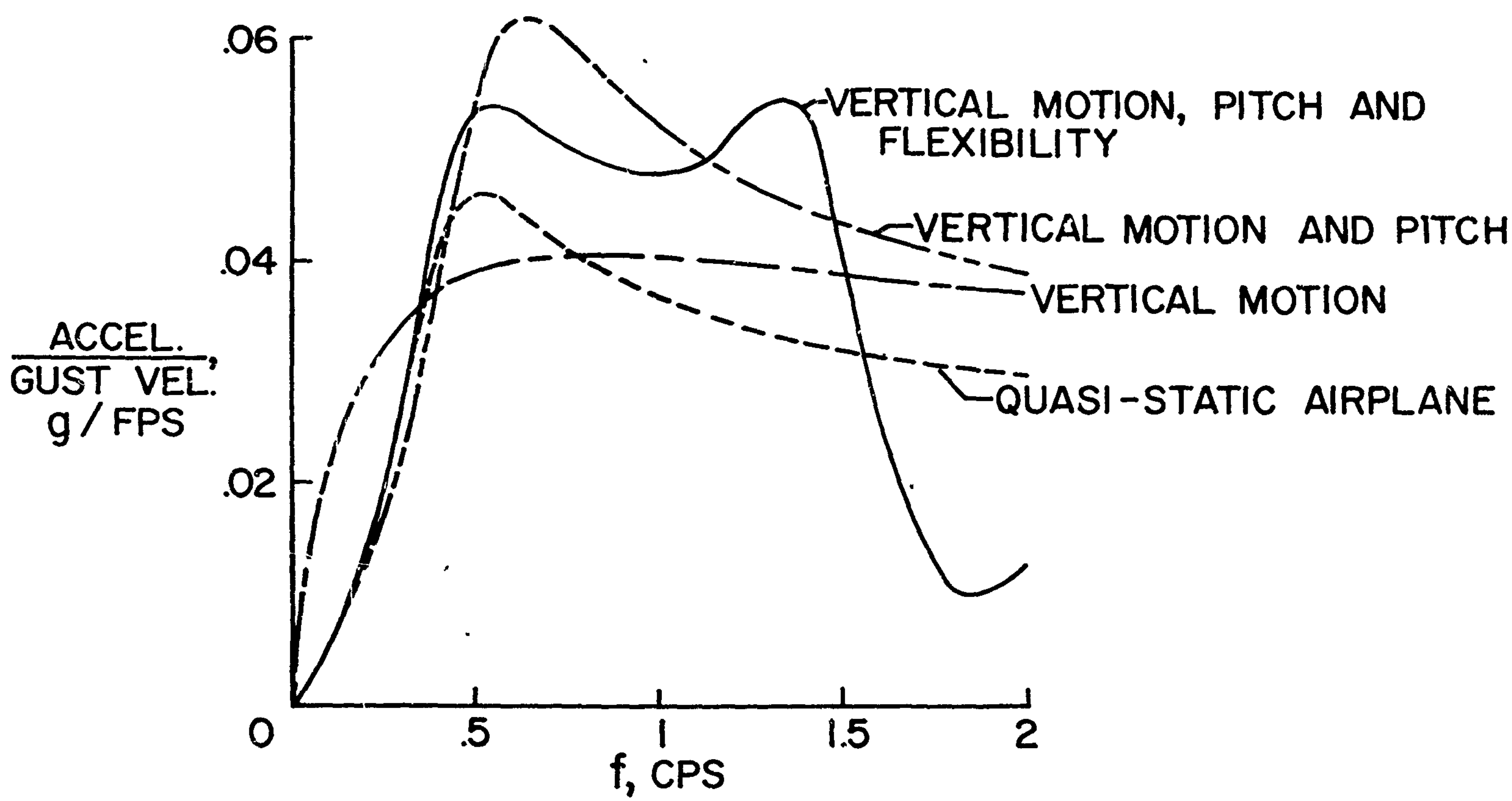


Figure 2.- Effects of vertical motion, pitch, and wing flexibility on the calculated center-of-gravity acceleration frequency response function.

comparable to calculated results. The remainder of this paper will be concerned with describing the power spectral approach to frequency response function estimation and some illustrative results obtained in applications.

METHODS FOR DETERMINING FREQUENCY RESPONSE FUNCTIONS

Basic Methods

The basic relations that are useful for frequency response determination for random-type data are shown in Figure 3. The sketch at the upper part of the figure illustrates the simple system visualized--namely a stationary random disturbance x, acting upon a linear airplane system characterized by the frequency response function $H(f)$, and yielding an output response y. For this case, the measurements of the input disturbance x and the output response y may be used to estimate the frequency response function $H(f)$. Two methods are known for this purpose and are given by equation (1) and (2) in Figure 3. Namely,

$$\left| H_S(f) \right|^2 = \frac{\Phi_y(f)}{\Phi_x(f)} \qquad (1)$$

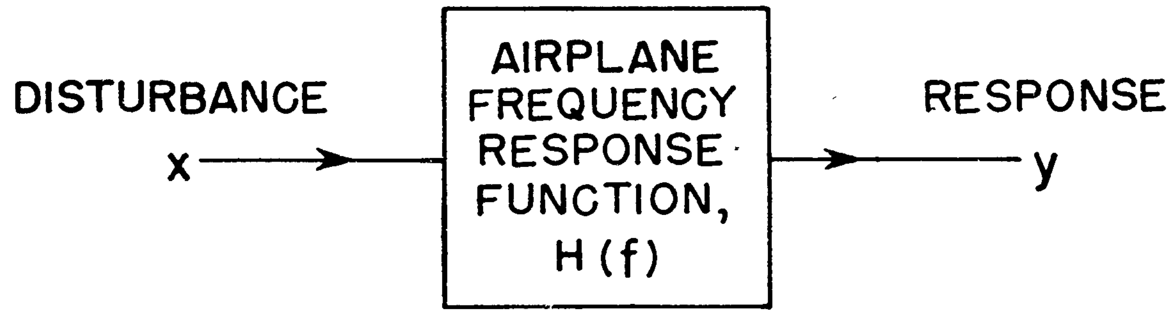
$$H_C(f) = \frac{\Phi_{xy}(f)}{\Phi_x(f)} \qquad (2)$$

$\Phi_x(f)$, $\Phi_y(f)$

$\Phi_{xy}(f)$

The vertical bars $||$ designate the absolute value of the complex quantity. The subscripts "s" and "c" are used to designate estimates obtained by the two methods; "s" for the spectrum method, which is based on a ratio of power spectra and "c" for the cross spectrum method which involves $\Phi_{xy}(f)$ the cross-spectrum between the disturbance x and the response y. One significant difference exists between these two methods. The spectrum method permits the estimation of only the amplitude of the frequency response function while the cross-spectrum is in general a complex quantity, and this method permits the estimation of both the amplitude and phase of the response. The cross-spectrum method is thus more general and for this reason more useful in many applications.

For the case of a stationary random input acting on a linear system, such as visualized in Figure 3, both methods should yield identical estimates of the amplitude of the frequency response function. In practice, this is seldom the case for very many reasons. First, the system of concern is seldom as simple as the one illustrated in Figure 3 but frequently involves extraneous elements or noises that complicate the picture. Such complicating elements will be discussed in greater detail later. In addition, sample sizes for $x(t)$



METHODS

I - SPECTRUM
$$|H_S(f)|^2 = \frac{\Phi_y(f)}{\Phi_x(f)} \quad (1)$$

II - CROSS SPECTRUM
$$H_C(f) = \frac{\Phi_{xy}(f)}{\Phi_x(f)} \quad (2)$$

COHERENCY FUNCTION
$$\gamma^2(f) = \frac{|H_C(f)|^2}{|H_S(f)|^2} = \frac{|\Phi_{xy}(f)|^2}{\Phi_x(f)\Phi_y(f)} \leq 1 \quad (3)$$

Figure 3.- Methods for experimentally determining frequency response functions and the coherency function.

and $y(t)$ are in general limited with differing effects on the two methods. As a consequence, the estimates obtained by the two methods are in general different and sometimes differ widely.

Coherency Function

If the ratio of $|H_C(f)|^2$ and $|H_S(f)|^2$ is taken as indicated by equation 3 of Figure 3 a very useful function is obtained. This function designated as $\gamma^2(f)$ and termed the coherency function is given by:

$$\gamma^2(f) = \frac{|\Phi_{xy}(f)|^2}{\Phi_x(f)\,\Phi_y(f)} \qquad (3)$$

The coherency function plays a central role in the analysis of the relations between two random processes and proves quite useful for present purposes. The value of the quantity $\gamma^2(f)$ will in general vary with frequency and its value at a given frequency may be viewed as a measure of the degree to which two processes are correlated or linearly related at the given frequency. For a simple system of the type shown on Figure 3, $\gamma^2(f)$ will be equal to 1 at all frequencies. At the other extreme, if two processes are unrelated $\gamma^2(f)$ will be equal to zero. For the cases of concern in practice the value of $\gamma^2(f)$ will, in general, lie between 0 and 1 with the departure from a value of 1 dependent upon the degree of extraneous effects and the nature of the complications.

From equation 3, it is clear that a reduction in $\gamma^2(f)$ from a value of 1 implies that the values of $|H_S(f)|$ and $|H_C(f)|$ differ. The interpretation of the results for this case is thus complicated and requires close analysis of the system and the source of the reduced coherency.

In general, reductions in coherency from a value of one reflect a loss in reliability of the frequency response function estimates in two distinct ways as illustrated in Figure 4. These are by:

1. A possible distortion in the estimates of the frequency response function from its true value (upper figure)

2. A reduction in the statistical reliability (lower figure) as measured by the width of confidence bands.

The upper sketch on Figure 4 illustrates the character of distortion effects. Such distortions may lead to under-estimates or over-estimates of the true frequency response function. The magnitude of such distortions will in general vary with frequency and is related to the reduction in coherency, as will be indicated.

In addition to the distortions, reduced coherency in general leads to reduced statistical reliability as illustrated by the widths of the confidence bands on the lower sketch on Figure 4. These bands define the limits within which the true values may be expected to lie; the relatively wider bands at high frequencies being associated with a lower coherency level at these frequencies for this case.

Sources of Reductions in Coherency.

It will be helpful to examine the mechanisms by which coherency reductions arise in practice and some of the specific sources for such reductions. The

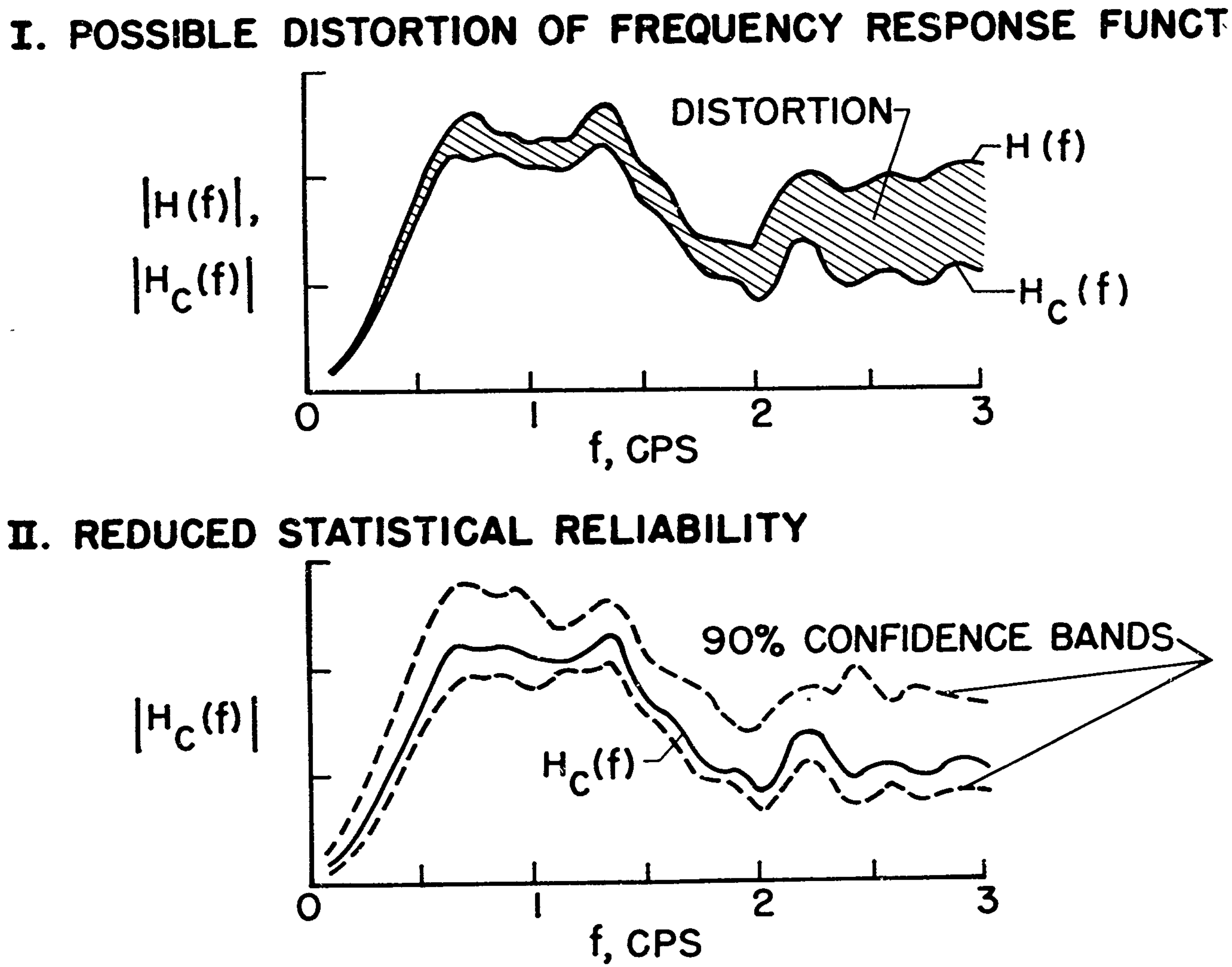


Figure 4.- Effects of reductions in coherency on frequency response functions.

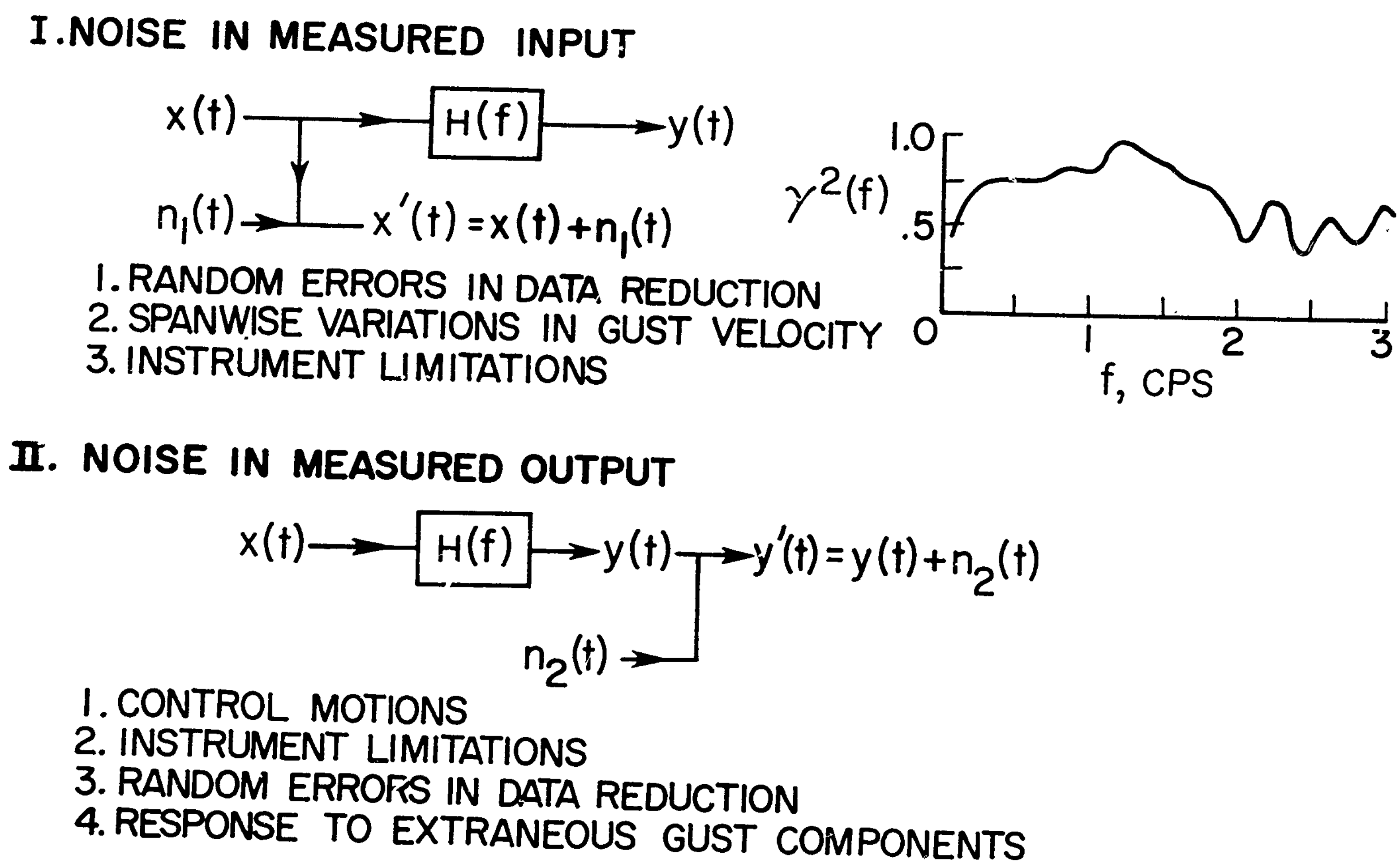


Figure 5.- Sources of reductions in coherency.

general picture is summarized on Figure 5. Two basic cases are considered: Noise in the input and noise in the output. In both cases the coherency is reduced but estimates of the frequency response functions obtained by the two methods are affected differently.

For the noise-in-input case, the block diagram shown indicates that the simple input-output system is complicated by the introduction of an extraneous disturbance $n_1(t)$ to the measurement of the input. In practice such disturbances arise from random errors in record reading, round-off errors, random instrument errors, and spanwise variations in gust velocities. If $x'(t)$ is measured in place of $x(t)$ the coherency will be reduced and substantial errors will arise.

For the noise-in-output case, the simple input-output system is complicated by the addition of a noise on the output side. In the gust response case, for example, such noises arise by the action of pilot control motions, instrumentation limitations, random errors in record reading, round-off errors, and the effects of extraneous gust components, such as the airplane longitudinal response arising from side gust disturbances.

In both cases the seriousness of the noise effects, in introducing distortions and in reducing statistical reliability, are dependent upon the relative level of the noise or the noise to signal ratios. The losses in statistical reliability can, in general, not be recovered; however, corrections for the distortions are possible and general expressions for the corrections for these distortions have been derived in Reference 1. For the special case of noises which are incoherent (unrelated) to the signals (input and output), the results are particularly simple. Fortunately this case appears to apply well in practice.

Corrections for Distortions.

The corrective adjustments required to compensate for the distorting effects introduced by incoherent noises are summarized in Figure 6. The adjustments required to the estimates obtained by the spectrum and cross-spectrum methods for three cases — noise-in-input, noise-in-output, and noise in both input and output are as follows:

$$|H(f)|_{true} = \frac{|H_C(f)|}{\gamma^2(f)} = \frac{|H_S(f)|}{\gamma^2(f)} \tag{4}$$

$$|H(f)|_{true} = |H_C(f)| = \gamma(f)|H_S(f)| \tag{5}$$

$$|H(f)|_{true} \approx |H_S(f)|\left(1 + \frac{\Phi_{n_1}(f)}{2\Phi_x(f)} - \frac{\Phi_{n_2}(f)}{2\Phi_y(f)}\right) \tag{6}$$

$$\approx |H_C(f)|\left(1 + \frac{\Phi_{n_1}(f)}{\Phi x(f)}\right) \tag{7}$$

Figure 6 - Corrections for distortions of frequency response functions due to noises in the input and output.

where $\Phi_{n_1}(f)$ and $\Phi_{n_2}(f)$ are the power spectra of the noise in the input and output respectively and are assumed to be independent. For both the noise-in-input and noise-in-output cases, adjustments to the estimates obtained by two methods, equations 4 and 5, are simple and straightforward and involve only the coherency function. Note that for the noise-in-output case, the cross spectrum method is undistorted and requires no correction for the reduction in coherency. A further point worth noting is that the phase angles obtained from the cross-spectrum method are generally relatively insensitive to the random noises considered. For the case in which both input and output are contaminated by noise, the situation is a little more complex and the adjustments required involve more than the coherency function alone. As indicated by equations 6 and 7 additional information is required on the ratios of the noise to signal power spectra for this case. Such information is sometimes available.

These results serve to indicate that distortions arising from extraneous noises can frequently be corrected.

Statistical Reliability

The effects of coherency level on statistical reliability are shown on Figure 7. The figure indicates the statistical reliability of the estimates of the frequency response function $H_C(f)$ in terms of the percent error in the amplitude and the phase angle error. The results are shown for the case of forty spectral estimates $(m = 40)$ over the frequency range. For smaller values of m, the statistical reliability is increased. Examination of Figure 7 indicates that statistical reliability increases with increasing sample size and increasing coherency. For a given level of reliability the sample size required increases rapidly as coherency is reduced. For example, as indicated by the dashed lines in the figures, for a 20-percent reliability in amplitude, the sample size increases from 350 at $\gamma^2 = .9$ to roughly 800 at $\gamma^2 = .75$ and about 2000 at $\gamma^2 = .50$. It is thus clear that relatively high levels of coherency are desirable in the interests of statistical reliability.

Data Reduction Considerations.

In the application of the foregoing techniques to frequency response estimation, considerable care is required in the selection of sample sizes and in the data reduction procedures. These procedures are discussed in detail in Reference 1 and are based on results reported in References 2 and 3. A few comments appear warranted here.

In general, the data analysis can proceed by digital or analog techniques. Only the digital approach will be discussed although similar considerations apply to the analog case. In the digital case, time history measurements are first converted to a sequence of successive readings taken at equally spaced intervals of time. The significant data reduction considerations are summarized in Figure 8. The time interval between readings Δt and the number of readings N define the sample time duration T. The choice of these two quantities, Δt and N, are quite important in practice and strongly affect the reliability of the results obtained.

The choice of the appropriate reading interval Δt in any application depends upon the highest frequency of interest. As indicated on Figure 8

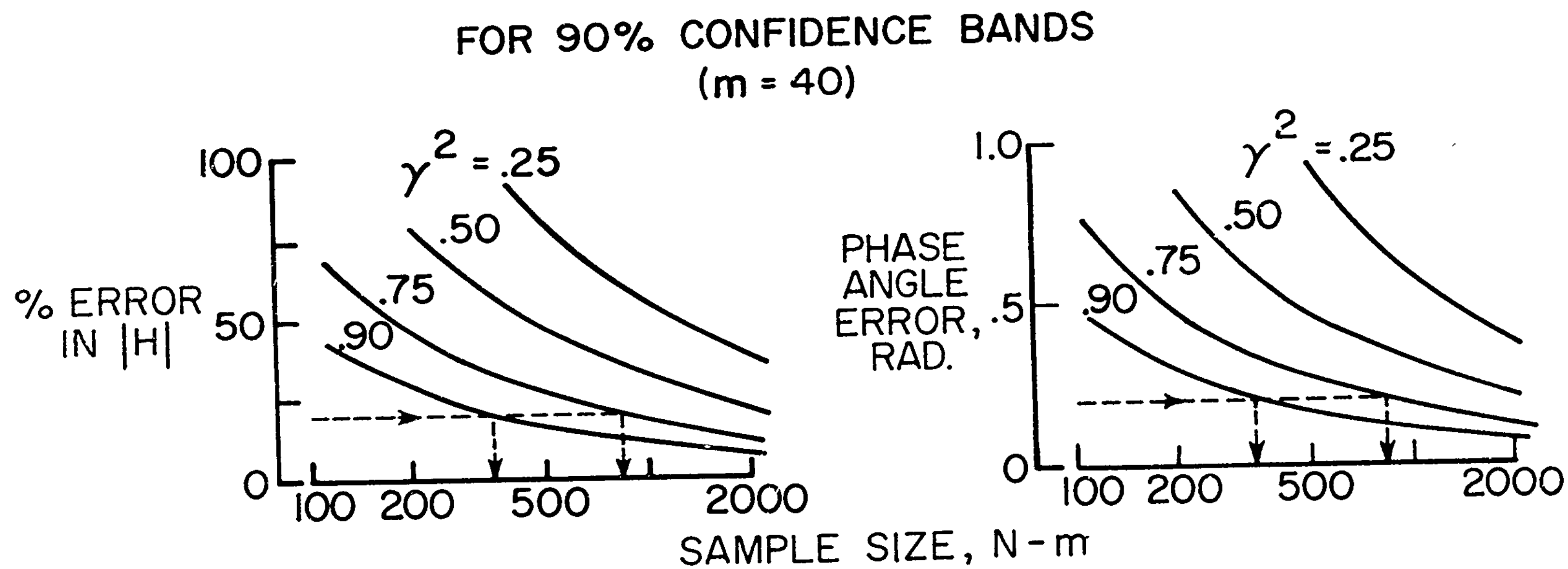


Figure 7.- Effect of sample size and coherency on the statistical reliability of frequency response functions.

SAMPLE LENGTH, $T = \Delta t N$

1. READING INTERVAL Δt

$$\Delta t = \frac{1}{2f_o}$$

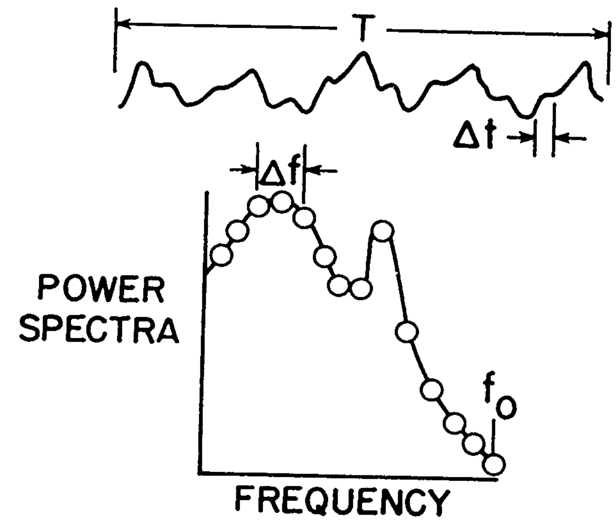
f_o IS HIGHEST FREQUENCY OF INTEREST

2. NUMBER OF READINGS N DEPENDS ON:
   (a) FREQUENCY RESOLUTION DESIRED

   NUMBER OF ESTIMATES $m = \dfrac{2f_o}{\Delta f}$

   (b) COHERENCY γ^2
   (c) STATISTICAL RELIABILITY DESIRED

3. ADDITIONAL CONSIDERATIONS
   (a) PREWHITENING
   (b) FOLDOVER



STATISTICAL RELIABILITY EXAMPLE

90% CONFIDENCE LEVEL FOR

$$\left.\begin{array}{l} N = 1500 \\ f_o = 5\text{ CPS} \\ \Delta f = .25\text{ CPS} \\ \gamma^2 = .75 \end{array}\right\} \text{IS:} \quad \begin{array}{l} \pm 15\%\text{ IN AMPLITUDE} \\ \pm 9^\circ\text{ PHASE ANGLE} \end{array}$$

Figure 8.- Sampling and data reduction considerations.

$$\Delta t = \frac{1}{2f_o}$$

where f_o may be considered the highest frequency of interest. However, if appreciable power exists in any of the signals at higher frequencies, precautions are necessary to prevent adverse effects. Specifically, power at frequencies above f_o tends to "foldover" in such digital calculations and show up as spurious power at related frequencies below f_o. For example, power at f_o+f_1 will show up at f_o-f_1. Such adverse effects can normally be avoided by filtering out the higher frequency content in the records before computing the power spectrum. In addition, if the power spectra are not very flat, it is frequently advisable to transform the time history by an appropriate linear operation to achieve a flatter spectrum. This operation has been termed "prewhitening" by Tukey. The usual digital computational sequence of auto-correlation function and power spectra determination and frequency response estimation is then performed. Appropriate compensation for any prefiltering and prewhitening must be included.

The number of readings N required in a given application, as indicated on Figure 8, depends upon three factors:

(a) The frequency resolution desired

(b) The coherency level γ^2

(c) The statistical reliability desired

Inasmuch as estimates obtained from finite data of power spectra and, in turn, frequency response functions, are at best averages over frequency band widths, some degree of "smearing" is unavoidable. The wider the averaging band width, the better the statistical reliability but the lower the resolution. Thus in practice a compromise is necessary. The effective averaging band width Δf can be approximately expressed as

$$\Delta f = \frac{2f_o}{m}$$

where m is the number of estimates obtained in the digital calculation over the frequency band 0 to f_o as indicated by the sketch on the upper right of Figure 8. The quantity Δf is thus simply twice the frequency interval between the successive estimates. The choice of m in the calculation thus determines Δf and the frequency resolution that is obtained. Peaks in power of width Δf will certainly be obscured.

The specification of the resolution desired Δf or m permits the choice of the required number of readings N. This choice is then based on the coherency level and the desired statistical reliability from charts such as Figure 7.

In order to indicate the levels of sample size and statistical reliability that may be achieved in practice, the results obtained for an example are shown on the lower right of Figure 8. The values of N, Δf and γ^2 chosen are representative of values that were employed. The reliability of the results in terms of the 90 percent confidence bands are $\pm$ 15 percent for the amplitude of the frequency response function and $\pm$ 9 degrees for the phase angle. These

provide very satisfactory levels of precision for most practical purposes.

If the coherency levels are high, the estimates of frequency response function are generally very reliable and in fact on a relative basis are even more reliable than the estimates of the individual power spectra. This condition contributes to making the present techniques particularly useful in practice.

ILLUSTRATIVE RESULTS

In the interest of concreteness, several examples of the results from the B-47 airplane gust response study will be presented briefly. Most of these results are discussed in further detail in reference 1.

Figure 9 shows the estimates obtained for the amplitude of the root bending strain response as obtained by the two methods — spectrum and cross spectrum methods. Also shown in the lower figure is the associated coherency function which varied between about .6 to almost 1 over the frequency range from about .25 cycles per second to almost 1.75 cycles per second. Above that value it dropped off rapidly. The frequency response estimates obtained with the two methods are reasonably consistent below 1.75 cycles per second, the differences essentially arising from the differing distortion effects associated with the coherency reduction. At 2 cycles per second and higher the results differ widely reflecting the very low coherency and in fact, the unreliable nature of the results in this region.

Figure 10 illustrates the spanwise variation in the effects of flexibility on the strains. Frequency response functions are shown for various spanwise positions $y/\frac{b}{2}$. Also shown is a reference curve based on the airplane center-of-gravity acceleration. For this comparison, all the results are shown in terms of the equivalent acceleration at the various spanwise positions. The results indicate the pronounced effects of the first mode in bending at about 1.5 cycles per second on the frequency response functions and in particular the increasing amplification as we move outboard on the wing.

Perhaps the most important results obtained from the application of the present techniques are experimental measurements that are independent of the disturbance input and are suitable for direct comparisons with calculations. Figure 11 shows such a comparison for the measured and calculated center-of-gravity acceleration for the B-47 airplane. In general the agreement between the calculations and measurements are reasonably good. Perhaps the most serious discrepancy is in the difference in frequency of the first mode in wing bending between the measured and calculated results. The calculations show the first mode peak at about 1.3 cycles per second, while the measurements indicate first mode at somewhat higher frequency. Such comparisons provide a basis for a detailed evaluation of the analytic technology and for its improvement.

CONCLUDING REMARKS

The present paper has served to outline methods for determining frequency response functions to gust disturbances from flight measurements in rough air. The power spectral techniques outlined are straightforward in application but require considerable care in data reduction and in interpretation of results.

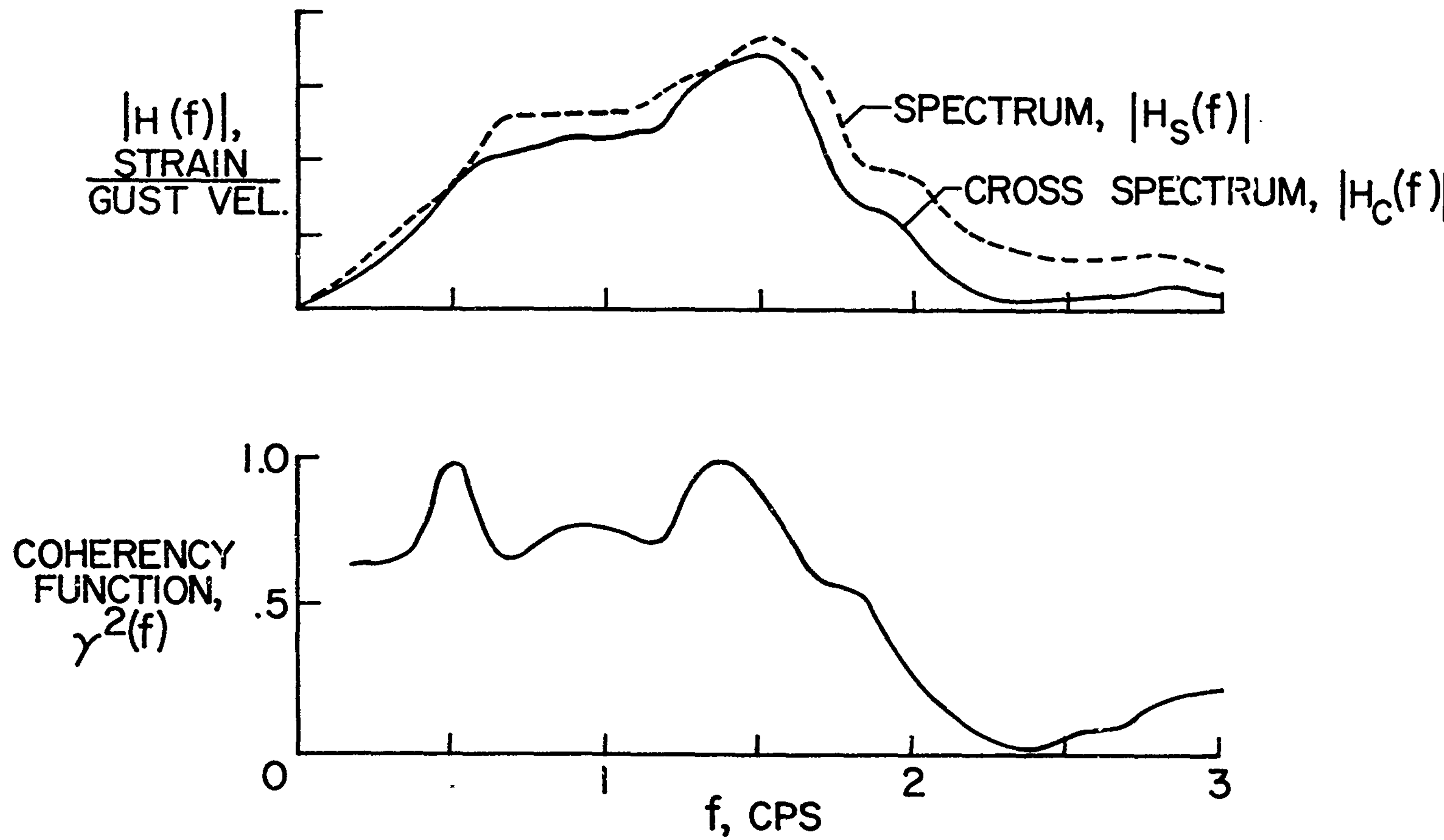


Figure 9.- Comparison of frequency response functions for wing root bending strain determined by the spectrum and cross spectrum methods.

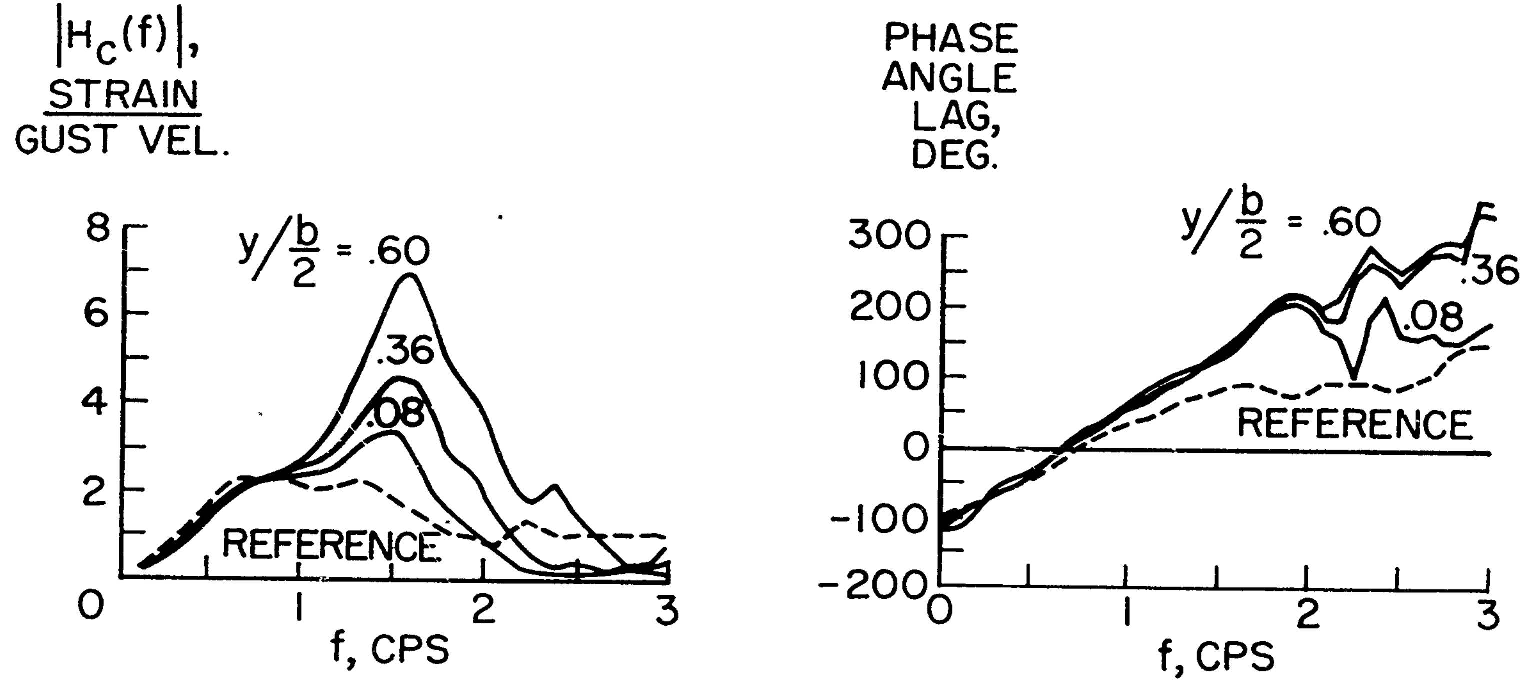


Figure 10.- Spanwise variation of the effects of flexibility on wing bending strains.

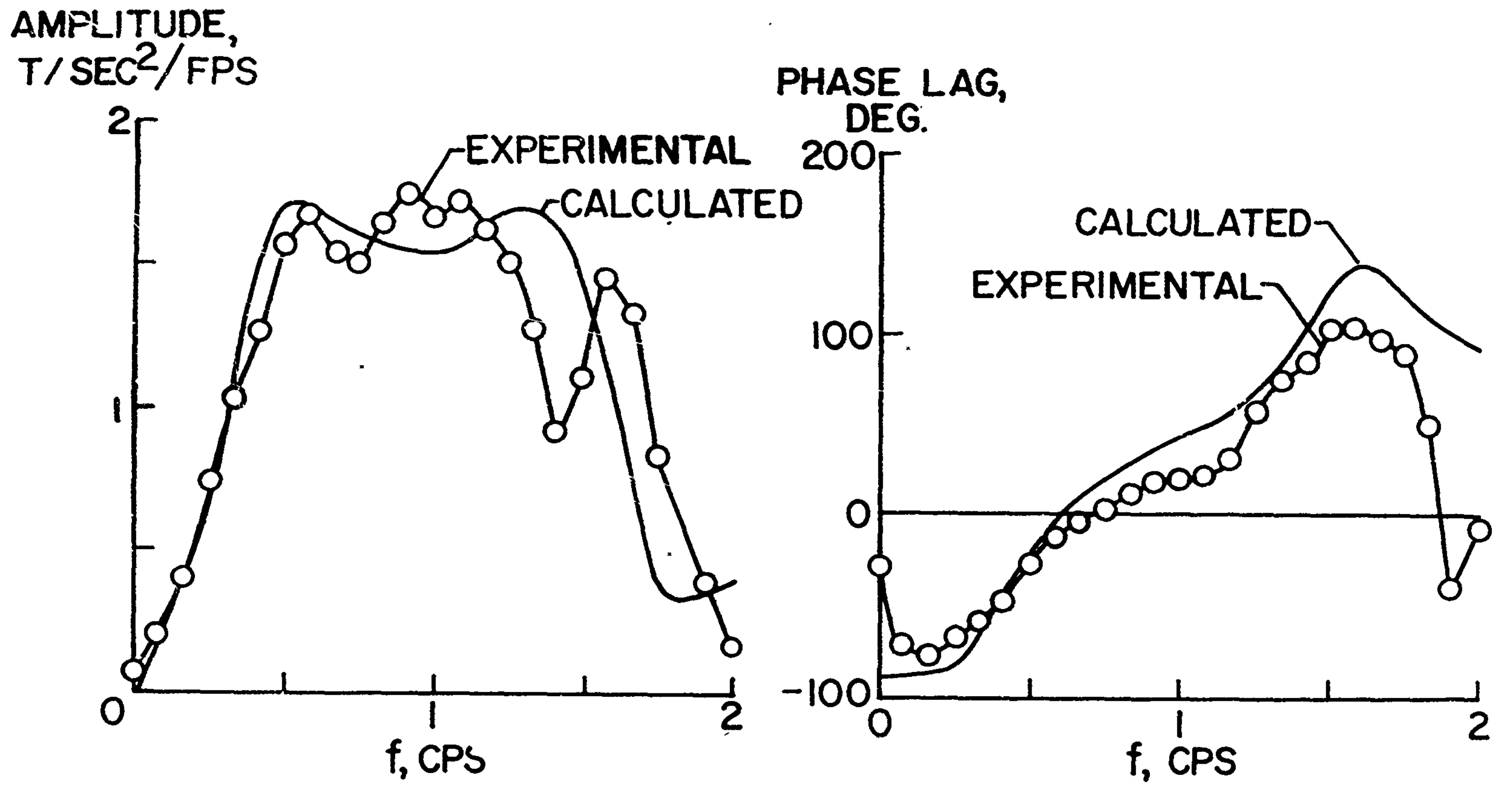


Figure 11.- Comparison of calculated and experimental frequency response functions for center-of-gravity acceleration.

Particular attention is required in order to account for the effects of extraneous noises. Illustrative results obtained in the application of these methods have also been presented. The significant conclusion reached is that reliable estimates of the frequency response functions can be obtained from flight test data. The methods also appear applicable to other response problems involving random type disturbances.

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DATA COLLECTION AND ANALYSIS SYSTEMS FOR THE B-66 GUST SURVEY PROJECT

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The Douglas Aircraft Company is currently engaged in conducting a low level turbulence survey project utilizing a B-66 airplane equipped with gust sensory and airplane response instrumentation. The data is being recorded by means of a standard frequency modulated airborne tape recording system. The airborne tapes are converted after flight to IBM digital format using a Douglas FM to digital tape data reduction facility. Because of limitations experienced with the FM system, consideration is being given to application of an airborne digital magnetic tape recorder and a simplified flight tape to computer tape conversion system to the project. Comparisons of the two data recording and recovery systems will be made as applied to collection and analysis of atmospheric gust and airplane response data.

TEST PROGRAM OBJECTIVES

The low level turbulence program is designed to sample and reduce to useable form the statistical representations of the gust intensity and frequency distribution. Turbulence characteristic of different kinds of terrain, terrain surface contour, surface winds, weather, atmospheric temperature gradients and other conditioning factors are to be determined. The B-66, as a probing device, has been instrumentated and programmed to sample turbulence over terrain of the Continental United States by making sampling runs as indicated on figures 1 and 2.

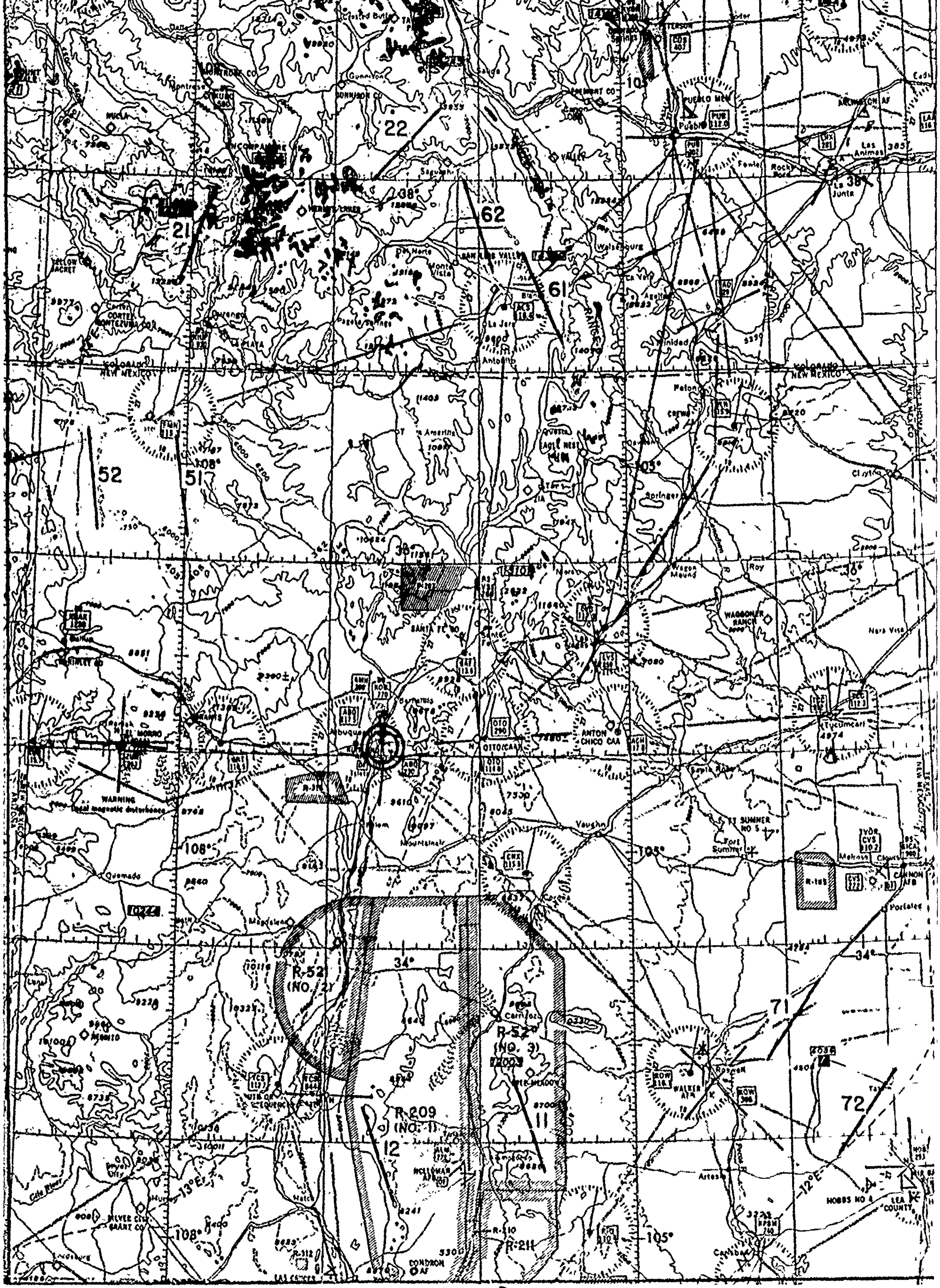


Figure 1

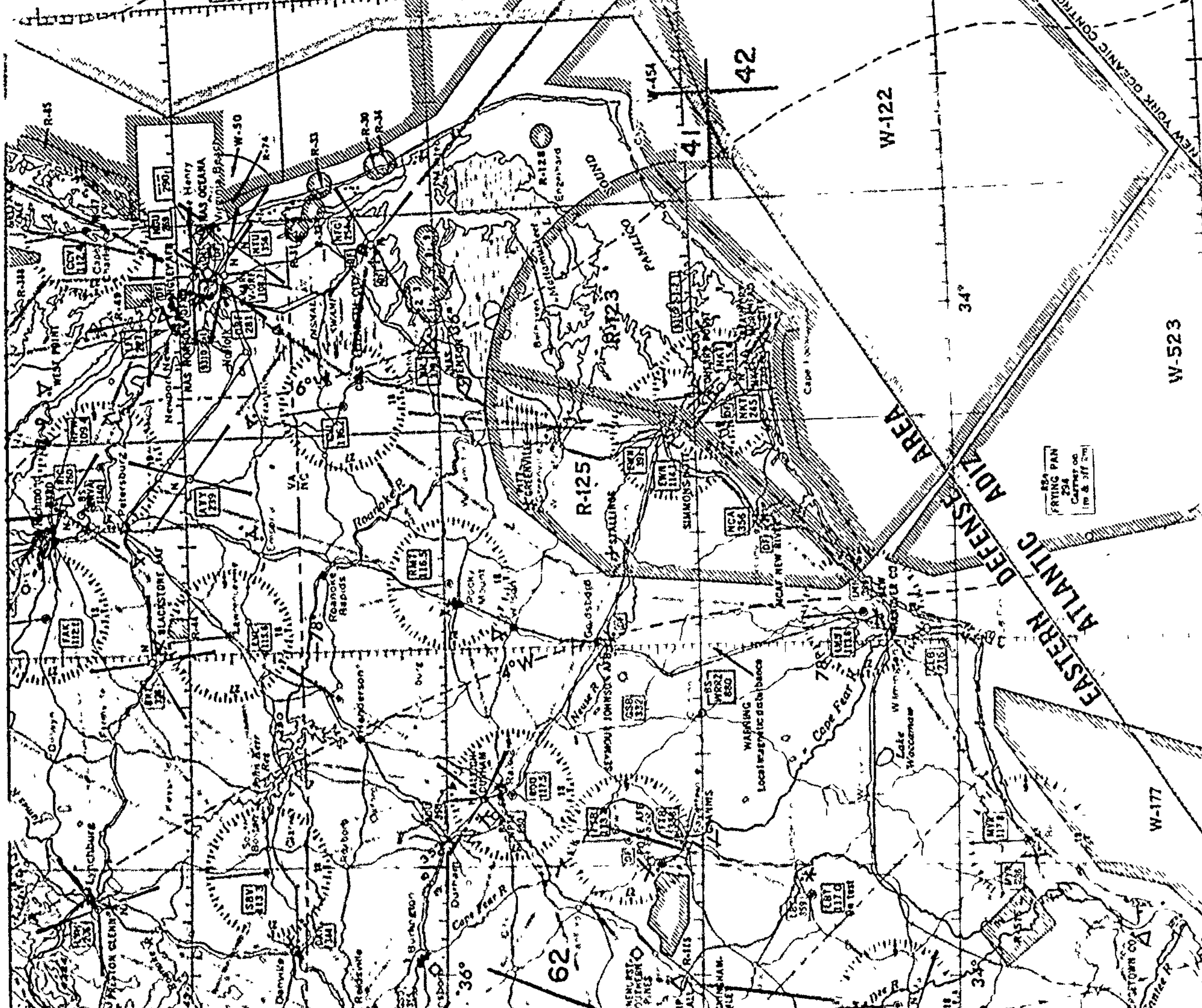

EASTERN ATLANTIC ADIZ AREA
NEW YORK OCEANIC CONTROL
W-122
W-523
W-177
R-125
FRYING PAN

Figure 2

In addition, the aircraft is equipped with aircraft response instrumentation capable of defining linear and angular accelerations, velocities, and displacements as well as structural loads and motions of the primary aerodynamic controls.

Instrumentation is also provided to measure the pilot's environment and reactions.

GUST SENSING

A differential pressure head attached to a rigid boom mounted on the radome nose of the B-66 provides the prime gust imput source. (See figures 3 to 6) The five probe differential pressure sensor furnishes pressures proportional to angle of attack, sideslip, airspeed and static pressure. Accelerometers also are installed in the head assembly for measuring acceleration at the point of sensing the freestream velocities and direction.

The most important single measurement to be made from an aircraft in determination of atmospheric gust intensity is the velocity and direction of the airstream relative to the aircraft as a function of time. The five probe sensor offers several advantages in making this measurement among which are:

1) high gain
2) good stability
3) all measurements required can be made in a small local region undisturbed by aircraft induced flow effects
4) good frequency response
5) known calibration characteristics

The gain of the sensing head in terms of differential pressure output vs. angle of attack input is shown in figure 7.

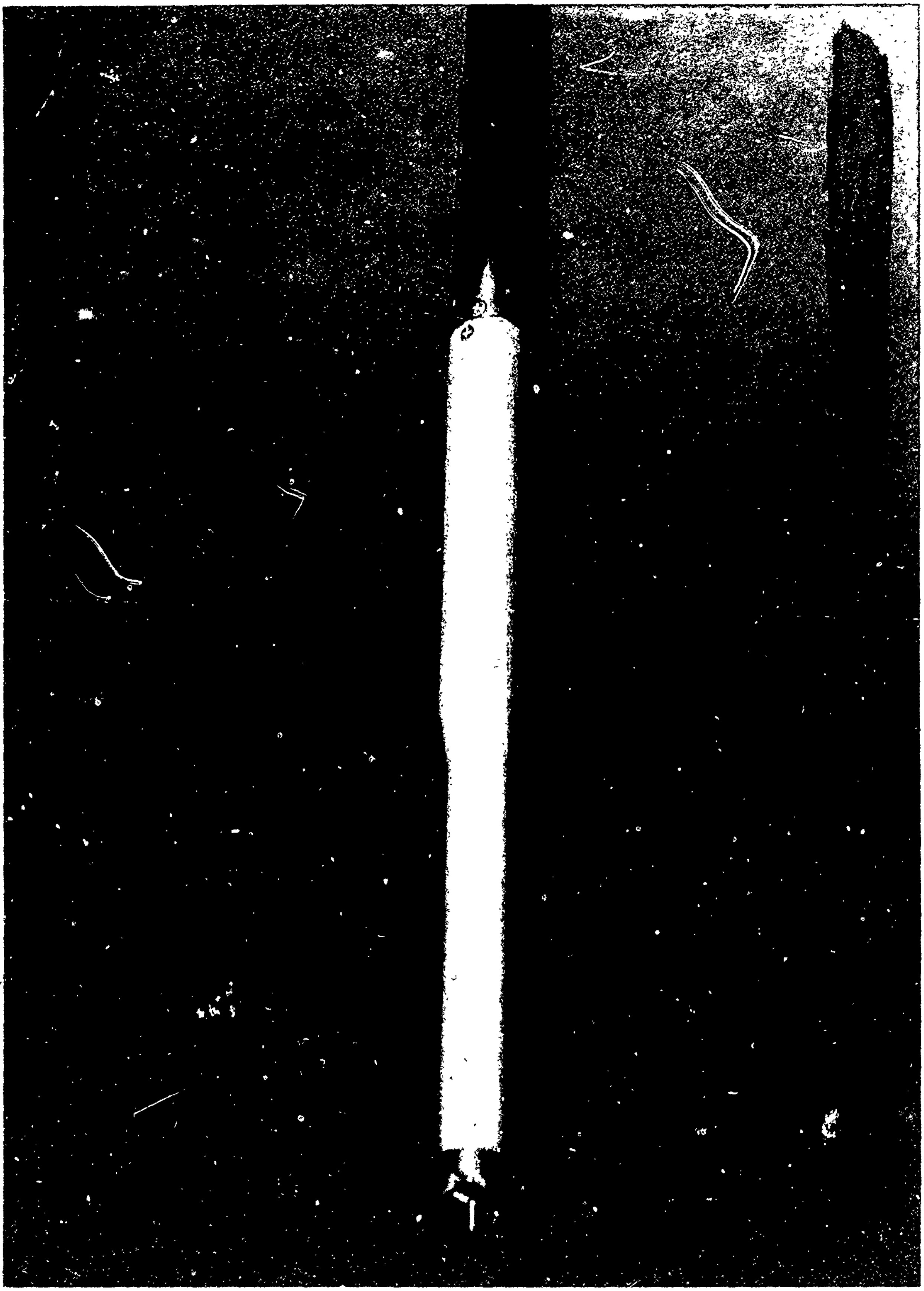


Figure 3

Figure 4

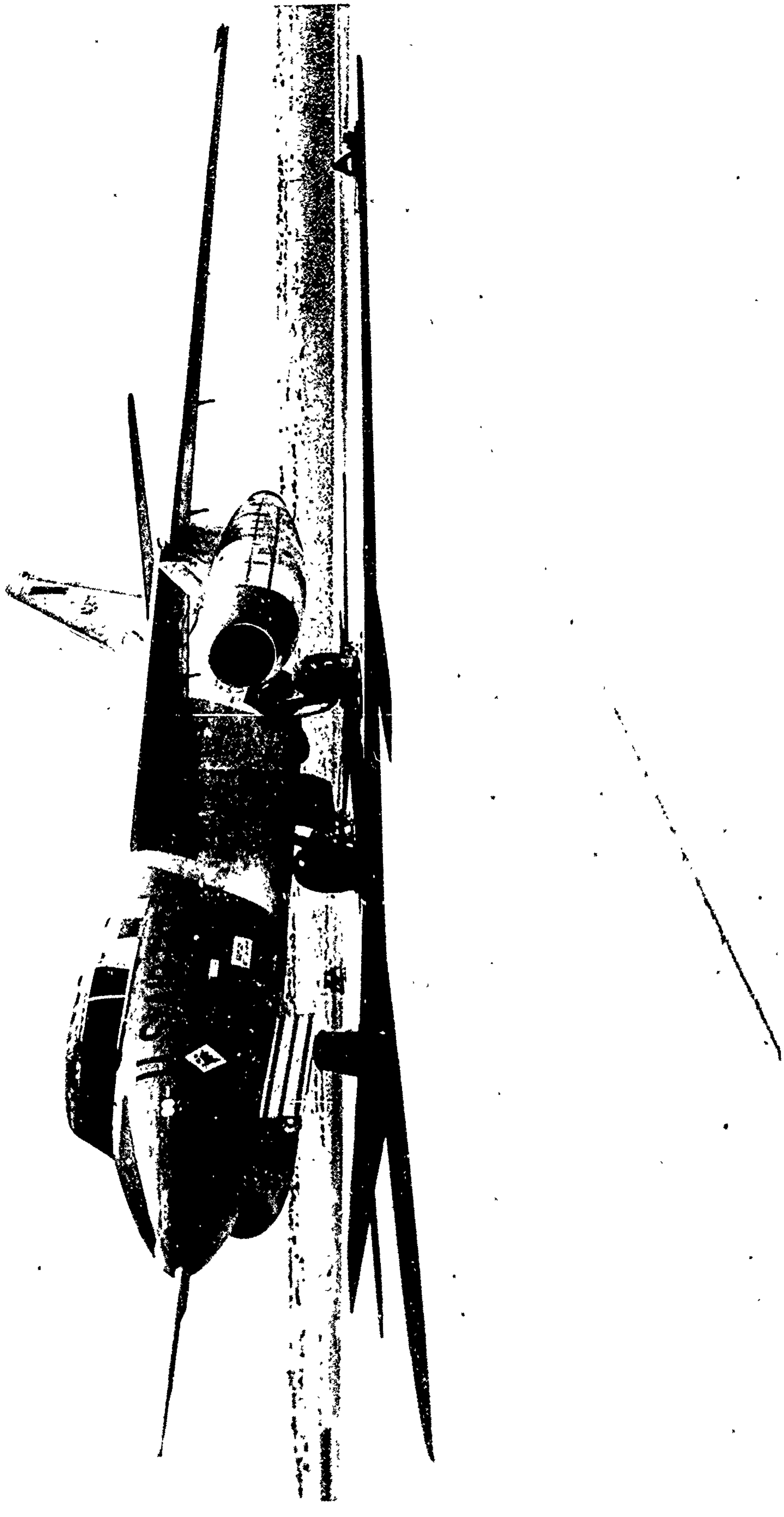


Figure 5

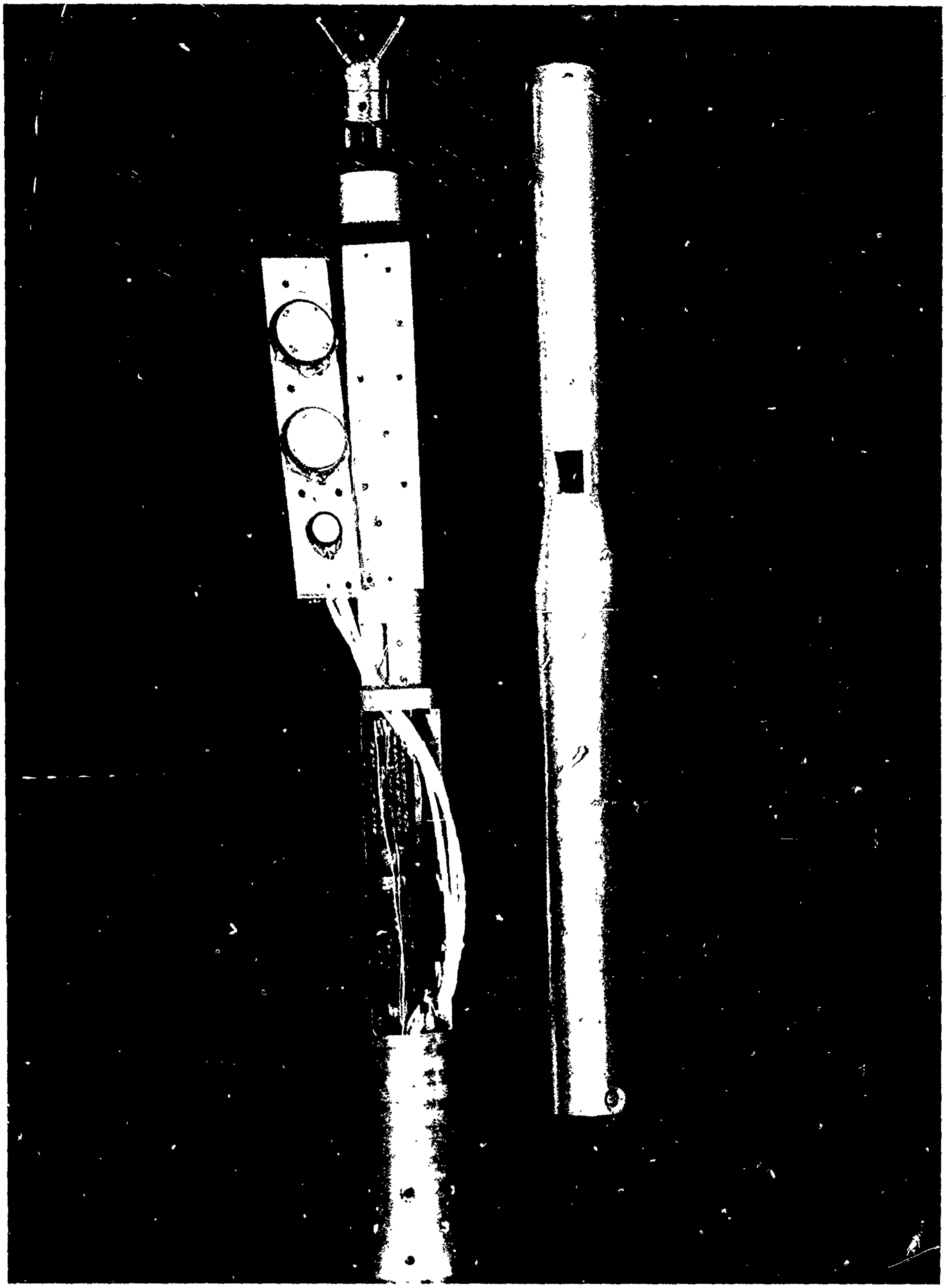


Figure 6

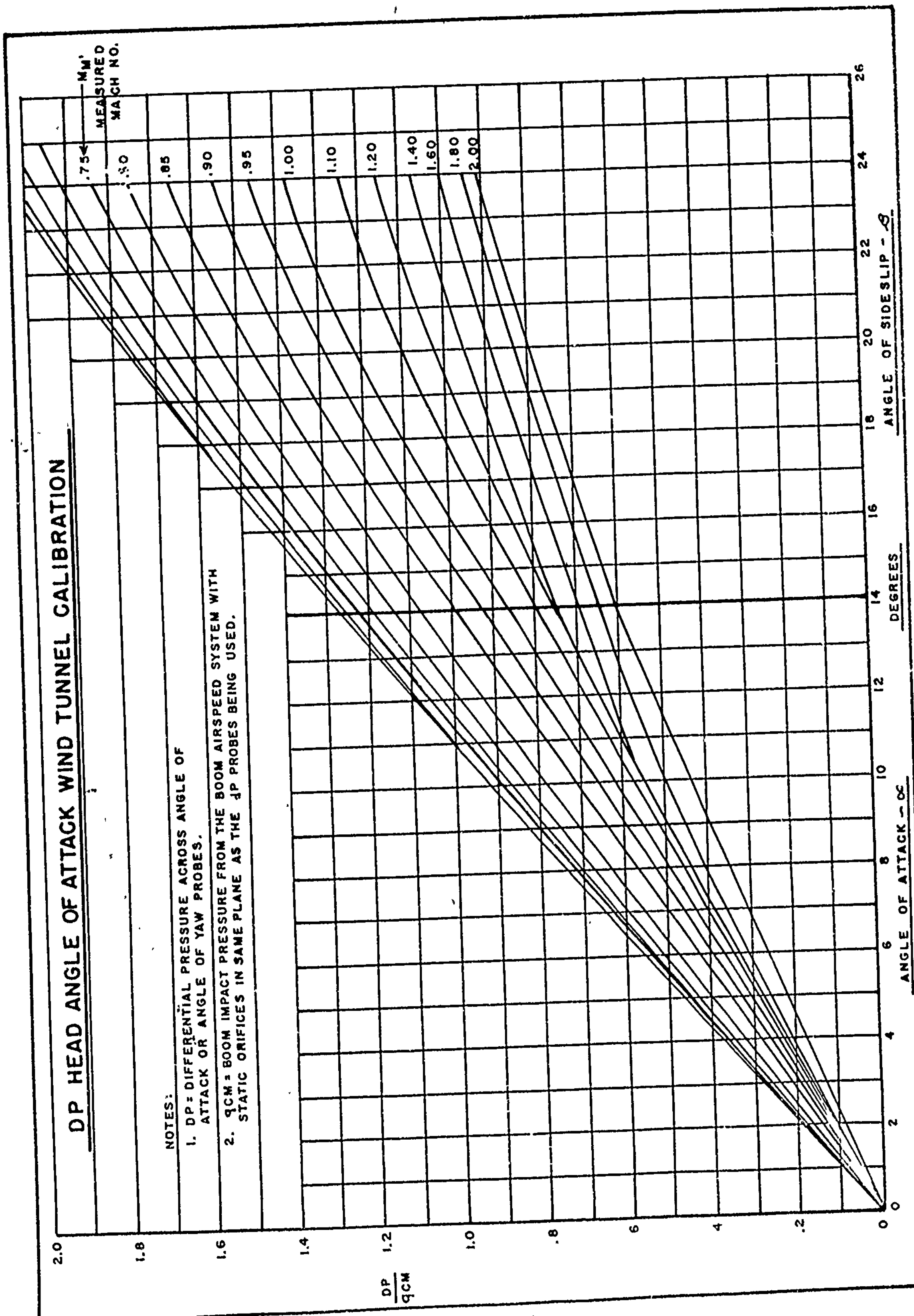

DP HEAD ANGLE OF ATTACK WIND TUNNEL CALIBRATION
NOTES:
1. DP = DIFFERENTIAL PRESSURE ACROSS ANGLE OF ATTACK OR ANGLE OF YAW PROBES.
2. qCM = BOOM IMPACT PRESSURE FROM THE BOOM AIRSPEED SYSTEM WITH STATIC ORIFICES IN SAME PLANE AS THE dP PROBES BEING USED.
MM' MEASURED MACH NO.
.75
.80
.85
.90
.95
1.00
1.10
1.20
1.40
1.60
1.80
2.00
DP/qCM
2.0
1.8
1.6
1.4
1.2
1.0
.8
.6
.4
.2
0
0
2
4
6
8
10
12
14
16
18
20
22
24
26
ANGLE OF ATTACK – α
DEGREES
ANGLE OF SIDESLIP – β

Figure 7

Since there are no moving parts in the head, the stability of the calibration is constant. The physical arrangement of the head allows angle of attack, sideslip, dynamic pressure, static pressure and acceleration to be made within a small region at the end of the nose boom ahead of airplane interference effects as proven by in flight position error determination. The head configuration allows for incorporation of the required pressure transducers and connecting tubing in an arrangement that minimizes pneumatic lag. Burst test designed to check internal piping frequency characteristics evidenced natural frequencies well over 100 cps, and oscillatory tests conducted on a head and boom assembly rotated in pitch through a frequency range determined the effects of high accelerations and aerodynamic lag to be negligible. Acceleration was checked from -7 to 7 g and aerodynamic lag was found to be of the order of .001 second for frequencies below 20 cps. Aerodynamic attenuation of angle of attack was found to be 2% at 15 cps and 4% at 20 cps.

The boom natural bending frequency as installed is 16.5 cps and bending corrections to angle of attack have been found to be negligible as measured by data from a boom mounted strain gage amplifier. Boom translation can be directly determined by the tip accelerometer.

AIRPLANE MOTIONS

When gusts are sensed from a moving platform, such as an airplane, it is necessary to remove from the apparent gust angle increments due to airflow magnitude and angle change resulting from platform rotational and translational deviations from constant speed, straight, undisturbed flight. In the case of the B-66 platform, rotations and translational velocity changes are detected by accelerometers, rate gyros and attitude gyro instruments. It is in this area that the greatest measurement difficulty has been experienced, particularly at the low frequencies characteristic of long wave length gusts. Very sensitive and stable gyros and accelerometers are needed. The accuracy and resolution requirements are such that it has been found most difficult to determine sensor calibration characteristics using commonly available instrument laboratory calibration fixtures and methods. In order to measure these motions

more accurately consideration is being given to the installation of a high quality stable platform in the B-66 from which it will be practical to establish more precisely angular deviations and accelerations for the low frequency spectrum. The table being considered has a characteristic drift rate of less than two degrees per hour. To simplify the data recording and analysis program two survey airspeeds have been selected for use on the program, 280 and 360 knots. The lower of these speeds favors obtaining well defined high frequency or short wave length gusts, and the higher provides a more stable airplane platform and more favorable angle of attack and sideslip gains for defining the low frequency information.

ATMOSPHERIC SAMPLING

Sampling runs have been made at average terrain clearances of 200, 400 and 600 feet. Two independent methods are employed to determine height above the ground employing APN-22 and APN-1 radars. The temperature lapse rate with height, wind speed and direction, and meteorological conditions such as cloud coverage, season, etc. are recorded at the time of the run and for the track locations.

HUMAN FACTORS

One oscillograph and three cameras are used to record pertinent human factors information. Three axis accelerometers are attached to the pilot's seat and to two locations on the pilot's body to determine his motions. The pilot's control forces and the aircraft's control positions are also recorded. Two cameras record the pilot's face and his forward field of view, and a third his head, shoulders and the aircraft control wheel.

DATA RECORDING

A FM magnetic tape system is used for recording the basic gust parameters. Fourteen information items plus a 14.5 KC wow and flutter compensation signal coming from a 0.02% crystal controlled oscillator

are recorded on five magnetic tape tracks of an Ampex Tape Series "800" recorder utilizing Bendix TOR-7 Oscillators. A sub-carrier full scale frequency deviation of 7.5% is used. Sub-carrier frequencies utilized vary from 2.3 to 7.35 KC. Among the items recorded are speed lock and time code. The sensed items recorded and recording arrangement is shown on figure 8.

The airborne FM tape record is calibrated prior to and after flight by adjusting each sub-carrier oscillator to a known balance and calibration condition.

A magnetic tape record of 375 feet length is made of each gust run accumulating approximately 2300 feet of record for each flight. After flight the tape is played back on an Ampex FR 114 tape reproducer driving EMR discriminators equipped with Gaussian output filters which cut off at 45 cps and attenuate at a rate of 48 db/octave before the signals are switched by an Epsco multiplexer. This multiplexer is equipped with a sample and hold feature which takes a one microsecond slice of all functions simultaneously and stores the information on capacitors. The data on the capacitors are then read out sequentially to the analog to digital converter which requires 23 microseconds for each information channel conversion.

The data from the A/D converter are output into a format control and buffer; from the buffer the data are loaded onto tape in IBM 727 form suitable for loading an IBM 704 computer. Once the data is recorded on the IBM tape it may be reduced by a Preliminary Processing Program to a form that will allow further analysis by Computer Analysis Programs to power spectra and other forms useful for data interpretation and presentation.

INSTRUMENTATION CHECKS - ANALOG SYSTEM

Because of the complexity of the data gathering, reduction and analysis programs it was found difficult to pinpoint error sources in the system. To establish that the overall system was operating properly

B-66B GUST PROGRAM
DATA COLLECTING & CONVERTING SCHEMATIC

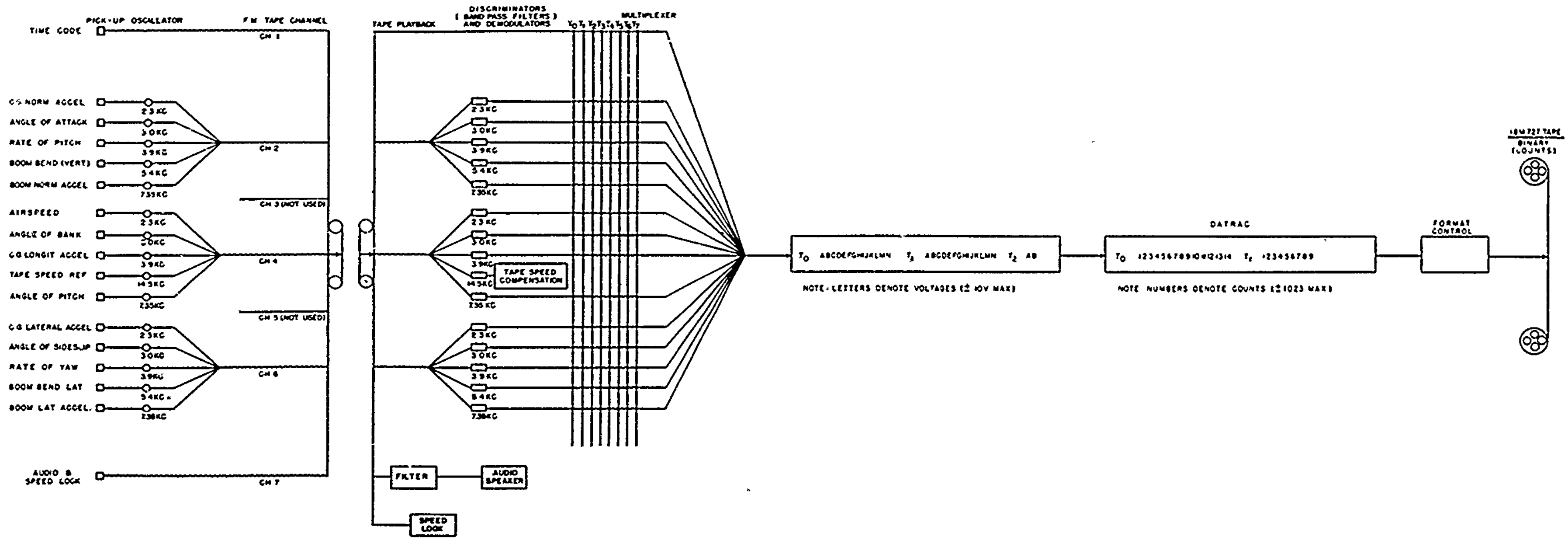


Figure 8

it was decided to insert known signals at various stages and check output. Also it was decided to pin down more exactly sensor characteristics such as,

threshold

linearity

frequency response

cross-coupling effects

noise

Additional thought was given to the frequency range of the measurements as they related to program objectives. In the following discussion some of the detailed checks of the analog recording system and sensors are reviewed.

Boom Angle of Attack - 2.5 psi differential pressure transducer type 82145 (double diaphragm) manufactured by Consolidated Electrodynamics Corp.

1. Burst tests were performed with a Wianco tester to determine transducer and connecting tubing response. The natural frequency was determined to be 320 cps with 0.3 critical damping. Frequency response flat within $\pm 1\%$ to 32 cps. Phase shift 3 degrees at 32 cps.

Airplane Pitch Angle - ± 10 degrees attitude gyro type XJG 7044A35 manufactured by Minneapolis-Honeywell.

1. A static drift check showed from 2 to 3° drift possible during a 5 minute gust run.

2. A linearity calibration showed linearity deviations to be less than 0.1 degrees.

Boom Vertical Acceleration - ±6 g accelerometer type A43-6-350 manufactured by Stathem.

1. Low level, low frequency response tests have been unsuccessful to date because of lack of precision calibration equipment. With existing equipment the attenuation of the accelerometers was found to be less than 3% from 0 to 2 cps.

Airplane Forward Velocity - ±2.5 psi differential pressure transducer type PM231TC-2.5-350 manufactured by C.E.C.

1. Burst tests conducted to establish frequency response of transducer and connecting lines showed the total pressure port to have a natural frequency of 490 cps with a critical damping factor. Flat response within ±1% was 49 cps. Static pressure port checks indicated a natural frequency of 460 cps, 0.5 critical damping, flat response within ±1% to 46 cps and phase shift of 6° at 46 cps.

The above mentioned items constitute the important sensed measurements for determining the vertical gust increment.

FM SYSTEM RESPONSE

The FM recording system elements were checked as follows:

Bendix Sub-carrier Oscillators - (Model TOR-7) and EMR Discriminators (Model 67-D)

1. Drift (at 20°C) was less than 0.2% of the total bandwidth. Five sets of oscillators and discriminators (2.3, 3.0, 3.9, 5.4 and 7.35 KC) were tested for 20 minutes.

Oscillator phase shift and attenuation tests have been performed but results are not complete at this writing.

Discriminator noise tests have been accomplished and noise levels have been determined in terms of counts out of a possible full scale value of 1023 as follows:

minimum	3 counts
maximum	25 counts

To maintain the discriminators at a noise level of 10 counts requires extreme care in the setting up of tape speed compensation - a manual operation for each data conversion run.

A series of tests accomplished on the FM to digital data reduction station have shown that the noise in the overall system including a typical airborne recording is approximately 1% of full scale when wow and flutter compensation are used. This overall check indicated a drift of 0.2% for the ground station. Errors of the airborne system due to common mode noise, non-linearity, gain drift, zero drift, and cross talk have not been determined. Tests for these errors combined with the oscillator frequency response tests are necessary before the overall accuracy of the FM data gathered to date can be established.

The FM system discriminators are equipped with Gaussian type output filter of the same type of frequency cut-off to avoid relative phase shifting between data channels. The following filter requirements were considered:

1) filter must have a cut-off frequency less than the maximum intelligence frequency of the lowest frequency discriminator in order to be compatible with tape speed compensation requirements.

2) the filter must have a rapid cut-off at one-half the digital sampling frequency to prevent fold

back of data.

A partial FM system response test has been performed by inserting square wave signals into a voltage control oscillator directly feeding a discriminator which was connected into the FM digital conversion system. After conversion to IBM digital tape the information was processed by the IBM 704 preliminary data processing and analysis programs to power spectrum form. The spectrum was then compared with the analytical representation of the input. (See figure 9) The check indicated a faithful reproduction of the input. Additional checks of this type involving resistance controlled oscillators and the airborne and ground tape recorder - reproducer system have not been made. The noise contribution due to tape speed errors is not known, although signal amplitude errors are partially corrected by the 14.5 KC tape speed compensation system.

DIGITAL DATA RECORDING SYSTEM

To avoid the possible contamination of the recorded data using a frequency modulated recording system, it has been decided to replace the airborne and ground data recording and reduction FM systems with a digital system. The digital system will prevent contamination possible in drifting of oscillators, discriminators, variations in tape speed and malfunctions of tape speed compensation systems. The digital system will also permit recording of data in flight in a form whose accuracy cannot be destroyed by human operator error as may happen when discriminator and tape speed compensator adjustments are made. The particular variety of digital system proposed will also permit periodic 5 point system calibration as required throughout the flight in addition to the pre- and post flight calibration usually performed. This calibration procedure is of a type that permits compensation for any drifting of transducer power supplies that may be experienced, or for changes in system gain.

DIGITAL SYSTEM DESCRIPTION (See figure 10)

The sensor signal is taken at low level into a balance and normal-

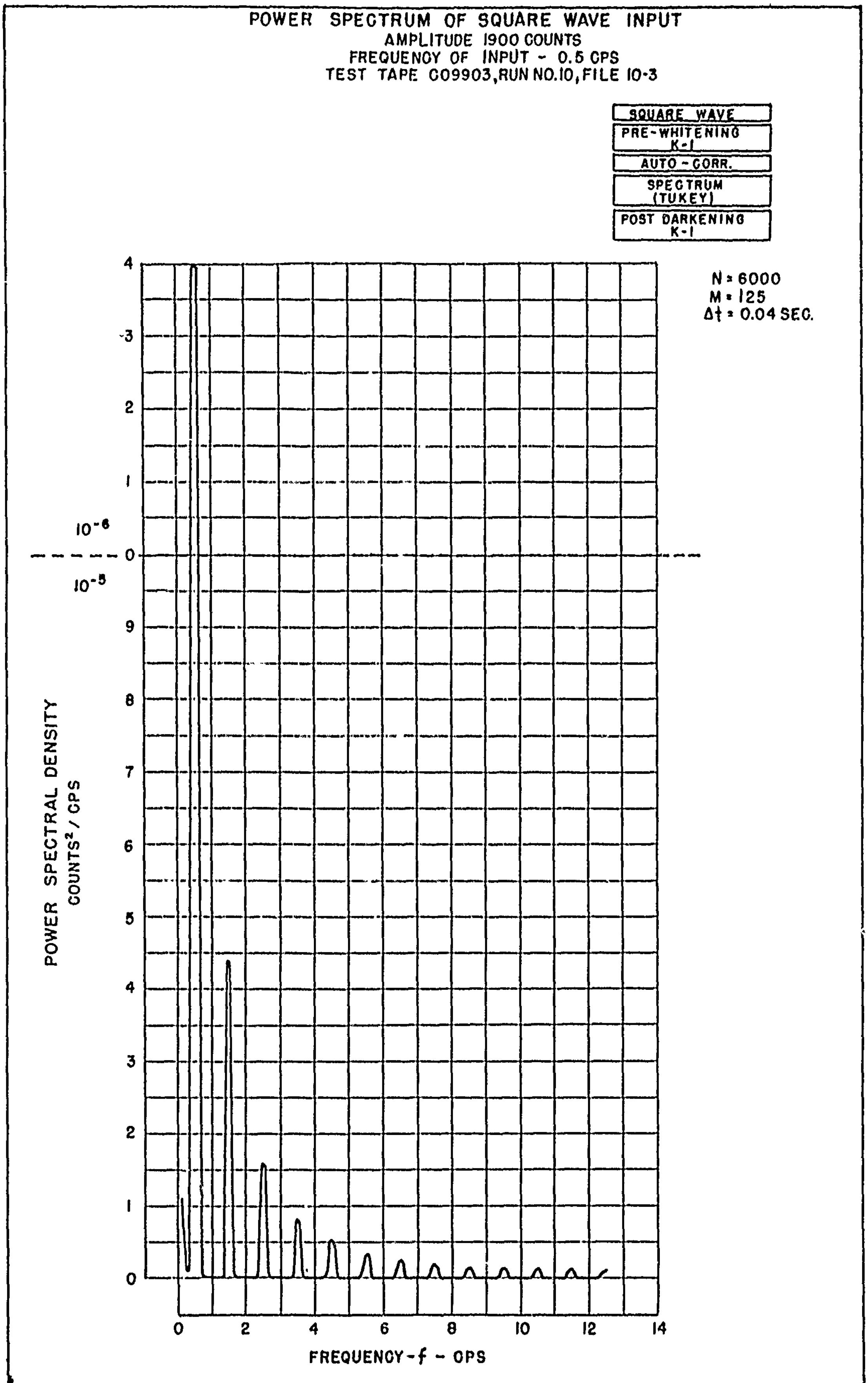
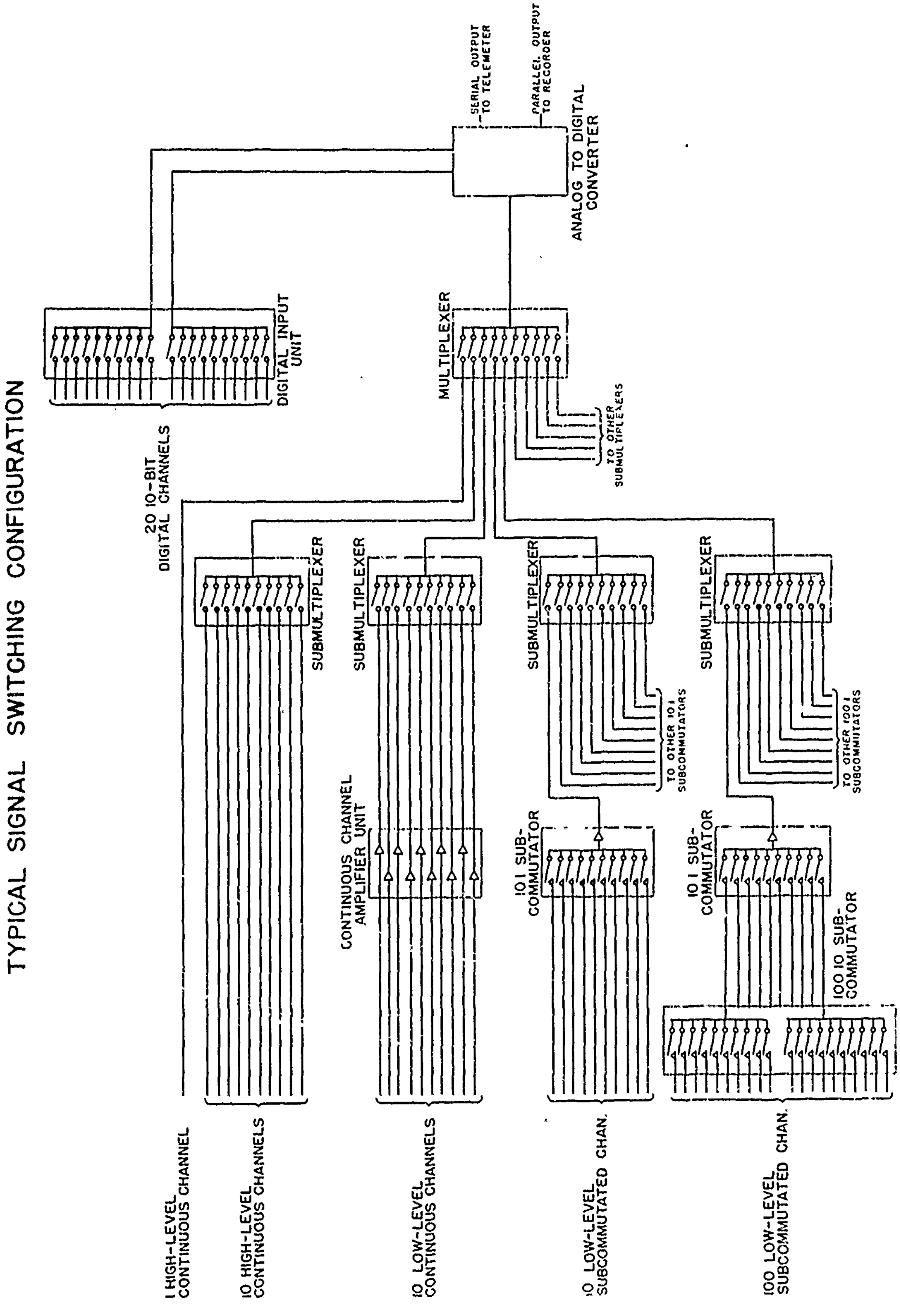

POWER SPECTRUM OF SQUARE WAVE INPUT
AMPLITUDE 1900 COUNTS
FREQUENCY OF INPUT - 0.5 CPS
TEST TAPE C09903, RUN NO. 10, FILE 10-3
SQUARE WAVE
PRE-WHITENING K-1
AUTO-CORR.
SPECTRUM (TUKEY)
POST DARKENING K-1
N = 6000
M = 125
Δt = 0.04 SEC.
10^{-6}
10^{-5}
POWER SPECTRAL DENSITY
COUNTS² / CPS
FREQUENCY - f - CPS
0
1
2
3
4
5
6
7
8
9
10
12
14

Figure 9



TYPICAL SIGNAL SWITCHING CONFIGURATION
SERIAL OUTPUT TO TELEMETER
PARALLEL OUTPUT TO RECORDER
ANALOG TO DIGITAL CONVERTER
DIGITAL INPUT UNIT
20 10-BIT DIGITAL CHANNELS
MULTIPLEXER
TO OTHER SUBMULTIPLEXERS
SUBMULTIPLEXER
SUBMULTIPLEXER
SUBMULTIPLEXER
SUBMULTIPLEXER
TO OTHER 10:1 SUBCOMMUTATORS
TO OTHER 100:1 SUBCOMMUTATORS
CONTINUOUS CHANNEL AMPLIFIER UNIT
10:1 SUB-COMMUTATOR
10:1 SUB-COMMUTATOR
100:10 SUB-COMMUTATOR
1 HIGH-LEVEL CONTINUOUS CHANNEL
10 HIGH-LEVEL CONTINUOUS CHANNELS
10 LOW-LEVEL CONTINUOUS CHANNELS
10 LOW-LEVEL SUBCOMMUTATED CHAN.
100 LOW-LEVEL SUBCOMMUTATED CHAN.

Figure 10

izing network where the full scale signal can be attenuated to +10 MV. The signal is then passed through a filter to a combination DC chopper - A.C. amplifier where the full scale signal level is set to +5 V. (See figure 11) After amplification the signal passes through a multiplexer where the signal is sampled 500 times per second for presentation to an analog-digital converter. From the analog-digital converter the digital signal is recorded on magnetic tape.

The processing of the flight digital tape into IBM 727 tape for 704 computer processing follows a procedure similar in nature to that used for the FM system described earlier, once the FM information has been converted to digital form. However, the ground equipment required is considerably reduced in complexity and number of units since the FM tape processing functions are not required.

A comparison of the number of steps required to convert the sensed data to digital form for the digital (PCM/FM) and FM systems is shown in figure 12. The drastic simplification of the data processing by the digital system is evident.

All data contamination that takes place in the digital system must take place during the processing of the analog signal up to and partially through the analog-digital converter. In recognition of this great care was taken in the techniques and devices used in this end of the system.

In the airborne digital system a sample and hold stage is not used since each information channel is sampled 500 times per second. The elimination of this device avoids an error source estimated at 0.2% from capacitor leakage and additional switching noise.

To avoid sampling noise a filter is combined with each input amplifier which attenuates approximately 45 db at 250 cps (the Nyquist frequency). The attenuation and phase shift characteristics versus frequency are shown in figures 13 and 14.

CONTINUOUS CHANNEL AMPLIFIER

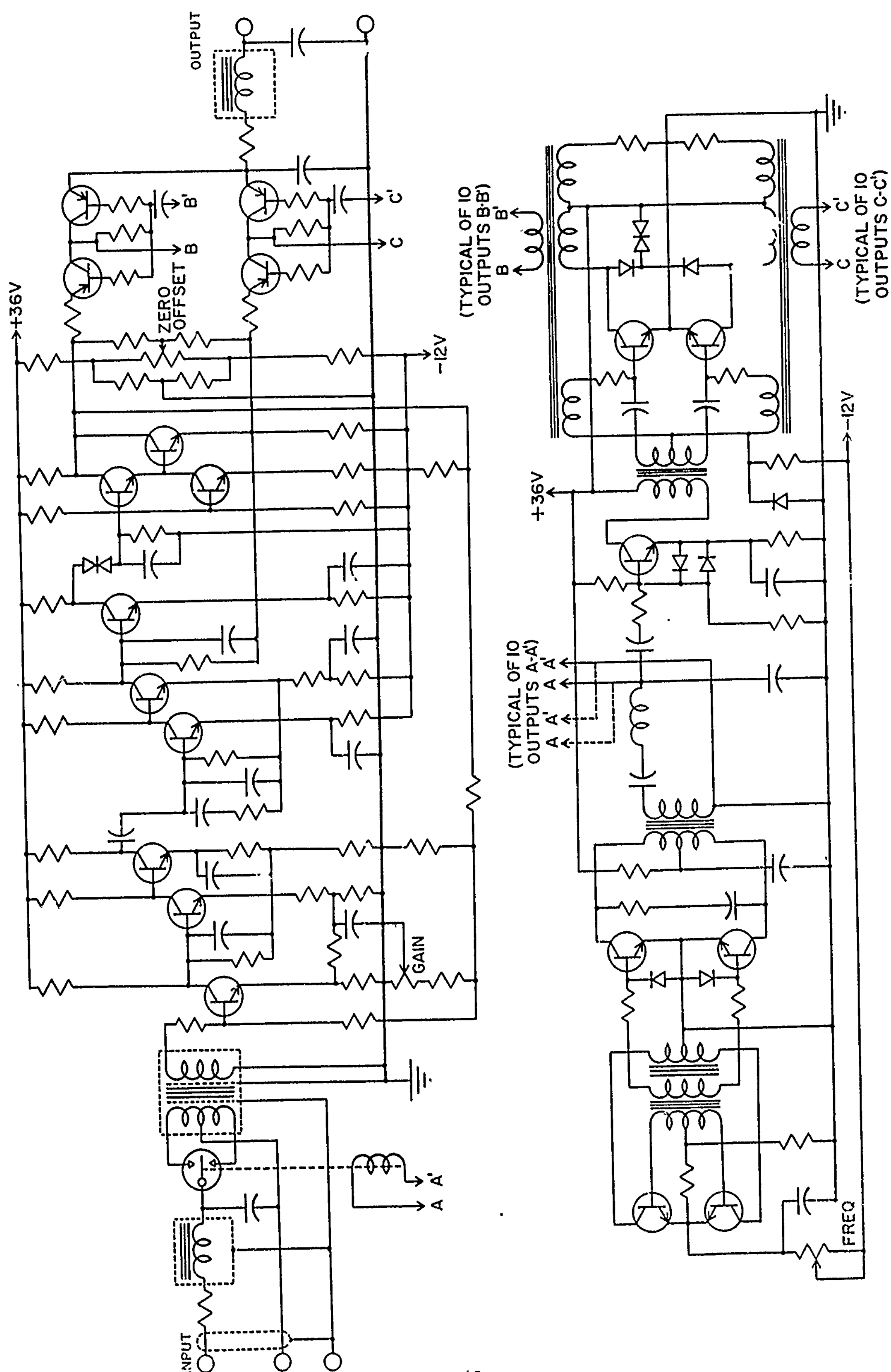


Figure 11

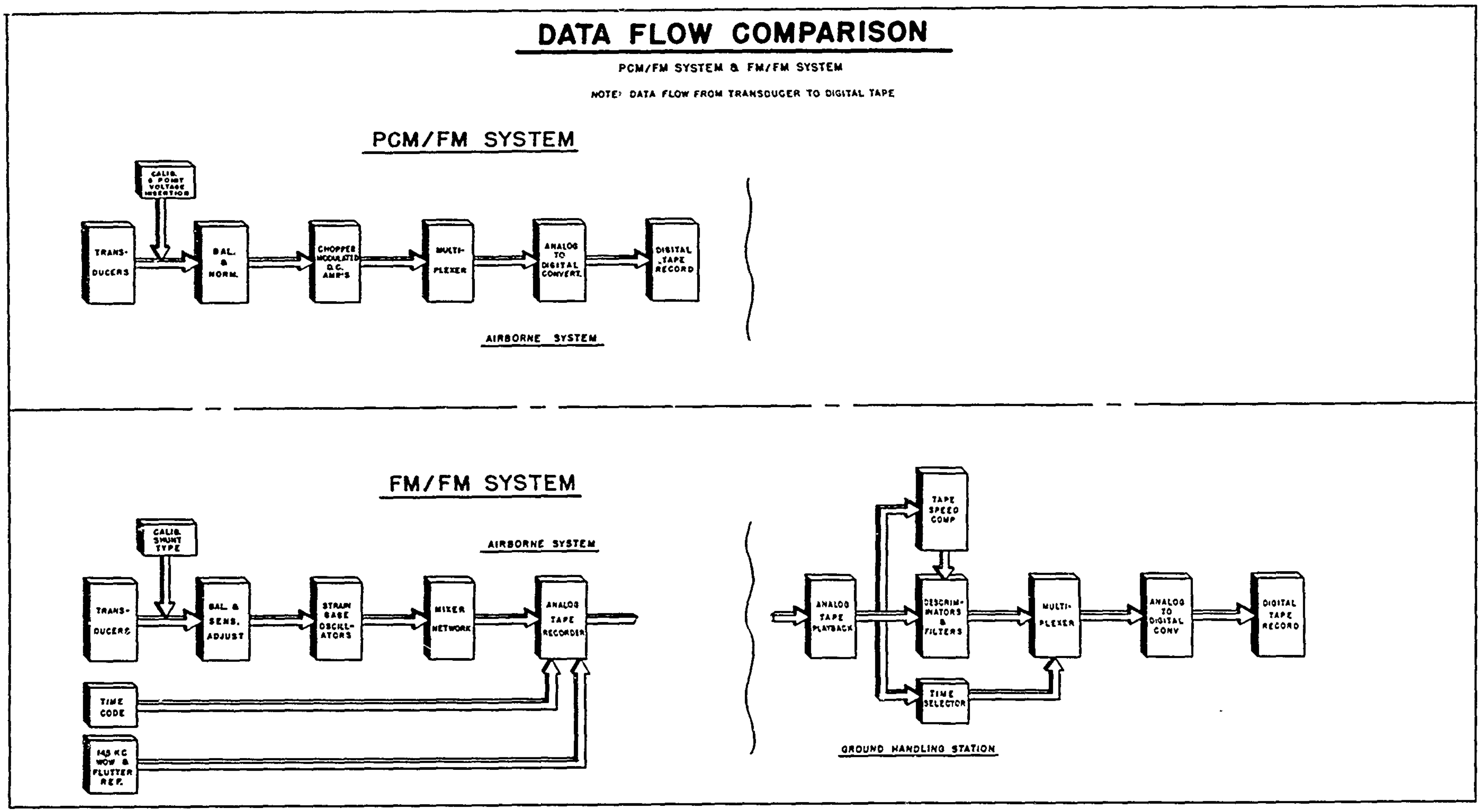

DATA FLOW COMPARISON
PCM/FM SYSTEM & FM/FM SYSTEM
NOTE: DATA FLOW FROM TRANSDUCER TO DIGITAL TAPE
PCM/FM SYSTEM
CALIB. & POINT VOLTAGE INSERTION
TRANS-DUCERS
BAL. & NORM.
CHOPPER MODULATED D.C. AMPS
MULTI-PLEXER
ANALOG TO DIGITAL CONVERT.
DIGITAL TAPE RECORD
AIRBORNE SYSTEM
FM/FM SYSTEM
AIRBORNE SYSTEM
CALIB. SHUNT TYPE
TRANS-DUCERS
BAL. & SENS. ADJUST
STRAIN GAGE OSCILL-ATORS
MIXER NETWORK
ANALOG TAPE RECORDER
TIME CODE
145 KC WOW & FLUTTER REF.
ANALOG TAPE PLAYBACK
TAPE SPEED COMP
DESCRIM-INATORS & FILTERS
TIME SELECTOR
MULTI-PLEXER
ANALOG TO DIGITAL CONV
DIGITAL TAPE RECORD
GROUND HANDLING STATION

Figure 12

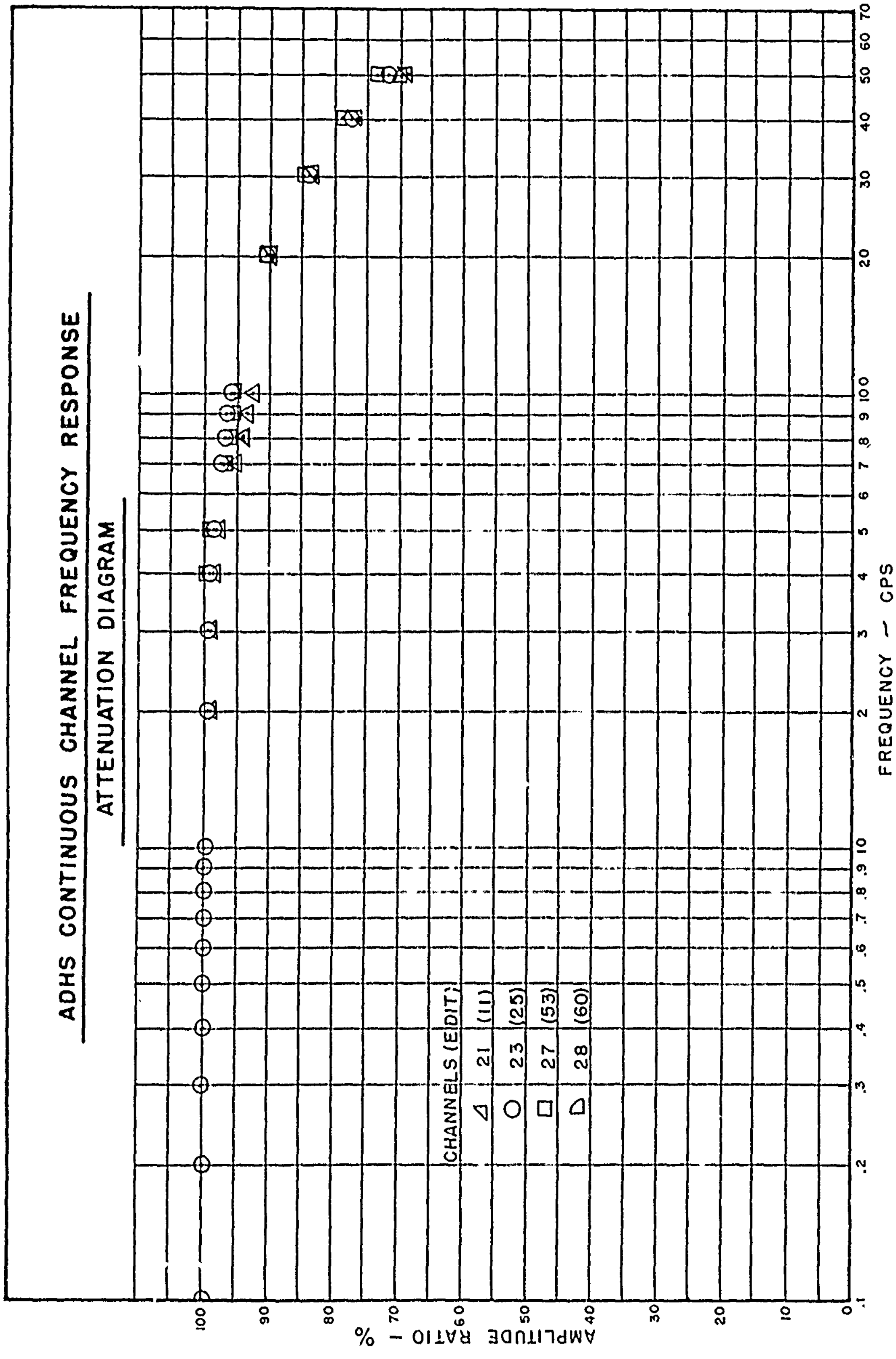
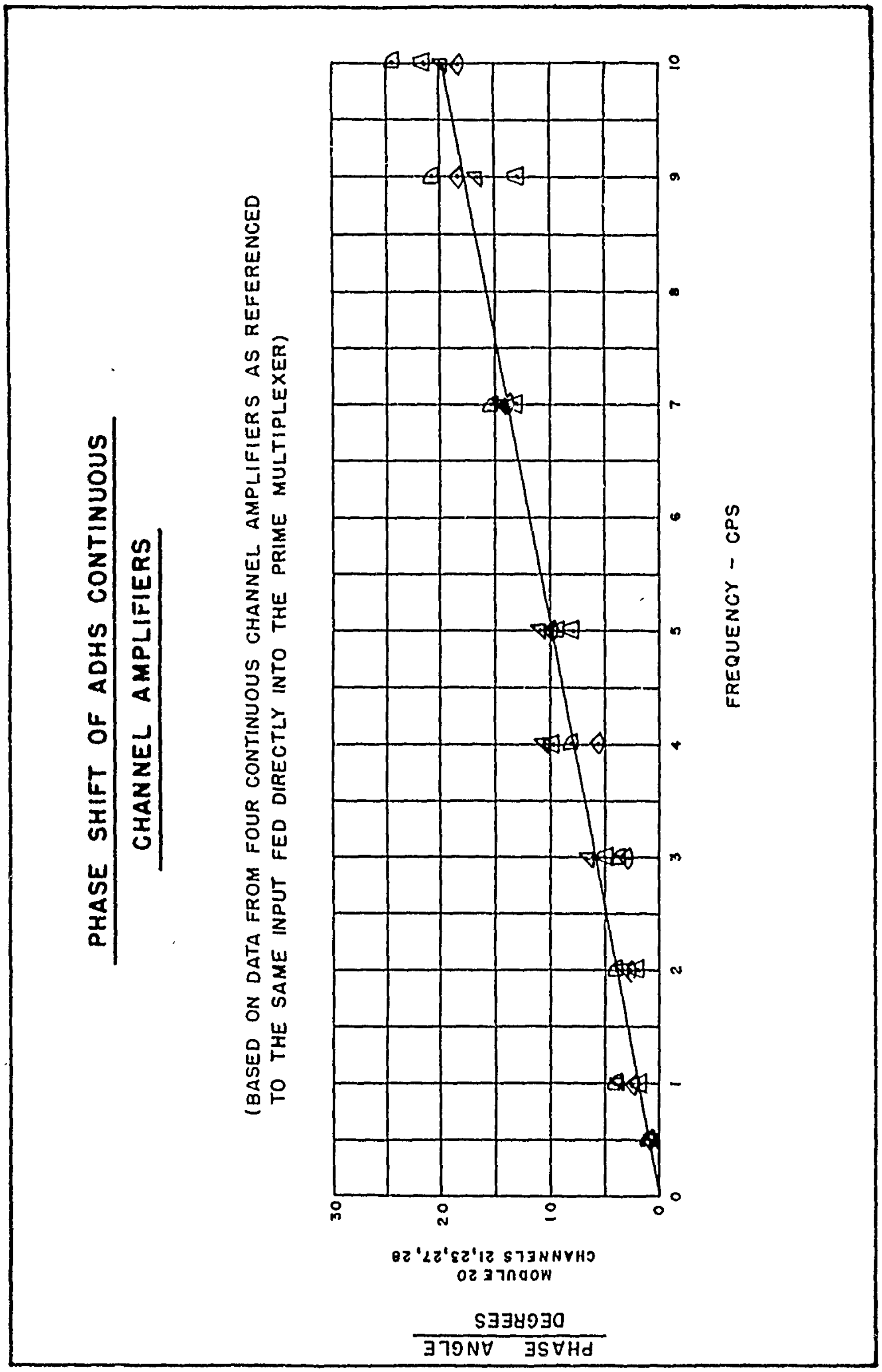

ADHS CONTINUOUS CHANNEL FREQUENCY RESPONSE
ATTENUATION DIAGRAM
CHANNELS (EDIT)
21 (11)
23 (25)
27 (53)
28 (60)
AMPLITUDE RATIO – %
100
90
80
70
60
50
40
30
20
10
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
10
2
3
4
5
6
7
8
9
100
20
30
40
50
60
70
FREQUENCY – CPS

Figure 13



PHASE SHIFT OF ADHS CONTINUOUS CHANNEL AMPLIFIERS
(BASED ON DATA FROM FOUR CONTINUOUS CHANNEL AMPLIFIERS AS REFERENCED TO THE SAME INPUT FED DIRECTLY INTO THE PRIME MULTIPLEXER)
PHASE ANGLE
DEGREES
MODULE 20
CHANNELS 21,23,27,28
0
10
20
30
0
1
2
3
4
5
6
7
8
9
10
FREQUENCY - CPS

Figure 14

Another filter will be used before switching during the gust program since most data of interest are below 10 cps and it is intended to edit the data tape at 25 times per second. To prevent contamination of data above the Nyquist frequency of 12.5 cps a filter will be used with an attenuation of 25 db at $12\frac{1}{2}$ cps. A linear filter which starts to roll off at 8 cps and falls off at 42 db per octave will meet this requirement.

Another insidious source of error in data recording is that due to timing. In the FM system time is controlled by speed lock which allows the playback tape recorder to be compensated for time shifts in the flight recorded tape up to one cycle per second frequency. Otherwise the lapse of flight time is not preserved throughout the system. The multiplexer of the ground digital conversion driven by a crystal controlled oscillator generates the time interval which subsequently appears on the IBM converted data. Any difference in time between the FM tape playback and the multiplexer oscillator will be reflected as an uncorrected data frequency distortion. The use of the time code generator 1 second, 1/10th second and 1/1000th second time pulses permit a qualitative indication of time phase errors but no automatic correction procedure using this information has been employed to date.

In the digital airborne system a precision temperature controlled crystal acts as the basic source of time for the program control which in turn controls the time code generator, multiplexer and analog-digital converter. Each tape recorded word of a data frame is put down in a precisely established timed sequence and two data words of each frame from the time code generator are used for time identification. Therefore, in the case of the digital system, the edited IBM data can recover the flight recorded data in exact timed sequence and also recover the time identification of each frame as written in flight. No intermediate processing time variations, as for example tape recorder or reproducer speed variations, can have any effect on the data time accuracy.

INSTRUMENTATION CHECKS - DIGITAL SYSTEM

Extensive laboratory and flight accuracy and reliability tests have been made on the digital system proposed for the gust program. The error sources of prime importance to this program are summarized and evaluated in the following discussion:

1. Repeatability error: A stability error and a measure of the system's ability to maintain a constant output for a constant input. This error was statistically determined from 29,881 measurements made during flight using six prime channels. The mean error determined was 2 counts with a range from 1 to 4 counts.

2. Linearity: The system linearity of output to input over full scale range was 2.2 counts.

3. Gain and zero adjustment drift: Zero drifted 1.6 counts in two hours; Gain drifted less than 1 count in two hours.

4. Cross talk: No cross talk was measureable in prime channels that will be used in the gust programs. Cross talk was checked by driving one input channel to full scale at frequencies from 1 to 100 cps while monitoring an adjacent channel.

5. Common mode: Common mode rejection tests were performed with D.C., 90 cps, and 400 cps A.C. common mode voltages at 0.5, 1, 5, and 10 volts level. No error was evident for D.C. voltage; at 400 cps 1 count was determined, and at 90 cps 2 counts were determined for 5 volts.

6. Frequency response: Frequency response was checked from 0 to 50 cps by driving several input channels with a common signal generator. Sinusoidal and ramp function wave shapes were used, the ramp being primarily useful to

detect phase shift and the sinusoid to determine amplitude ratio of input to output. Figures 13 and 14 show the amplitude variation and phase shift characteristics of interest. The attenuation of the system is 5% at 10 cps and the phase shift is linear from 0 to 10 cps. Typical time histories of checks at 1 and 10 cps are shown in figures 15 and 16. By means of these types of curves and computed digital data it has been determined that reproducibility of results channel to channel is within 5 degrees in phase angle and 3% in amplitude ratio through 10 cps.

If an R.M.S. sum of the errors listed above are made the overall digital system accuracy appears as follow:

$$\text{System Error} = \sqrt{\sum(\text{Characteristic Errors})^2}$$

Characteristic Errors:

Repeatability	=	2.0 counts
Linearity	=	2.2 counts
Zero Stability	=	1.6 counts
Gain Stability	=	1.0 counts
Cross Talk	=	0.0 counts
Common Mode	(5 Volts)	
400 cps	=	1.0 counts
90 cps	=	2.0 counts

System Error = 4.17 counts

or 0.42%

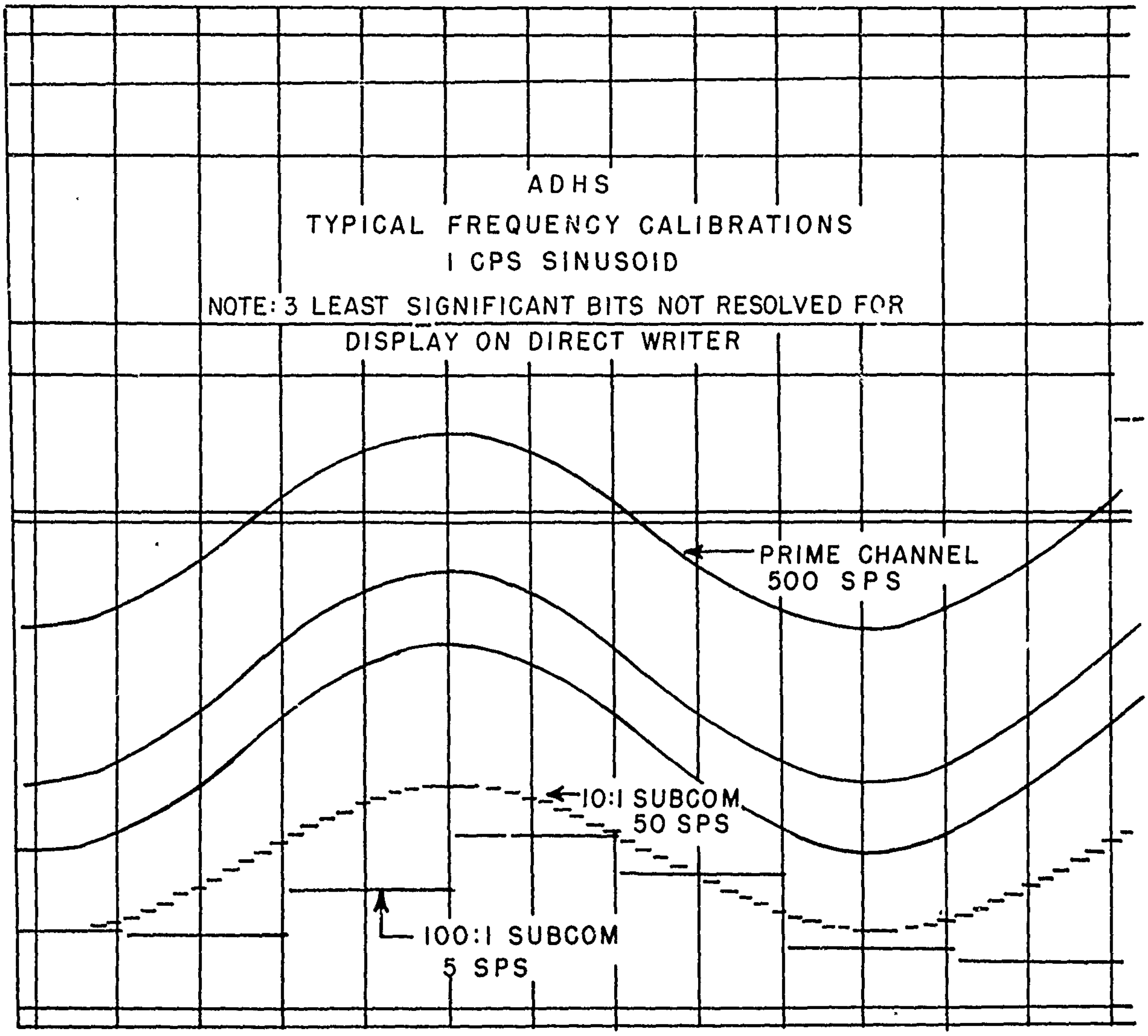

ADHS
TYPICAL FREQUENCY CALIBRATIONS
1 CPS SINUSOID
NOTE: 3 LEAST SIGNIFICANT BITS NOT RESOLVED FOR
DISPLAY ON DIRECT WRITER
PRIME CHANNEL
500 SPS
10:1 SUBCOM
50 SPS
100:1 SUBCOM
5 SPS

Figure 15

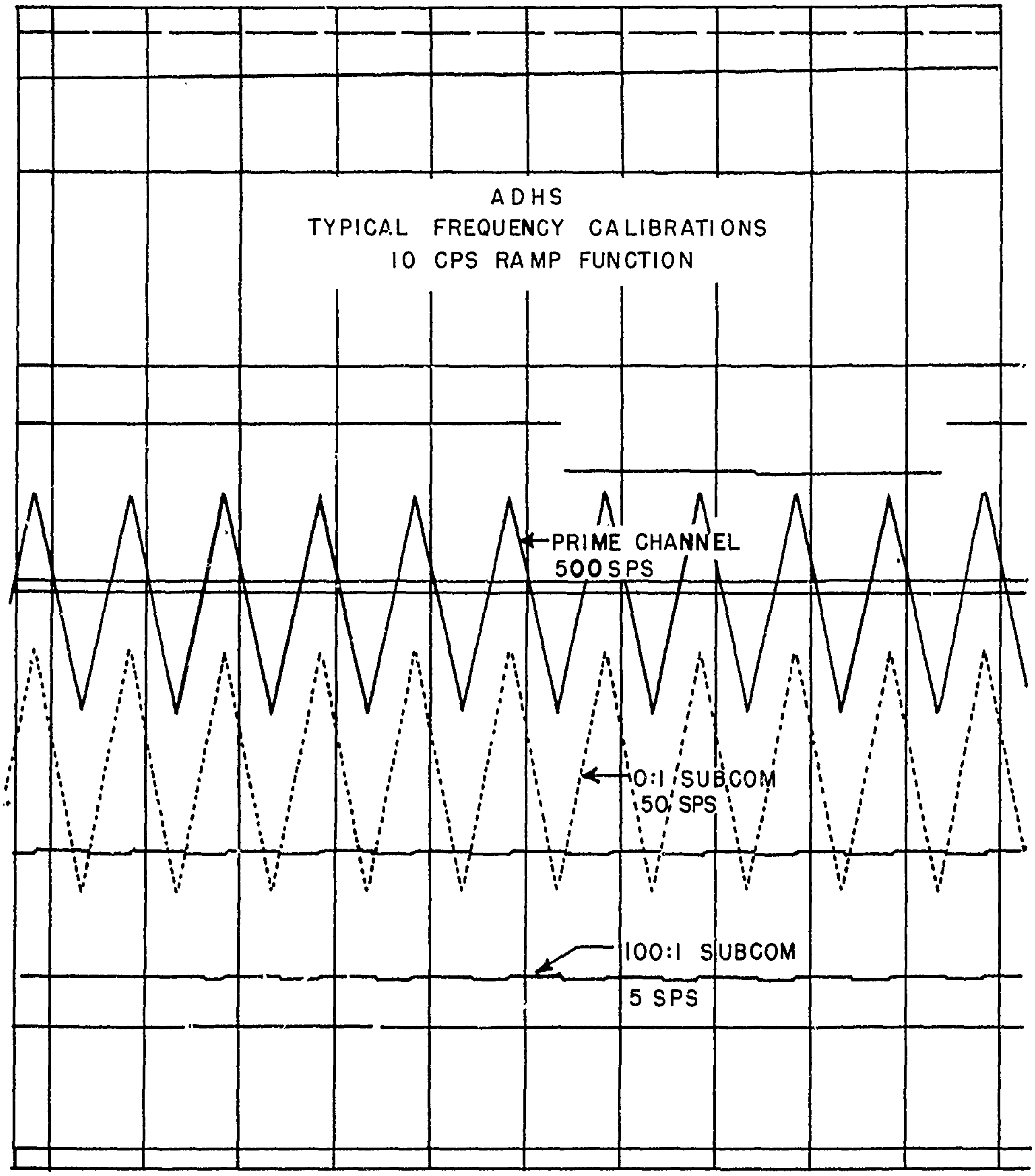

ADHS
TYPICAL FREQUENCY CALIBRATIONS
10 CPS RAMP FUNCTION
PRIME CHANNEL
500 SPS
10:1 SUBCOM
50 SPS
100:1 SUBCOM
5 SPS

Figure 16

COMPARISON OF EFFECTS OF ERRORS

As stated previously 10 counts of noise can be achieved with the FM system if great care is taken with discriminator and tape speed compensation adjustment. The expected total error of the digital system appears to be slightly in excess of 4 counts. The effects of this error or "noise" in the measurement system can be illustrated by examing the effects on the vertical gust power spectrum of recording noise as it affects the pitch and angle of attack terms contribution to the spectrum. 20 counts of noise in the pitch term produces a power three times as great as the vertical gust power. 2 counts of noise produces a power between 1/100 and 1/1000 of the vertical gust power. 20 counts of noise in the angle of attack term produces a noise approximately 1/10 of the value of the gust power, and 2 counts of noise produces a power between 1/1000 and 1/10000 the gust power. Because of this the importance of maintaining recording system noise power at low levels is evident.

NEW SENSORY INSTRUMENTATION

When the digital system is installed in the B-66 it is intended to change the sensors as shown by figures 17 and 18. Additional changes in the sensor instrumentation are planned if it is found that the stable platform can be adapted to the aircraft. From the changes shown improvement in sensor range and accuracy is evident for the important gust parameters.

PRELIMINARY PROCESSING PROGRAM

A graphical flow diagram of the preliminary processing program is shown in figure 19 for the FM and Digital data handling system. One important difference between the two programs is the calibration procedure. The ADHS program permits a five point calibration permitting zero adjustment, gain adjustment, and linearization to be performed as needed throughout the flight. The FM calibration is made before or after flight and is made only at two conditions, zero and in the range of 50 to 80 percent of full scale.

GUST SURVEY INSTRUMENTATION

CHAN NO.	MEASUREMENT	MANUFACTURER	INSTRUMENT RANGE	FM/FM RANGE USED	NOISE LEVEL ±10cts	INSTR. LOCATION	INSTRUMENT ACCURACY
1	Angle of Attack	C.E.C.	±2.5PSID	1.1PSID	±.011PSID	Boom Head	1%
2	Pitch Attitude	Minn-Honey	±85°	±10°	±.1°	Bomb Bay	.1°
3	Nose Vert Accel	Statham	N.A.	N.A.	N.A.	N.A.	N.A.
4	Angle of Sideslip	C.E.C.	±2.5PSID	1.2PSID	±.012PSID	Boom Head	1%
5	Yaw Attitude	N.A.	N.A.	N.A.	N.A.	Bomb Bay	N.A.
6	Nose Lat Accel	Statham	N.A.	N.A.	N.A.	N.A.	N.A.
7	Roll Attitude	Minn-Honey	±180°	Oscil	Oscil	Bomb Bay	.1°
8	Nose Long Accel	Statham	N.A.	N.A.	N.A.	N.A.	N.A.
9	Airspeed	C.E.C.	±2.5PSID	1.6PSID	±.016PSID	Boom Head	1%
10	CG Vert Accel	Statham	±2 g	1.5 g	±.015 g	Bomb Bay	1% (.02 g)
11	CG Lat Accel	Statham	±.6 g	±.4 g	±.004 g	Bomb Bay	1% (.006 g)
12	Pitch Rate	Giannini	±15°/sec	±10°/sec	±.1°/sec	Radome	1%
13	Yaw Rate	Giannini	±15°/sec	±11°/sec	±.1°/sec	Radome	1%
14	Roll Rate	Giannini	±15°/sec	Oscil	Oscil	Radome	1%
15	Boom Vert Accel	Statham	±6 g	±4.5 g	±.05 g	Boom Head	1%
16	Boom Lat Accel	Statham	±6 g	±6 g	±.06 g	Boom Head	1%
17	O. A. T.	D.A.C.	----	Oscil	Oscil	Bottom of A/C	1%
18	Pressure Alt (Alt)	Statham	0 - 15PSIA	Oscil	Oscil	Radome	.1%
19	Diff Pres Alt	Statham	±1 PSID	Oscil	Oscil	Radome	1%
20	Grd Clearance	Sylvania	0 - 1000Ft	Oscil	Oscil		5%

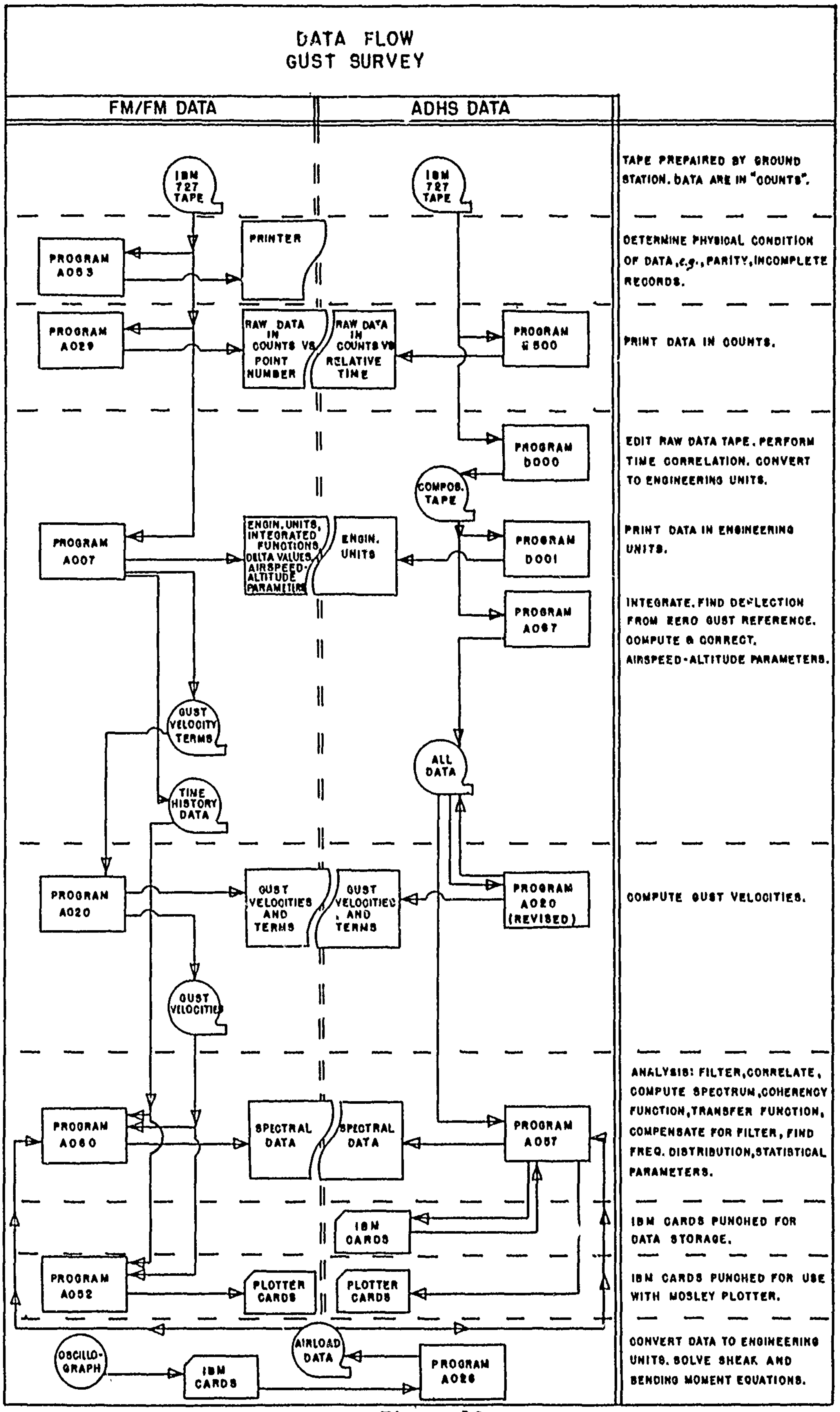
N.A. - Not Available

Figure 17

GUST SURVEY ADHS INSTRUMENTATION

CHAN NO.	MEASUREMENT	MANUFACTURER	INSTRUMENT RANGE	RANGE USED (Full Scale ADHS)	NOISE LEVEL ± 2 cts	INSTR. LOCATION	INSTRUMENT ACCURACY
1	Angle of Attack	C.E.C.	± 2.5PSID	± 1 PSID	$\pm .002$PSID	Boom Head	1% (.025 PSID)
2	Pitch Attitude	Lear or Kearfott	$\pm 85^{\circ}$	$\pm 5^{\circ}$	$\pm .010^{\circ}$	Radome	$.05^{\circ}$ ($.05^{\circ}$)
3	Nose Vert Accel	Donner	± 2 1/2 g	$\pm .8$ g	$\pm .0016$ g	Radome	.05% (.00125 g)
4	Angle of Sideslip	C.E.C.	± 2.5PSID	± 1.0 PSID	$\pm .002$PSID	Boom Head	1% (.025 PSID)
5	Yaw Attitude	Lear or Kearfott	0 - 360	$\pm 5^{\circ}$	$\pm .010^{\circ}$	Radome	$.05^{\circ}$ ($.05^{\circ}$)
6	Nose Lat Accel	Donner	± 1 g	$\pm .8$ g	$\pm .0016$ g	Radome	.05% (.0005 g)
7	Roll Attitude	Lear or Kearfott	$\pm 180^{\circ}$	$\pm 15^{\circ}$	$\pm .030^{\circ}$	Radome	$.05^{\circ}$ ($.05^{\circ}$)
8	Nose Long Accel	Donner	± 1 g	$\pm .5$ g	$\pm .001$ g	Radome	.05% (.0005 g)
9	Airspeed	C.E.C.	± 2.5PSID	0 - 3.3PSID	$\pm .007$PSID	Boom Head	1% (.025 PSID)
10	CG Vert Accel	Donner	± 2 1/2 g	$\pm .5$ g	$\pm .001$ g	Bomb Bay	.05% (.00125 g)
11	CG Lat Accel	Donner	± 1 g	$\pm .5$ g	$\pm .001$ g	Bomb Bay	.05% (.0005 g)
12	Pitch Rate	Giannini	$\pm 15^{\circ}$/sec	$\pm 5^{\circ}$/sec	$\pm .010^{\circ}$	Radome	1% ($.15^{\circ}$/sec)
13	Yaw Rate	Giannini	$\pm 15^{\circ}$/sec	$\pm 5^{\circ}$/sec	$\pm .010^{\circ}$	Radome	1% ($.15^{\circ}$/sec)
14	Roll Rate	Giannini	$\pm 15^{\circ}$/sec	$\pm 10^{\circ}$/sec	$\pm .02^{\circ}$/sec	Radome	1% ($.15^{\circ}$/sec)
15	Boom Vert Accel	Statham	± 6 g	± 5 g	$\pm .010^{\circ}$	Boom Head	1% (.06 g)
16	Boom Lat Accel	Statham	± 6 g	± 5 g	$\pm .010^{\circ}$	Boom Head	1% (.06 g)
17	O. A. T.	D. A. C.	-----	1 - 100°F	$\pm .2^{\circ}$	Bottom of A/C	1% ($\approx 2^{\circ}$F)
18	Press. Altitude (alt)	Statham	0 - 15 PSIA	15 PSIA - 8.3	$\pm .013$PSIA	Radome	.1% (.015 PSIA)
19	Diff Pres Alt	Statham	± 1 PSID	$\pm .5$ PSID	$\pm .002$ PSID	Radome	1% (.01 PSID)
20	Grd Clearance	Sylvania	0 - 1000 FT	0 - 1000 FT	± 2 FT	-----	5% (50 FT)

Figure 18



DATA FLOW
GUST SURVEY
FM/FM DATA
ADHS DATA
IBM 727 TAPE
IBM 727 TAPE
TAPE PREPAIRED BY GROUND STATION. DATA ARE IN "COUNTS".
PROGRAM A053
PRINTER
DETERMINE PHYSICAL CONDITION OF DATA, e.g., PARITY, INCOMPLETE RECORDS.
PROGRAM A029
RAW DATA IN COUNTS VS POINT NUMBER
RAW DATA IN COUNTS VS RELATIVE TIME
PROGRAM G500
PRINT DATA IN COUNTS.
PROGRAM D000
EDIT RAW DATA TAPE. PERFORM TIME CORRELATION. CONVERT TO ENGINEERING UNITS.
COMPOS. TAPE
PROGRAM A007
ENGIN. UNITS, INTEGRATED FUNCTIONS DELTA VALUES, AIRSPEED-ALTITUDE PARAMETERS
ENGIN. UNITS
PROGRAM D001
PRINT DATA IN ENGINEERING UNITS.
PROGRAM A057
INTEGRATE. FIND DEFLECTION FROM ZERO GUST REFERENCE. COMPUTE & CORRECT. AIRSPEED-ALTITUDE PARAMETERS.
GUST VELOCITY TERMS
ALL DATA
TIME HISTORY DATA
PROGRAM A020
GUST VELOCITIES AND TERMS
GUST VELOCITIES AND TERMS
PROGRAM A020 (REVISED)
COMPUTE GUST VELOCITIES.
GUST VELOCITIES
PROGRAM A060
SPECTRAL DATA
SPECTRAL DATA
PROGRAM A057
ANALYSIS: FILTER, CORRELATE, COMPUTE SPECTRUM, COHERENCY FUNCTION, TRANSFER FUNCTION, COMPENSATE FOR FILTER, FIND FREQ. DISTRIBUTION, STATISTICAL PARAMETERS.
IBM CARDS
IBM CARDS PUNCHED FOR DATA STORAGE.
PROGRAM A052
PLOTTER CARDS
PLOTTER CARDS
IBM CARDS PUNCHED FOR USE WITH MOSLEY PLOTTER.
OSCILLO-GRAPH
IBM CARDS
AIRLOAD DATA
PROGRAM A026
CONVERT DATA TO ENGINEERING UNITS. SOLVE SHEAR AND BENDING MOMENT EQUATIONS.

Figure 19

In the case of the FM system the preliminary processing program accepts IBM 727 tape from the tape handler of the ground FM to digital converter. The program then executes the functions of data identification, time selection, calibration, etc. that are required to reduce data in "counts" from the analog-digital converter to engineering values useful in the subsequent analytic processes or for print out for visual examination.

The preliminary processing program for the digital system accomplishes the same results as above and in addition accomplishes the more sophisticated calibration.

ANALYSIS PROGRAM

Gust Equations Program

The gust equations program receives taped data of each variable measured from the preliminary processing program. These variables are substituted in the gust equation, and gust velocity is calculated for the three gust components. The gust velocities are output on tape for the same time interval as the input data or in multiples of that time interval.

Spectrum Program

The analysis programs obtain an estimate of the power spectrum density function of a stationary stochastic (random) process for a sample record of length T by the use of Tukey's method and certain refinements thereto.

At present, the A060 program (see figure 19) handles the major portion of the analysis. It will eventually be replaced by a more automatic and comprehensive revision.

The mathematics of spectrum computation being well known, no attempt will be made to develop the theory, other than noting that, in accordance with data being processed digitally, the prime method of making the estimates is the Tukey procedure.

Rather, the purpose here is to discuss refinements and additions made necessary by particular problems encountered at Douglas in the reduction of time histories. Data analysis will be discussed in each of it's seven stages.

First, suppose that the data to be reduced is corrected vertical gust sampled at equal time interval Δ t. This data is screened and corrected for drop out, either singular or serial, and for low frequency, long term trends. An explicit definition of drop out is as follows: If $\{X_i\}$ is the original time sequence, then let

$$\psi_i \equiv \frac{|X_{i-1} - X_i| + |X_i - X_{i+1}|}{2}$$

If $\psi_i > \mu \sum_{i=0}^{N} \psi_i^2$, then X_i is considered to be drop out relative to μ.

(Five is a standard value for μ)

Multiple point drop out may be expressed similarly. Drift and constant of integration error are removed by a root-mean square fit of a polynomial.

Second, statistical checks may be made on the data to see if it is stationary random and Gaussian in character. Due to the long term drift error frequently found in the data, good estimates of some of the standard parameters are not obtainable from the time series at this point in the data processing. The variance, for example, can be more accurately computed from the spectrum itself, as will be noted later.

Third, numerical filtering may be performed. Numerical filters are related to the electrical kind, differing chiefly in that they are periodic - that is to say in the frequency domain their pattern repeats above the Nyquist folding frequency. Numerical filter patterns are available to do high and low pass, moving average and special purpose filtering. Some of these filters have been designed so as to have no phase change whatsoever. Compensation for the filtering must be applied to the spectrum later in the analysis.

Fourth, the time auto correlation function is calculated. This is an intermediate step in the spectrum computation.

Fifth, the spectrum is computed from the auto correlation. As the validity of this procedure, when certain data restrictions are met is well known, no detail of this need be given.

The net effect is similar to results obtained by a wave analyzer, although the results are not strictly comparable due to inherent mechanical limitations in each system. Indeed, the numerical method uses a lag window ("Hamming") which is related to the variable, narrow band and pass filter of the wave analyzer.

Sixth, compensation for filtering is applied. Compensation must be made for both the above mentioned numerical filter and for analog filtering applied to the data in the hardwave.

Last, the variance of the original time series is computed from the spectrum. Low frequency drift terms make computation from the

time series prone to be much too large.

Typical values for these computations are as follows:

Length of run, T = 240 seconds

Sampling interval, $\Delta t = 0.04$

Number of degrees of freedom = 100

From the above, it follows that:

Total number of samples per run, N = 6000

Number of auto-correlation lags and spectral estimates, m = 125

Nyquist folding frequency, $f_n = 12.5$ cps

Filtering is generally of the form:

$$Y_i = A_o Y_i - 1 + A_1 X_i - 1 + A_2 X_i \qquad \text{(Cornell - Type filter)}$$

The above is a sketch of the basic spectral computation procedure (due to Tukey) in addition to the general method, spectra may be computed by the cascade method, and the statistical parameter method (due to K. D. Saunders).

Also, frequency distributions, coherency functions, cross spectra, quadrature spectra, cospectra, transfer functions from spectra and so forth may all be computed from the recorded data.

The present IBM 704 program that does the computing is a combination of eleven programs. It is being revised to save computer time and to add flexibility to reduction procedures.

ACKNOWLEDGEMENT

The author wishes to acknowledge the assistance of the members of the Douglas Testing Division B-66 Low Level Gust Project in the preparation of this report.

FORUM - SESSION IIA

Session Chairman:
Mr. John H. Meyer, Atom Apply

Panel Members:
Mr. Mel Stone, Douglas Aircraft Company
Mr. Ken Eldred, Western Electro-Acoustic Laboratory
Dr. W. L. Howland, Lockheed Aircraft Corporation
Dr. J. C. Houbolt, Langley Research Center, NASA
Mr. R. Steiner, Langley Research Center, NASA
Mr. Charles F. Jackson, Boeing Airplane Company
Mr. J. Howard Wright, National Bureau of Standards
Mr. Cyril G. Peckham, University of Dayton
Mr. Harry Press, NASA, Washington, D. C.
Mr. C. E. Pettingall, Douglas Aircraft Company

Editorial Note: Attention is directed to the editorial policies presented in the Preface which were followed in editing the discussions of the Forum.

CHAIRMAN, MR. MEYER:

We have three papers which we didn't have time to present but we felt were so important that they should be mentioned. In our forum session we'll start off by just a brief summary of each paper. The subjects of the summaries and the speakers in the order of presentation are: "Predicting Aircraft Service Experience" by Mr. Cyril G. Peckham, Director of the Division of Data Processing, University of Dayton; "Application of Power Spectral Techniques to Dynamic Response Flight Testing" by Mr. Harry Press, Chief of Structures, Materials Division, NASA, Washington, D. C.; "Data Collection and Analysis Systems for the B-66 Gust Survey Project," by Mr. C. E. Pettingall, Chief of Aerodynamics, Flight Test Division, Douglas Aircraft Company.

MR. PECKHAM:

The title of this paper is "A Two Dimensional Acceleration Peak Frequency Model for Predicting Flight Envelopes" and the author is Mrs. Jeanne Truett of the Division of Data Processing, University of Dayton. The paper describes the procedure for developing an analytical expression for rate of occurrence of acceleration peaks and constructing prediction envelopes. The formal aspects of the problem of fitting bivariate frequency data for prediction of extreme maneuver loads are described in Section I. Section II contains the history of our efforts to develop a solution to the problem, and a brief outline of the related efforts by other workers. A procedure for constructing prediction envelopes which meets the requirements of flexibility, simplicity and general applicability desired of the solution is described and illustrated in Section III. Because of time, we are skipping the body of the talk which will be included in the printing of the proceedings of the symposium and I will just read the conclusions and show you one example.

The procedure described here originated in the idea of fitting a general exponential type of joint probability function by expanding the exponent in a Taylor series and estimating the coefficients by the method of least squares. However,

there is nothing in the assumptions or method to force the estimated function to yield an exponential function having the properties of a probability function. Rather than solving the distribution problem, this approach reconstructs it within the compass of the powerful methods of general regression theory. The problem becomes one of constructing a flexible regression model to represent, exponentially, a set of rates of occurrence. This method has been applied to data that was collected on VGH recorders of the Hathaway type on F-86 aircraft at Nellis Air Force Base from 1954 to 1956. The types of flight, gunnery flights and aerial acrobatics, have all been lumped into one group of data which we classify as a composite. The first slide will show you a plot of the observed frequencies against the theoretical frequencies. May we have the first slide, please?

You can see that the agreement of the logarithms of the observed and computed frequencies are very good since the data are plotted on logarithmic paper. The second slide will show you the actual data; superimposed on it are the curved lines which represent the envelopes which will predict the rate of occurrence of one G per airspeed range for a given number of hours of flight. You will notice it is not too good down in the negative range since we don't have too many points down there to work with. To obtain a probability curve of the type that Colonel Taylor used yesterday in his talk, it is only necessary to integrate this surface with respect to velocity.

MR. PRESS:

The title of the paper is high-falutin'. I might really define it more precisely. This paper is concerned with methods of estimating frequency response functions from flight test data of vehicles exposed to random type disturbances. The subject was touched on briefly this afternoon in several papers. The desirability of such determinations was clearly shown to be one of the essential parts of any fatigue prediction procedure. The basic approach utilized is essentially a power spectral approach of the type described in several other papers today. The power spectral theory provides two basic relations which mean measurement of an input and output. They also involve a second equation, the relationship between the input and the cross spectrum between input and output, so these two relations provide independent means of estimating at least the amplitude of the frequency response function. As in any flight test when you have two measurements of the same quantity, you fairly soon find out that you are in some kind of trouble and this is precisely the case here. If you employ these in the practical cases, two independent estimates of the frequency response function, invariably the estimates differ by very large amounts. We stumbled into this accidentally and then had to battle our way out. It turns out however that there are some relatively simple ways out of the dilemma. It turns out that both estimates are frequently in error by relatively large amounts. It is however possible to use them in combination and come out with rather excellent estimates of frequency response. We have been able to achieve reliability of the order of plus or minus 10% on amplitude. In successive tests of very short duration using the cross spectrum method we were able to obtain - the first three times I saw it I didn't believe it - phase angles to five degrees. So I think the situation here is relatively good. I might mention it is quite surprising because frequently the actual reliability of the individual spectra are well below this. It just turns out that these ratios are very often much more reliable than individual spectra. I think this should turn out to be quite useful in the flight test programs people now have under way.

MR. PETTINGALL:

In this particular program we have completed twenty-three flights to date in various locations throughout the United States. We have used an FM recording system in conjunction with some special sensory devices developed by the Douglas Company for establishing the angle of attack to the required precision and also establishing the gust input together with the instrumentation required to take out the effects of the airplane's motions and the motion of the sensing devices in contaminating the gust input. To date we have found it somewhat difficult to cover the data with the required precision, so we are at the present time installing a digital recording system to reduce the noise injected in the system by recording and also to make possible more reliable processing of the data throughout the remaining analysis procedure. We have found it possible at the present time to establish the power spectrum of the gust over a frequency range varying from about a tenth of a cycle per second to ten cycles per second. This is at an airspeed of approximately six hundred feet per second. We have found the effect of noise in the system to be very, very significant in establishing the power spectrum. For example, typical noise figures experienced with the analog system have varied somewhere between ten to twenty counts out of a thousand counts full range of the article you're looking at and this is found in some instances to mask considerably the power spectrum of the real gust. So with the additional sensory equipment that we are presently planning and installing in the airplane - and with the installation of the digital system - we hope to knock this noise level down significantly and make it a factor of two to four and thus establish more reliable data.

CHAIRMAN, MR. MEYER:

Now we get down to the questions you ladies and gentlemen have been turning in all afternoon, and let's just make the circuit on these in the same order in which the speakers presented their material. Perhaps by having each speaker answer all of the questions, we can get a better picture. Mr. Stone, would you start in and do what you can with your questions please?

MR. STONE:

The first question is from Mr. Chernoff, Republic Aviation Corp: "Has any method been proposed for deriving transfer functions from flight data?" On the B-66 program which I didn't have too much time to mention, an Air Force sponsored program, we are going to attempt to derive the transfer function experimentally. In other words, when we obtain the output response and divide this by the input, we hope to get good transfer data. Now we have accomplished this for the CG acceleration data and it looked pretty good. We have an instrumented airplane as far as bending moment is concerned and we are going to make the check as far as bending moment transfer functions are concerned and, I hope, get good results.

Mr. Thompson, Cessna Aircraft Company asks: "Would you or Mr. Christensen of your company care to comment on the promise that 'Fatigue Damage Monitors' have, as indicators of fatigue damage levels, in primary aircraft structure?" I know what he is talking about but I didn't speak of it. Well, you are all familiar with the fatigue coupons that we use in the testing machine. We have applied this to the airplane structure by bonding each end of this so-called fatigue coupon which, therefore, receives the same strain history as the actual structural element itself. This is what, by the way, is meant by the fatigue monitor and the answer to this question is that so far we are using them on the C-133. We have

approximately 800 hours to 1000 hours of flight. We didn't get any to break within this particular time limit. We calculated that in approximately 1200 to 1500 hours of flight we should have our first fatigue monitor break. We have a series of five of these which have different thicknesses to them, and, therefore, they actually operate at different levels. When you get the first one to break, this gives you an indication as to when the second one will break, say, at 2000 hours, the third one at such and such. By that time you've got a pretty good plot, and it's supposed to give you an indication when your highly loaded element is actually supposed to sever. So far we haven't had any sever, Mr. Thompson.

I have another one by Mr. H. W. Foster, Lockheed Aircraft Corp: "On your discrete gust statistics, what is your amplitude parameter?" I wonder if you are talking about the one signal that I mentioned as a discrete signal. The one I showed on the chart was the elevation versus the distance on the ground and I said that this could be used as a time signal for gust. Our experience, for instance, on the B-66 is such that if you get a true vertical gust time history, this can be correlated with the so-called normal or rough air turbulence in terms of the gust velocity and peaks per second. I don't know whether I've answered your question. Would you like to amplify, Mr. Foster?

MR. FOSTER:

You've answered the question pretty well. I was mainly interested in whether you were measuring the peak relative to the zero load rather than range.

MR. STONE:

The next question is by Mr. Melcon, Lockheed Aircraft Corp: "Have you made any structural fatigue tests in which the input loading was purely random exclusive of sonic fatigue tests?" I notice it is a loaded question! As a matter of fact, the answer is no, except in our vibration study, where some of the specimens were panels. We did place on our shaker, and excite to 4G amplitudes panels of various types of construction. We had hot sections and every imaginable type of element on the panel, and we did derive or try to derive a loading for this; but, of course, this is vibratory in nature and you're asking the question that we are all working on at the present time. We have a shaker that we are going to put through its paces to try and develop this particular structure as far as random fatigue testing is concerned.

This next question is by Mr. Bouton of Norair: "In the light of yesterday's emphasis on the fact that fatigue is a local effect, do you think a simple load factor spectrum is sufficient even for the wing? Would it handle Dr. Howland's case on rear spar fatigue on C-121; the B-47 root and midspan failures, etc?" I don't think that I expounded on any particular type of loading that would represent all of these. I did mention the fact that a discrete representation should not be lost, because of the fact that we have correlated this on many airplanes, and I mentioned the DC-3 because I know that it wouldn't arouse any particular comments. But they can be used on other airplanes. In this particular case, for instance on the B-47, where you do have the swept-back version (which by the way the response characteristics are entirely different from the ones on the B-66 and that's why this is such a strange business), I would say that the simplication of just the load factor representation would not be correct at all.

This is another question by Mr. Bouton of Norair: "Can you use runway power spectra in analysis? What kind of transfer function do you use when strut friction gives a discontinuity in the response?" Well, if you can't get a linear

representation as far as transfer functions are concerned, then we'd better use the load data that goes right into the structure itself from the gear. If you want to use linear representations - fine. We can use some of these power spectral techniques but when you get a discontinuity and I've seen this on linearity, I think we had better use another method. Is this what you had in mind? Fine.

The last question is by Professor Hiller, Columbia University: "Can you describe the load spectra by some mathematical means, such as an exponential spectrum?" The input spectrum that I showed on the second slide which is the power spectra of the vertical gust velocity is normally plotted so that the area under this particular curve expresses the variance, and if we take the square root of this - of course - this is the sigma term. Now in the report that Mr. Houbolt and NACA Report 1272 showed there have been some liberties taken and they have been represented mathematically by actually using the term sigma square and a mathematical representation which shows a shape but you have to actually assume the scale of turbulence. Now if this was the power spectrum equation or representation you're talking about, then maybe I have answered your question.

CHAIRMAN:

The question was "Was there a mathematical equation for your input?"

MR. PRESS:

The history of spectral gust data is that we have been able to measure spectra over a limited frequency range. We've been reasonably successful over a range of gust waves that stem from ten feet up to several thousand, at least. In the two cases we've been able to take them from twenty up to fifty thousand. Now for most analytical purposes it is very convenient to have analytical representation and for the range of data we have, we found there were excellent approximations to this and these are essentially some of the expressions that have a long history of isotrophic theory or at least in the study of wind tunnel turbulence they take the form of one plus three times the product of ΩL where both Ω and L are squared terms. The spectral shape of $1/\Omega^2$ is approached when we go to very high frequencies. Now I don't know what is meant by the term liberties you referred to. I know that this has been a very satisfactory approximation for the range of the data we have so far. There are some cases now which require us to look at these things more closely. For example, the situation gets a lot more complex at low altitude and we're required to use more complicated expressions involving more than the two parameters we use in a simple case.

CHAIRMAN:

Panel, we're going to have to step up the tempo because we are running out of time and we would like to have a few minutes at the end of these formal questions to give the audience another whack at the experts up here, so let's see if we can make it fairly snappy and go on down the line. Mr. Eldred, would you give your questions, please?

MR. ELDRED:

The first question in this small stack here is from Mr. Arnold Galof of Radioplane Division of Northrop: "Is there a Mach number effect on level of

Boundary Layer Noise? It seemed that you sho' .d only a 'q' effect." I ran rather short of time by the time I hit that - B' ındary Layer Noise - and I would say this: That the "q" correlation that is s' .vn would be expected on reasonably simple bodies of revolution - not in liftin anges such as in the upper surface of a wing aft - that at any point on a veh .e once we had passed Mach one, the "q" that would be referred to would b ıe "q" that is external to the local boundary layer, not the free-stream ' ı". Another effect we would expect is that as one goes to very high velocities, kilometer boundary layer will extend much further out due to the heat exchange from the boundary layer into the missile and the correlation in any case would only be expected to be held to where one could find that the boundary layers were similar. Now there is very little data above Mach one. The data that is available does show that in only one position on the aircraft, there is in the transonic region sometimes a local maximum for pressure which will go up above this curve where apparently local shock effects are taking place and in some cases are dead. In some case where one gets into the supersonic range, the level seems to come back to the approximate level of predicted body local free-stream "q".

The second question is by Mr. Forney of the Wright Air Development Center: "Your Figure 1 shows SPL values on the order of 194-195 db for turbojets?" I think the coordinate was slightly off the screen. The ordinant was not db at sound pressure level which is the one we most customarily hear but it was db at power level. 130 db at the power level is actually equal to one watt. Now in the very near field of turbojets, the maximum measurements or range are approximately 165 to 175 db or 169, I think. This is very close to the turbojet, right on the edge of the moving stream. The highest of these values that have been measured very close to rockets - approximately two feet from the stream - are in the order of 172 db, so these are considerably lower than these 194 and 195 db power levels.

The next question is by Lt. Wrobel of the Wright Air Development Center: "Do you think the new by-pass axial flow turbojets will have reduced power levels for equal engine power outputs, where the high velocity engine jet is shrouded by a low velocity jet?" These by-pass engines have been shown to have lower power levels and if the by-pass were a complete mixture between the by-pass gas and the hot gas power before the nozzle, one could expect that the power would decrease on the order of the velocity change to the eighth power in accordance with Litel. In most cases the by-pass area is the shrouding type flow which will not particularly disturb the central core, consequently, the change in acoustic power, the decrease, is not as much as would be expected if the mixture could take place prior to the nozzle, so that the effective velocity profile across the nozzle would be a square profile but at a lower velocity.

The next one is from Mr. Ryan of the Wright Air Development Center: "In this symposium I have twice heard it said that exhaust jet noise results from turbulent gases. Is this in conflict with statements I have previously heard that noise is generated by the high shearing effect between the high speed jet and the surrounding air?" Well, this is not in conflict because the turbulence is generated in the shear zone as a result of the shearing stresses that are in the shear zone, so these two explanations are essentially similar in nature.

The next question is from Mr. W. T. Shuler of Lockheed Aircraft Corporation, Georgia Division: "Could you provide a list of publications dealing with boundary layer and turbine acoustical noise level calculations?" I'd be glad to talk with you about this after the meeting. It is a little long.

The last question is from Mr. O'Brien of the Wright Air Development Center: "Do you think it feasible to use acoustic properties of full scale airframes by artificially applying a sound wave to determine if any fatigue damage is present in service airplanes?" In my opinion this would only be feasible on a research basis and would not yet be practical for any large surface such as would be considered with aircraft or missiles.

CHAIRMAN:

Moving along to Dr. Howland, would you give your questions, please?

DR. HOWLAND:

The first question is by Mr. Fleugenhoff of North American Aviation, who asks: "Do you think that strain gauges at four to five spanwise wing stations can be used to measure wing load distribution accurately?" It depends upon a lot of conditions; whether the wing has external stores, whether it has pods on it. On a simple wing with no discontinuities in forces and loads, that is, if it has no tip tanks, no pods, no engines, you can certainly get good measurements of distribution with four or five stations, particularly if you take your measured data and fit it into analytical work at the same time. In the case I spoke about, we were making experimental work and theory go hand in hand. We take bending moment measurements and fit them into a theoretical distribution and get the best fit which, of course, when differentiated gives the shear loading which you know should be continuous or not. Differentiating again gives you the normal loading which is compared with a good polynomial relationship for spanwise distribution. Then get the best fit for your measured data. Furthermore at a given station you can get a least square fit to the measurement versus acceleration on roller-coaster maneuvers. So here again is a case where you can improve the accuracy in your measured data by getting the data into the best known theory and arriving at an empirical relationship. I described this in my paper a couple of years ago but that's the general idea and I think may help.

One other question from Mr. Creek of Westinghouse: "How do you determine the proper location for strain gauges?" This does take quite a lot of experience to locate the proper place for strain gauges. One of the techniques we use is measuring loads on the static structure. That is putting gauges on the static structure, and then selecting the best gauge location to put on the flight test article. Also of course, you frequently integrate several gauges to determine the proper output for a given quantity. In other words, to separate bending moment from torsion, you want an output proportional to torsion. It does take quite a lot of intuition to know where to put the gauges to assess just torsion. And then of course you calibrate it to be sure that you're getting more or less of a pure signal and you can isolate bending and torsion to where you get a cross-effect of maybe two or three percent, which I consider satisfactory. I'd be glad to talk to Mr. Creek later on this because it is quite a problem locating gauges properly.

CHAIRMAN:

We have a question from the floor for Dr. Howland.

MR. O'BRIEN:

My name is O'Brien from the Wright Air Development Center. "Do you ever use stress coat for picking out high stress concentrations?"

DR. HOWLAND:

Oh, yes, where we've had a fatigue problem on a landing gear part and we're not sure where to place a gauge, we frequently have used a stress coat in order to position the strain gauge. Primarily you've observed the crack or you're not sure where the crack started and you want to try to find where the high stress concentration point is.

CHAIRMAN:

Any questions for you, Mr. Jackson?

MR. JACKSON:

The first question was turned in by Mr. Berdahl of SAC Hq 2d Air Force: "Thirty years ago a method of non-destructive inspection of major primary structures (such as railroad bridges) was a check of natural frequency by vibration generated. With the present quality of instrumentation could not such an inspection method identify incipient failure of such primary structures as B-47 wing attachments?" Well, there are two aspects of that problem that I would like to mention. The first is that we have of course done resonant frequency testing of the B-47 as in all of our airplanes and in order to do this and get adequate repeatability we need to bring the airplanes into the hangar out of the wind, take the wheels and tires off of them and set them on a firm structural foundation lacking vibration. To do this of course would mean putting a SAC airplane out of service for some time and I believe would not be feasible as a solution. While you were trying to do this of course you could be looking at the holes directly rather than fiddling around trying to get the frequency. The second aspect is that in this particular case on the B-47, the gross area that was affected by the crack was quite often very, very small and the effect therefore was felt only over a very short length of the wingspan. This would result in an infinitesimal fraction of one percent of the change in the frequency and I'm not sure that repeatability-wise, we would be confident that we would be measuring the effect of the crack or some other thing. Does that answer the question?

The second question is from Mr. H. T. Jensen of Sikorsky Aircraft: "Explain the system by which we get the peak counting or the counting capability for gust load experience?" This is a rather lengthy thing which is covered in some detail in the paper. I will just mention briefly though that the peak data that we are using now is being reduced by Mr. Peckham of the University of Dayton and he might want to add or correct what I have to say. But the data being recorded currently on the Century oscillographs is being manually reduced. The controlled airplane data is recorded on tape, laid out for a quick look prior to complete analysis, converted into the digital system, and analyzed in the 1103A, and there will be more comment upon that later. Any more comments on that, Mr. Peckham?

MR. PECKHAM:

None.

CHAIRMAN:

Mr. Houbolt?

MR. HOUBOLT:

Well, I have three questions here the answers to which can be given very quickly. Before I take up the questions I would like just to make a comment on one of the questions that was asked of Mr. Stone which is in reference as to whether it is possible to derive frequency response from flights through turbulent air, etc. I would like to mention a Technical Note 4291 which was by Mr. Press and Mr. Coleman which deals specifically with this subject and goes into it very thoroughly, and, as a matter of fact, it is a subject of the paper which Mr. Press submitted to this meeting and will be included in the proceedings of this symposium. So this paper and TN 4291 will give you a very good account of how to deduce frequency response functions from flight measurements.

The first question is from Mr. Philleo of the General Electric Company: "Weren't the labels reversed on your next to last slide? Data were shown for airplanes to 245,000 feet altitude and for missiles to 45,000 feet." I think our artist was just simply carried away with all the slides he's been making recently on missiles and he inadvertently put down 245,000. The 2 should not be there, so the altitude should agree for the missiles and the airplanes in all cases.

The second question is from Mr. Galef of the Radioplane Division of Northrop: "The effect of aircraft flexibility was indicated to be significant. Yet, most of our turbulence data was measured on real, therefore flexible aircraft. Doesn't our turbulence data therefore contain a possibly substantial exaggeration factor?" It's a good point, Mr. Galef, but I would like to mention that in the work-up of the data, an attempt was made to correct for this amplification factor due to flexibility. For some of the aircraft which were judged to be very stiff and have very little pitch effect - no correction was deemed necessary in the data, but for the other half that were rather flexible — up to 20% reduction in the derived data were made in an effort to take into account this exaggeration effect that you speak of here. So the data that we've presented should be essentially the corrected and real velocities that you experience.

The third question is from Mr. Parmley of the Wright Air Development Center: "The techniques of Power Spectral Density is capable of predicting the frequency of peaks. But, as shown by Mr. Jackson's paper, these peaks occur about varying means. What are the relationships of the peaks as given by power spectral density to the means about which they occur?" A good question and it is simple for me because I'm going to refer the question over to Mr. Press. Mr. Press is one of the outstanding pioneers in the power spectral density techniques to aircraft application. He has given many leading papers on this subject and was the co-author with Mr. Steiner on load prediction techniques in one of the references I indicated. I am sure he can give a very satisfactory answer to this question.

MR. PRESS:

First let me say that Mr. Jackson's technique is only one of many that has been used. It used to be that almost any expert had his own special way of converting counts from a time history to some number of values. These are generally very similar. Generally two methods of counting lead to about the same relative position in two different cases. Most of the judgments you use in making these counts as near as I can tell seem fairly qualitative but with not too strong a physical basis.

Now to answer the question. In general if we are talking about random processes of a stationary type, it is possible from the spectrum to determine any characteristics of this type that you're concerned with - any particular way of figuring it, by purely analog device techniques. Simply take input noise and count any combination that you want to make. This can be a fairly expensive job and whether this is worth doing is somewhat of a question perhaps. Analytically these things are almost impossible to determine when you go to compute a case of this kind. I think, however, in the long run that you ought to take a view of the whole matter. I think it might be well to look for some other system for representing the damage of random signals transposed and maybe the answer might come up a lot simpler if we look at some simple description of the signal with the power spectrum especially for these cases for which the power spectrum contains all the information about the signal. It contains at least all the different information about the signal, so you would presume then that the damage must be some function of the power spectrum. This is usually a fairly simple thing, a simple shape, it may be described by a three or four parameter family. And I would certainly suggest that somebody take a look at this kind as opposed to the damage rate approach. I have been wanting to get this pitch off for a long time, thank you.

MR. JACKSON:

Well, Harry, you are certainly right. There are a number of ways to analyze and interpret the load data that are obtained. However, we believe that the approach that we suggested is a more rational approach to the problem than this peak count approach. It does represent and furthermore there is a substantial difference at least in the case that we have observed between peak count and load cycle approach. I think that your suggestion of going to some other fundamental approach is fine only the problem is that you are faced with a lot of peak data based upon conventional S-N curves and that a limited number of samples will always be using these type of curves and therefore need some approach such as we have suggested.

CHAIRMAN:

Mr. Wright, did you get some questions?

MR. WRIGHT:

There is a question from Mr. Rolfe of the McDonnell Aircraft: "Are any recorders placed on airplanes in the combined military program without the pilot's knowledge. If not, how can we be sure of good maneuver data at high values?" This I don't know. I should remark, in orientation, that the Naval Air Development Center at Johnsville has the responsibility for determining the sample size and how to validate this data and so forth. I see there are some gentlemen from BuAir here. Would you care to comment on that?

MR. GRIFFIN:

My name is Griffin from BuAir. We can't be absolutely sure but in taking the philosophy that the pilot has one of these instruments in the airplane for a long period of time he will more or less forget about it. He will be somewhat conscious of it at first but he has a lot of things to be occupied with. Now we feel that by keeping the instrument busy in the airplane over a long period of time we will minimize the conscious effect of such a thing with the pilot.

CHAIRMAN:

Thank you. You might stand there in case you want to answer another!

MR. WRIGHT:

Another question is from Mr. Foster of Lockheed Aircraft: "What kind of study resulted in a maximum frequency response in the recorder of six cycles per second? Does this apply to large airplanes only?" Well first, let me mention that the Aircraft Structures Laboratory of the Naval Air Materiel Center at Philadelphia and the Structures Branch of the Aircraft Laboratory here at WADC are the two branches in the military that are operating this program and are getting it started. Perhaps one of them might care to comment? I might say that when these data actually are digitized, the sampling rate involved which will probably be five per second and maybe ten per second, and for practical purposes, this amounts to requiring a filter ahead of the sampling process that cuts off substantially lower frequencies. Would you care to comment?

MR. GRIFFIN:

We selected a sampling rate of five per second on the basis of power spectrums that were derived by NASA on F-86 and F-84 airplanes, I believe, and we found in these instances that there were no power in the maneuver loads beyond frequencies of two cycles per second. The highest frequencies occurred in maneuvering pitching accelerations and rolling accelerations. The sampling rate of five per second would give you all the information throughout that frequency period.

MR. WRIGHT:

There is another question from Mr. Galef of Radioplane Division of Northrop: "Differentiation of data is usually an unsatisfactory process because the noise present in the signal is exaggerated by the process of differentiation. How do you avoid this problem in converting roll rate to angular accelerations?" It's the lesser of two evils apparently. There has been some discussion in the services over this. There is some data taken by NASA, data involving accelerometers - the Navy used accelerometers. It works badly for integrating the accelerations too and you have zero areas in this. You have pretty bad errors in getting velocity out of acceleration. The compromise made was to decide to use angular velocity measurements to get accelerations from those.

CHAIRMAN:

Thank you, Mr. Wright. Gentlemen, we are out of time. It is five-twenty and the busses load at seven o'clock, so we are going to have to cut this off now. There are still a few more questions. If any of you gentlemen would care to come forward, I am sure the panel would be glad to discuss them with you. I want to thank the panel for their efforts in this symposium. I want to thank you for your kind attention, and last but not least, I want to thank the Air Force on a very excellent job of putting on this session.

INTRODUCTION TO SESSION II-B

COLONEL JOHN P. TAYLOR

WRIGHT AIR DEVELOPMENT CENTER

The program chairman for our session on the "Fatigue Mechanisms and Properties of Materials" will be Dr. Richard W. Fountain of the Union Carbide Metals Company.

Dr. Fountain has been associated with Union Carbide since 1951 and is currently serving as Director of their High Temperature Materials Research program.

He received his undergraduate training at the University of Illinois and completed his graduate requirements at Leheigh University, receiving a Doctorate Degree in 1951.

He is a member of the Materials Advisory Board and a number of technical societies, and has authored numerous articles for trade and scientific journals on subjects pertaining to metals research and application.

It is my pleasure to present to you, Dr. Fountain.

PROGRESSIVE DAMAGE DUE TO REPEATED LOADING

by

T. J. Dolan and H. T. Corten

Department of Theoretical and Applied Mechanics
University of Illinois

ABSTRACT

A brief review is presented of the mechanisms leading to the nucleation and growth of fatigue cracks in metals. The statistical nature of the variation in fatigue life is emphasized by the localized nature of the plastic deformations that develop on a microscale under repeated loading. Refinement in dislocation theory of deformation and direct observations of dislocation movement have provided new information on the nature of the readjustment and interactions which precede the nucleation of a fatigue crack.

Based on these mechanisms a simplified "mental picture" of the accumulation of fatigue damage is developed and expressed in mathematical form for the prediction of fatigue life under a complex stress history. The number of damage nucleii initiated is determined by the peak stress encountered; the rates of propagation of damage are assumed to be proportional to a power function of the stress level. The resulting equation is compared with extensive experimental data on small metal specimens, and the discussion is extended to include additional factors that affect the fatigue behavior of fabricated structures. A specific example is included to illustrate the application of the relationship to the prediction of fatigue life in a particular structural application. This involves a tabulation of the expected load history from typical service measurements, and simplified laboratory test data that are necessary to evaluate two parameters that express response of the member to repeated loads. Good agreement has been observed between estimated and measured fatigue life for several complex stress histories.

THE MECHANISM OF FATIGUE

In discussing mechanisms of fatigue, it is important to realize that what is basic to the designer is on an entirely different level of inference and observation than what is basic to a research engineer; their ideas may in turn stem from an entirely different level of observation than that of the physicist or scientist. Thus, we must recognize different levels of "mechanism" depending upon the viewpoint of the individual.

Fatigue failure in metals is one manifestation of the behavior of crystalline solids under stress; we may divide our concepts and observations of this behavior into three general categories:

1. Large scale or phenomenological behavior characterized primarily by visual observation or by measurements on large volumes of material.
2. Microscopic and sub-microscopic phenomena, apparent only by use of special equipment and techniques such as microscopes, x-ray defraction patterns, etc.
3. Interactions between the atoms which compose a metal which cannot be observed directly and, hence, must depend upon observations of related effects to prove or disprove hypothetical theories of the behavior mechanisms.

A great deal is known about fatigue failure and the basic mechanisms for each of these levels of observation. However, the "state of the art" is inadequate because we have not been able to formulate accurate quantitative design rules for members subjected to repeated stressing. Actually, more is known about fatigue than is available on the mechanisms of brittle fracture, creep, etc. It is the statistical nature of the fatigue phenomenon that makes it impossible to evolve quantitative design rules. One can predict performance on the average for a large number of like components or predict failure in a certain percentage of cases, but to set an exact limit for a fatigue life or fatigue strength of a given member is impossible. Further complications arise because of our inability to make quantitative predictions of the significant stresses due to surface blemishes, service induced nicks (or due to the irregular contours of the part itself), and internal "notches" in the form of non-metallic inclusions, grain

boundaries, segregated streaks, etc., from which "stress-raisers" fatigue failures initiate.

At the phenomenological level of observation one finds that, after a large number of cycles of repeated stressing, one or more small cracks become visible; these grow progressively with repeated stressing but without large scale distortion of the member such as is visible under a single static load. Cracks grow at an accelerated rate that may be somewhat irregular, depending upon the size of the zone of high stress and the number of cracks that are developing. Visible cracks gradually progress and join until, upon subsequent cycling, they result in final sudden fracture of the part.

On the microscopic scale a metal appears to be composed of an aggregate of randomly oriented and shaped crystalline grains that are far from homogeneous. Localized inelastic deformations in "slip bands" develop within parts of crystals in highly stressed zones. Upon subsequent repeated stressing the slip bands multiply until at least a portion of the crystalline grains are greatly fragmented. The development of damage is a statistical process depending largely upon chance orientation and distribution of the weaker crystals and hardening constituents in the highly stressed zones.

In general, fatigue cracks initiate in the exposed surface of a metal part unless unusual circumstances or specialized processing treatments develop surface layers of substantially higher strength than the interior. The surface represents an imperfection in the atomic lattice arrangement of the crystals since this layer is not confined by bonds with adjoining grains. A free surface is thus more susceptible to the initiation of disruptions and voids that lead up to the initiation of a fatigue crack.

Forsyth (1)* and Wood (2) have shown by observations on a microscale, that cyclic stressing first develops a set of localized slip bands which gradually widen into a "striation" by a mechanism that consists of a to and fro slipping on closely spaced bands or adjacent planes under the reversals of loading. Forsyth indicates that the spacing between the striations is that of the slip bands produced on the first stress cycle and that the number of deformed zones appearing on the face of the crystal is proportional to the stress level in the region involved. This to and fro slippage results in notches and ridges in the surface by a sort of "ratchet" mechanism, leaving small crevasses surrounded by slight hills. In some instances actual extrusions have been raised out of the surface with neighboring intrusions which constitute voids or cracks in the surface. With the aid of the electron microscope, Love (3) and Craig (4)

* Numbers in parentheses refer to the list of references appended to this paper.

found that minute fatigue cracks could be detected within the striations at an extremely early stage of the fatigue process (as low as one tenth of 1 per cent of the nominal fatigue life). Thus, permanent progressive damage is formed by the generation of vacancies and microscopic voids within the slip band striations. Upon subsequent cycling these voids will grow, join, and finally result in a visible crack.

These observations emphasize the localized nature of the generation of voids and the development of large inelastic deformations on a microscale as contrasted to the general yielding that occurs under excessive static loading. They further emphasize the statistical nature of the phenomenon in which the initiation of slip and striations and the development of voids takes place in a random agglomeration of anisotropic crystals with mixed phases of constituents having elastic characteristics that differ physically from the matrix. Textural stresses on a microscale are undoubtedly developed at interfaces between the various constituents because of their differences in physical and mechanical characteristics. These variations in microstress together with the road blocks to the development of cracks offered by grain boundaries between the various constituents and impurities, contribute markedly to the many chance effects that control the initiation of voids and the generation or development of cracks in commercial metals as the repeated stressing progresses.

In appraising progressive damage from this viewpoint it is not difficult to see why large amounts of scattering in fatigue life occur. Even under carefully controlled laboratory conditions using duplicate test specimens from the same bar, it is not unusual to encounter ratios of 10 to 1 in cycles of a given stress to fracture. Furthermore, it is seldom feasible economically to run enough experiments to obtain the statistical information necessary to design a part for limited life with a predictable degree of certainty. Because of the added uncertainties that occur in the actual service loadings and minor differences in manufacturing tolerances and procedures, it becomes impossible to predict accurately a precise fatigue life for an individual member such as an airplane wing.

Most of the basic concepts of the mechanism of failure at the atomic level have been built upon dislocation theory from idealized pictures of movements of edge and screw dislocations within the crystalline lattice. That is, movement by slip within a crystalline lattice is visualized as a step by step displacement of a dislocation or mismatch in the regularly spaced atoms of the lattice. Thus far the theory is not sufficiently advanced to deal successfully with polycrystalline metals and the multitude

of foreign atoms present in commercial metals, nor with the complexities introduced by reversal and repetition of stress. Recent studies of direct observations of dislocation behavior have taken the dislocation concept out of the realm of imagination and established it as a reality. Such studies include electron microscope observations of thin films of stainless steel and aluminum (5) and etch pit studies of lithium fluoride (6) and germanium. Similarly, two dimensional studies of bubble raft models to simulate crystal lattice action, illustrate many of the features of dislocation behavior and have been observed under cyclic shear strains to simulate fatigue (7). In the bubble raft, fatigue cracks develop and grow by dislocation interaction and movement rather than by diffusion or condensation of individual vacancies. The regions of the bubble raft adjacent to the "crack" are cleared of dislocations which move to develop the crack. Fatigue tests of aluminum at liquid helium temperature by Rosenberg (8) also suggest that cracks probably form and grow due to dislocation motion. Vacancy condensation or corrosion are not necessary to the development of fatigue cracks since these processes are practically inhibited by depressing the temperature this low. However, they may contribute to fatigue damage at higher temperatures.

Keith and Gilman (9) indicate three mechanisms that have been proposed to account for the fatigue cracks seen to originate in glide bands at or near the surface of a specimen:

1. Dislocation pile-ups: densely packed groups of dislocations resulting from blocking by barriers; these act as incipient cracks.
2. Surface contour changes: mechanisms based on this concept involve the formation of notches due either to back and forth glide or to the extrusion phenomenon. These notches act as stress concentrations and generate micro-cracks.
3. Point defect clusters: accumulations of point defects, especially vacancies which result from non-conservative dislocation motions, weaken the glide planes. Fatigue cracks may develop as a result of stress concentrations around such clusters.

The dislocation pile-up concept requires the presence of obstacles to motion over an extended area; these are difficult to conceive of as being present and of sufficient strength. Keith and Gilman's observations suggest there is little change in dislocation density after a few strain cycles in lithium fluoride crystals. They prefer the mechanism of point defect clusters which coalesce into cracks during repeated

loading. The continual creation of new point defects would result in an increase in the maximum size of clusters until: (a) the cluster would attain a critical size to constitute a stable crack, or (b) the glide band would become so perforated with point defect clusters that the remaining material would tear under the influence of stress.

Since most of the theory and observations on the dislocation activity have been confined to relatively pure crystals, it is difficult to extend the observations to polycrystalline metals of commercial quality because of the presence of many foreign atoms, hardening constituents, non-metallic inclusions, etc., and the reversals of stressing which complicate the phenomena. There is also a wide gap between the observations on a single crystal and an evaluation of the modifying influences of the heterogeneous conditions in a commercial metal. Presently there is no method of statistical analysis by which one can predict quantitatively the behavior of a polycrystalline metal from the concepts of dislocation movement in a single crystal. Therefore, the engineer finds greater satisfaction out of the inferences obtained from the micro-mechanisms of progressive fracture. However, even these concepts have to be utilized on a qualitative basis and evaluated quantitatively by means of carefully planned and conducted experimental programs.

ENGINEERING DESCRIPTION OF CUMULATIVE FATIGUE DAMAGE

In this section a simplified "mental picture" of the development of cumulative fatigue damage is based on the foregoing observations, and a mathematical relationship is developed for the prediction of fatigue life under complex stress history. The resulting equations for fatigue life are compared with extensive experimental data on small samples of material to establish the validity of the simplifying assumptions and to evaluate a parameter of the material which expresses rate of propagation of damage nucleii. In the balance of the paper this relation is extended to a consideration of the implications with regard to the fatigue behavior of structures under complex stress history, with a specific example to illustrate the application and the evaluation of the parameters which affect the fatigue life.

A useful engineering description of fatigue damage must be simple and must combine the important variables into an expression that predicts fatigue life accurately and reliably. One simple expression that has been widely used is the linear summation of cycle ratios hypothesis,

$$\Sigma n/N = 1$$

where $\underline{n}$ is the number of cycles sustained, and $\underline{N}$ is the nominal life at stress σ.

The most obvious defect in this simple expression is that it does not give accurate estimates of fatigue life. Inherently, this hypothesis erroneously assumes that the stress history has no effect upon the response of the material to subsequent cycles of repeated loading and that stresses below the fatigue limit develop no damage.

As an alternative, the mechanism of the development of fatigue damage has been introduced by the authors into the development of an equation which possesses the desired simplicity and accuracy by including the important influence of previous stress history (10). That is, fatigue damage is visualized in terms of the accumulation of submicroscopic or microscopic cracks which propagate with each cycle of stress, with the number of cracks and rates of propagation dependent upon the stress levels. In general, the development of any void or disruption, even of sub-microscopic size, constitutes irreparable fatigue damage if larger than some lower limiting stable size. The mechanism of crack growth probably consists of a gradual preferential enlargement of sub-microscopic voids until eventually they join. The development may include jumping together by static fracture of some of the small regions between neighboring voids. Observation with the electron microscope reveals several short cracks within one slip band striation, each of which exhibits a jagged, irregular shape. This suggests that crack joining may constitute a significant part of crack growth.

This leads to the concept that two of the factors governing the fatigue life are: (a) the number of voids or cracks formed, and (b) the rate of propagation of these cracks. In the analysis which follows the "damage" is visualized as the accumulation of voids or cracking expressed in terms of the number of nuclei formed and the rate of propagation of these nuclei. The number of damage nuclei (visualized as sub-microscopic voids) that form throughout a critical section of the member will increase as the stress increases. Thereafter, the damage is propagated at rates that increase with number of cycles or that increase if the stress amplitude increases. Once formed, the damage at any nucleus may continue to be propagated at stress levels lower than that required to initiate damage.

Changes in the structure of the material are caused by extensive localized slip and in some cases by precipitation and phase changes in metasable materials. In addition, residual stresses on the microscopic and/or macroscopic scales may be introduced or modified during repeated cycles of stress. While these changes continue almost indefinitely, they occur rapidly during the early cycles of stress, and the rate of change during later cycles of stress is frequently very small. In terms of fatigue life,

it is this "modified" structure that is significant in progressive fracture. Since the highest stress experienced produces the largest changes in the structure, this peak stress determines the modified structure and number of damage nuclei. Extensive experimental data (11) have established these hypotheses for cumulative fatigue damage:

(a) The fatigue lives determined from different complex stress histories are each proportional to the fatigue life from a constant amplitude test at the peak stress.

(b) The process described as crack propagation includes practically the entire fatigue life of the specimen.

(c) The rate of propagation is governed by the amplitude of the repeated stress and the modified structure of the material.

In order to account for the damage that occurs in the structure during a random stress history it is convenient and adequate to equate this to the damage that would occur during constant amplitude cycling at the peak stress only. During cycling at any stress (σ_i) the corresponding fatigue damage, D_i, is expressed as

$$D_i = m_i r_i N_i^{a_i} \tag{1}$$

where m_i is the number of initial crack nuclei that were formed and eventually join to contribute to the final failure of the member. The quantity r_i is a coefficient of crack propagation for the modified structure of the material, and the term $N_i^{a_i}$, expresses increasing rate of crack propagation with increasing numbers of cycles of stress. Since the modified structure of the material is determined by the peak stress during the crack initiation stage, r_i may be expressed as a function of stress as

$$r_i = f(\sigma_i)$$

Experiments have shown that the exponent "a" may be taken as a constant for all values of σ_1.*

The assumptions introduced describe the phenomenon of cumulative fatigue damage with the same degree of accuracy as the measurement of fatigue life in carefully controlled experiments, It is undoubtedly true that refinements might be made in the theory; however, at the present time it is not possible to experimentally evaluate such refinements from fatigue life measurements. Conversely, a point of diminishing returns probably has been reached beyond which the theory and expression for fatigue life may become only more complicated without improving the accuracy of the estimates.

* This means that crack propagation at different stress amplitudes may be described by a family of curves of the same mathematical form.

For several different stress levels, $\sigma_1 > \sigma_2 > \sigma_3$, the variation in fatigue damage with number of cycles may be represented from Eq. 1 as shown in Fig. 1. Fatigue damage accumulation is shown by smooth curves with increasing slope which reach 100 per cent damage at failure. The influence of a previous stress history is introduced by using for the number of nuclei, m_1, that contribute to the final fracture, the value governed by the peak stress. Once initiated, cracks are propagated throughout the remaining life. Thus, for a complex stress history involving three levels of stress, σ_1, σ_2, and σ_3, damage may be pictured as accumulating as illustrated in Fig. 2. Note that m_1 is the same for all three curves in Fig. 2; however, r_2' and r_3' describe the rate of propagation in the metal modified by cycles of σ_1. For a finite number of steps in a stress history, individual terms derived from Eq. 1 may be summed as increments of damage. If the stress history consists of a large number of steps or blocks, it is convenient to represent the total damage by a summation of the individual contributions ΔD, as:

$$D = \Sigma \Delta D = \int_1^{N_g} dD \tag{3}$$

in which N_g is the total number of stress cycles for failure. By equating this damage at failure with that from Eq. 1 for constant amplitude cycling at the peak stresses, the expression for total life under a complex stress history is obtained:

$$N_g = \frac{i_1}{\Sigma a_i \, (r_i'/r_1)^{1/a}} \tag{4}$$

where a_i is the percentage of cycles at each stress σ_i and the subscript 1 refers in each case to the quantities associated with cycling at the peak stress. In this expression, the ratio $(r_i'/r_1)^{1/a}$ represents a group of parameters involving relative rates of propagation of voids and whose value dependsonly upon the amplitude of the stresses, σ_i and the peak stress, σ_1; for convenience this ratio will be referred to as $R^{1/a}$.

Two-stress repeated-block stress histories, of the type shown in Fig. 3, have been employed to investigate the dependence of fatigue life, N_g, upon values of a_1, the percentage of cycles at the peak stress. For a variety of pairs of stress levels Eq. 4 adequately described the total fatigue life, N_g, as shown in Fig. 4 for 7075-T6 aluminum alloy. Values of the quantity $R^{1/a}$ which best fitted the data for each combination of σ_1 and σ_2 were employed to compute the solid lines in Fig. 4. The

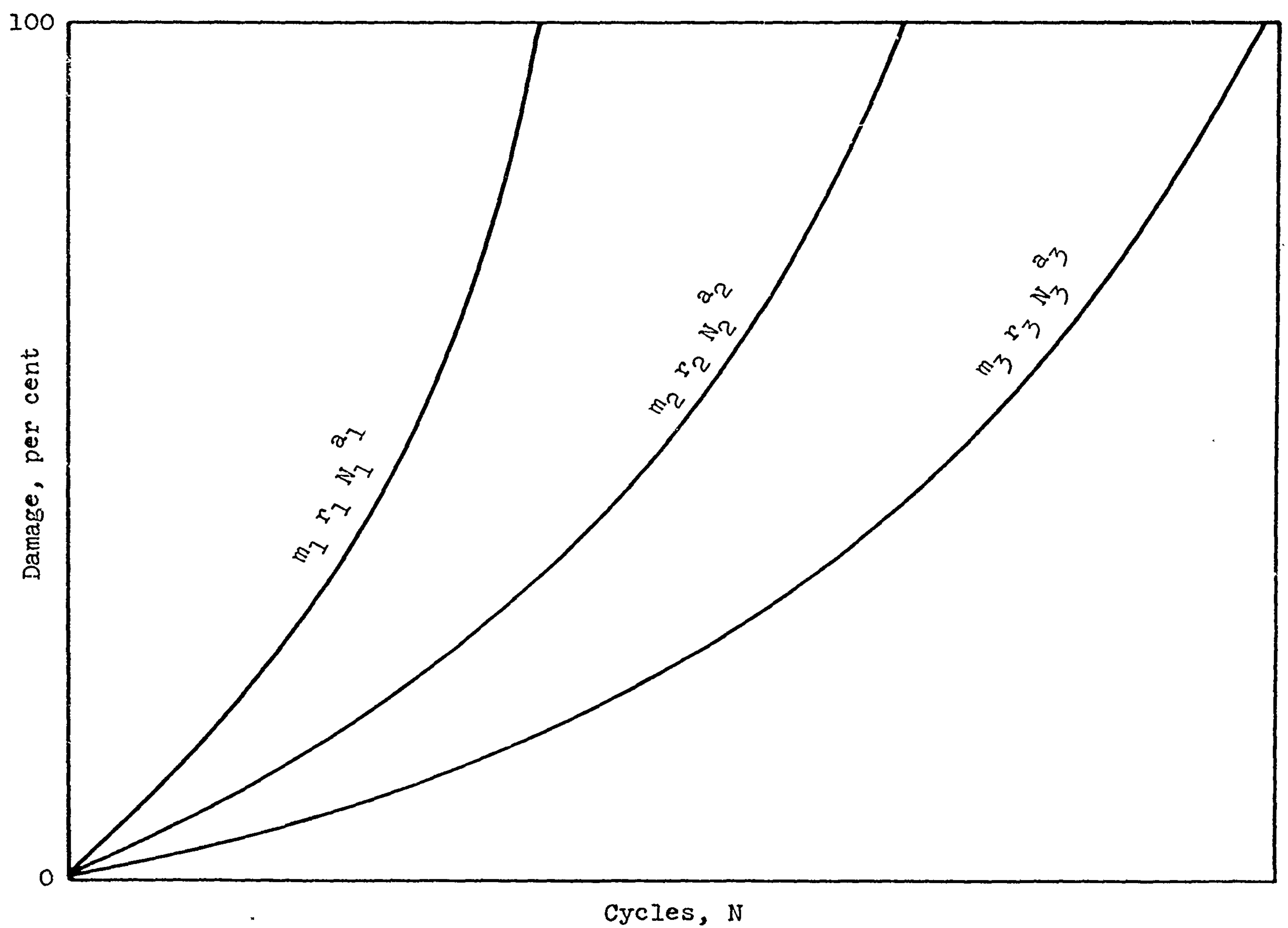


FIG. 1 SCHEMATIC REPRESENTATION OF ACCUMULATION OF FATIGUE DAMAGE AT THREE STRESS AMPLITUDES, $\sigma_1 > \sigma_2 > \sigma_3$

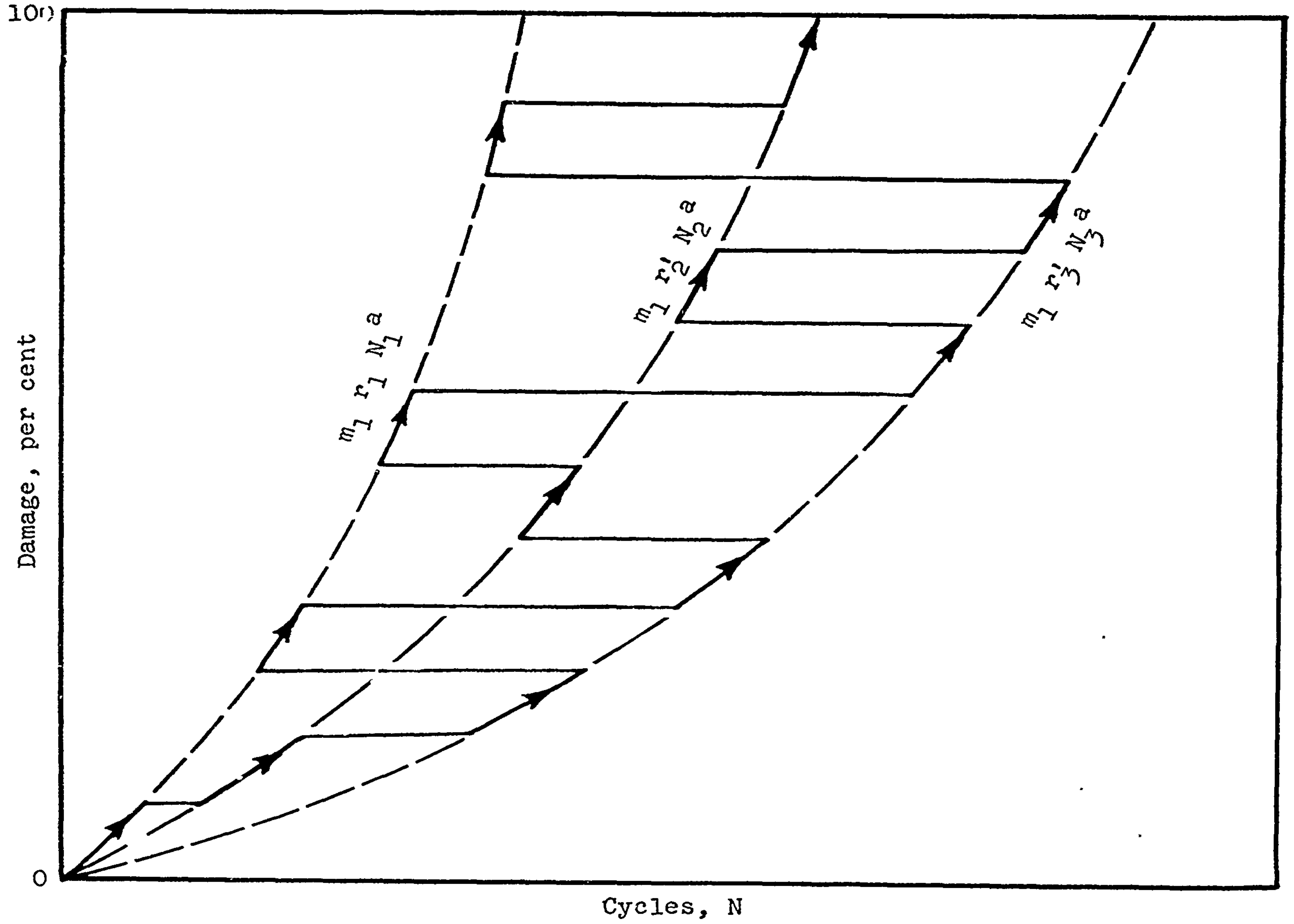


FIG. 2 SCHEMATIC REPRESENTATION OF ACCUMULATION OF FATIGUE DAMAGE DURING A COMPLEX STRESS HISTORY INVOLVING THREE STRESS AMPLITUDES

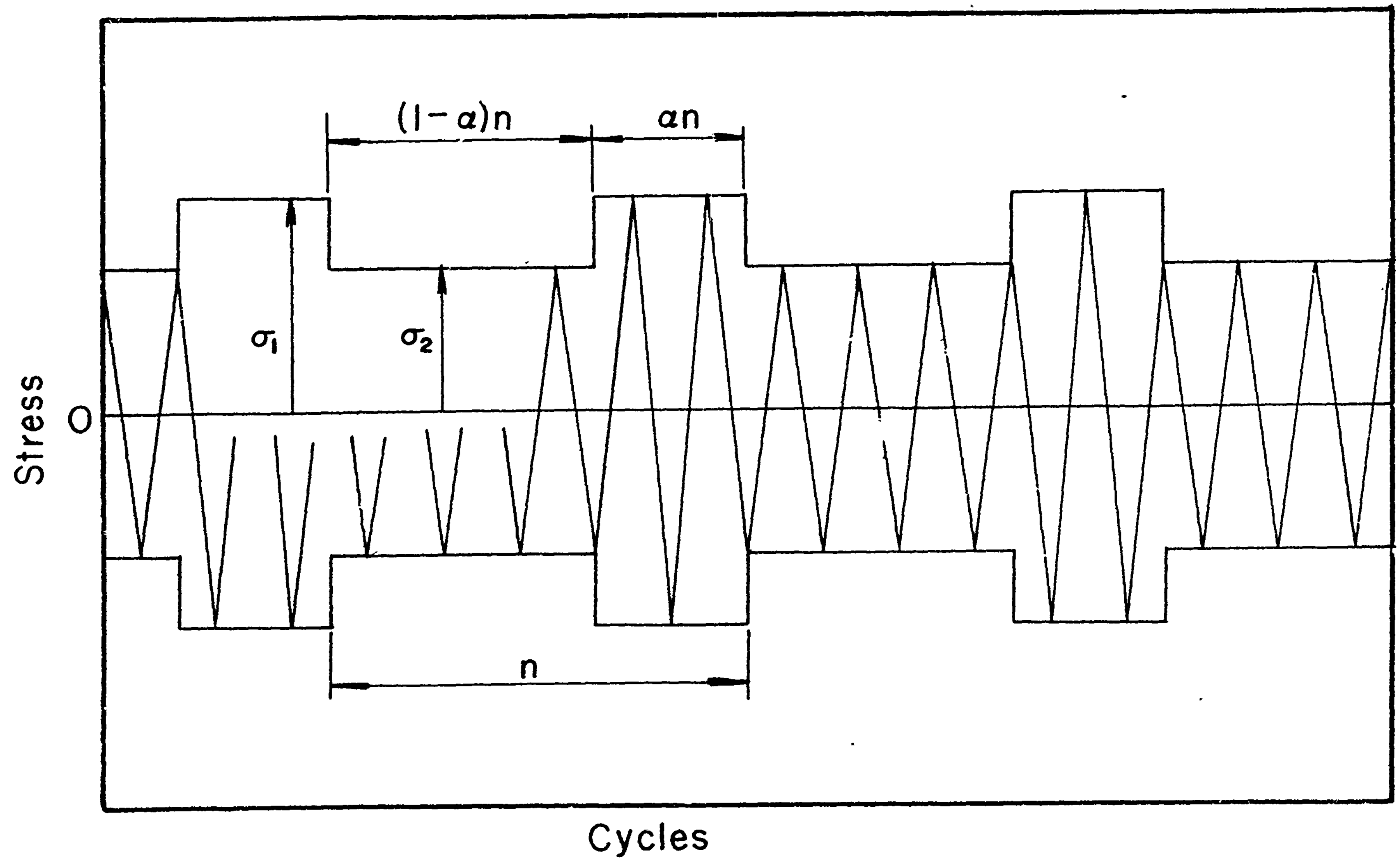


FIG. 3 STRESS HISTORY OF TWO STRESS REPEATED BLOCK EXPERIMENTS

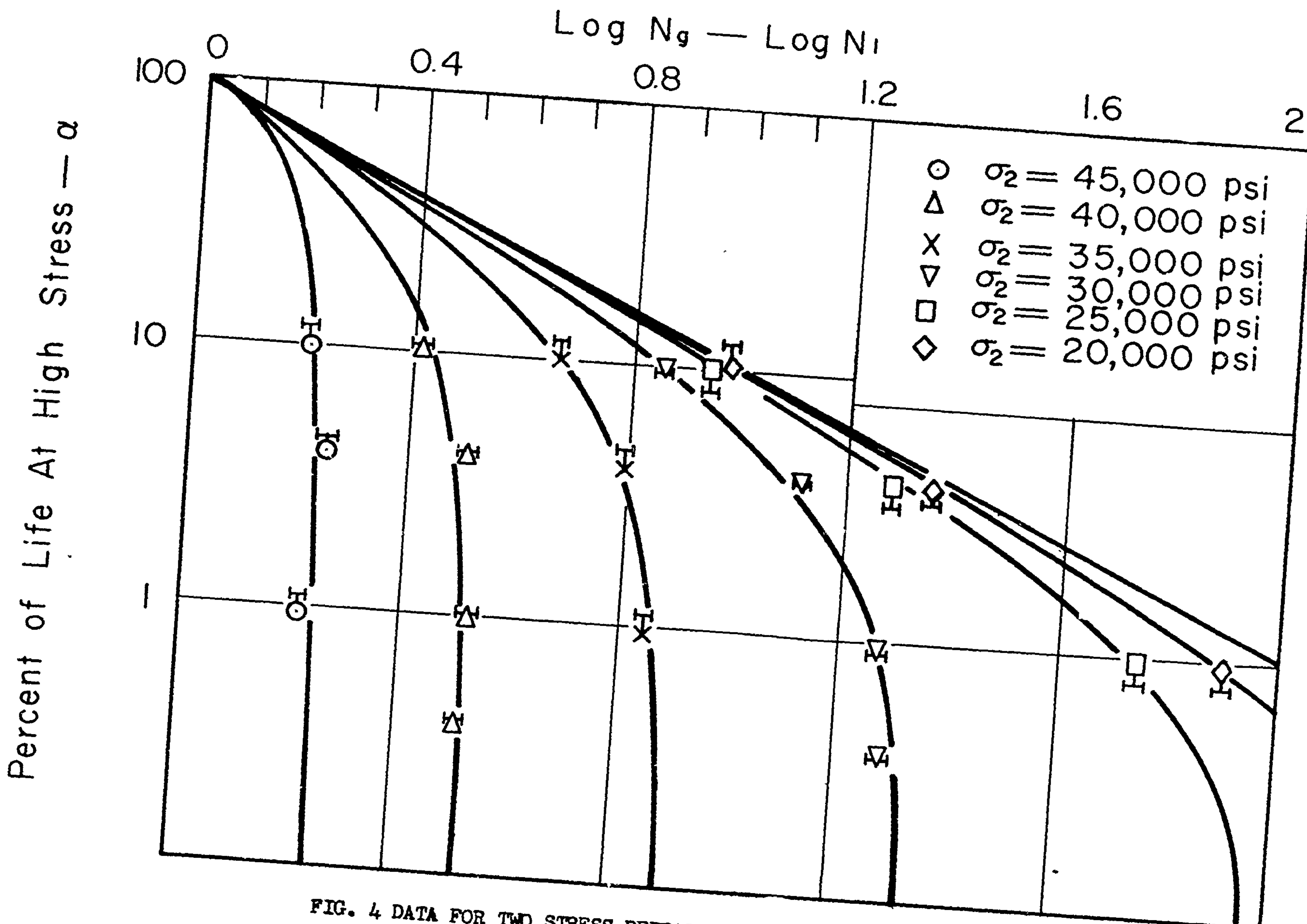


FIG. 4 DATA FOR TWO STRESS REPEATED BLOCK EXPERIMENTS WITH A HIGH STRESS OF 50,000 PSI, BLOCK SIZE 10,000 CYCLES, 7075-T6 ALUMINUM ALLOY WIRE, VASELINE COATED, MACHINE 4.

excellent agreement obtained for each pair of stresses confirmed the linear variation of fatigue life with N_1 and α_1 indicated by Eq. 4.

In Fig. 4 the value of peak stress was 50,000 psi and the values of σ_2 were as indicated for the different sets of data. Note that each point shown represents the mean (or median) value from a group of approximately 20 specimens. For this alloy three other values of peak stress, (σ_1 = 70,000 psi, 60,000 psi and 40,000 psi) were also employed resulting in similar diagrams.

From this large amount of data and the values determined for the ratio $R^{1/a}$ it was possible to study the relation between this parameter and the relative stress levels. In Fig. 5, the experimentally determined values of the quantity $R^{1/a}$ are plotted as a function of the ratio (σ_i/σ_1); it is evident that a straight line passing through the origin represents adequately all of the data shown. Thus, the relationship may be expressed:

$$R^{1/a} = (r_i'/r_1)^{1/a} = (\sigma_i/\sigma_1)^d \tag{5}$$

where d is the slope of the line in Fig. 5. Substituting into Eq. 4 gives the following simplified expression for total fatigue life:

$$N_g = \frac{N_1}{\Sigma \alpha_i (\sigma_i/\sigma_1)^d} \tag{6}$$

By employing this expression and the value of "d" found in the two-stress experiments, the life was predicted for a group of specimens subjected to tests involving various complex stress histories at different levels of stress. One typical complex stress history used is shown in Fig. 6; the stress amplitude is plotted versus the number of cycles in each sequence which consisted of a total of 10,000 cycles. This pattern was repeated a number of times in each experiment. Typical analyses of the number of cycles applied for each stress increment are shown in Fig. 7, in which the stress amplitude is plotted as ordinate and the corresponding fraction of total life α_i is the abscissa. A total of 26 different complex stress histories were applied to a group of four different metals (two aluminum alloys and two steels). For these experiments, the ratio of the experimentally measured life N_e, to the life calculated from Eq. 6 is shown in Fig. 8. The data are separated into four groups corresponding to the four different metals, and arranged in terms of the peak stress applied during the complex stress history. Each open circle represents data from 20 identical wire specimens subjected to a complex stress history. It should be noted that many of the

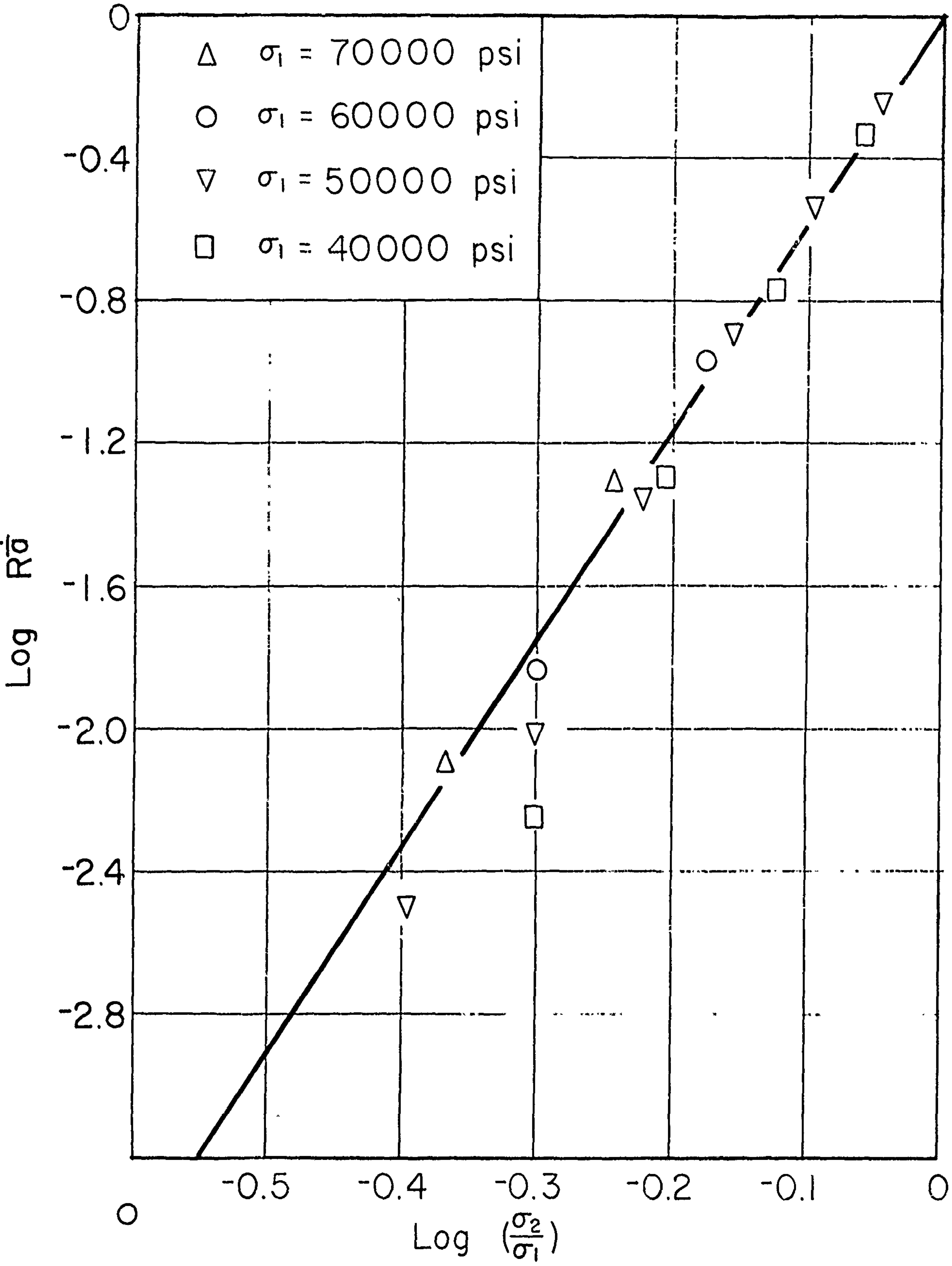


FIG. 5 CORRELATION BETWEEN RELATIVE RATES OF DAMAGE PROPAGATION AND AMPLITUDE OF IMPOSED STRESSES, 7075-T6 ALUMINUM ALLOY WIRE.

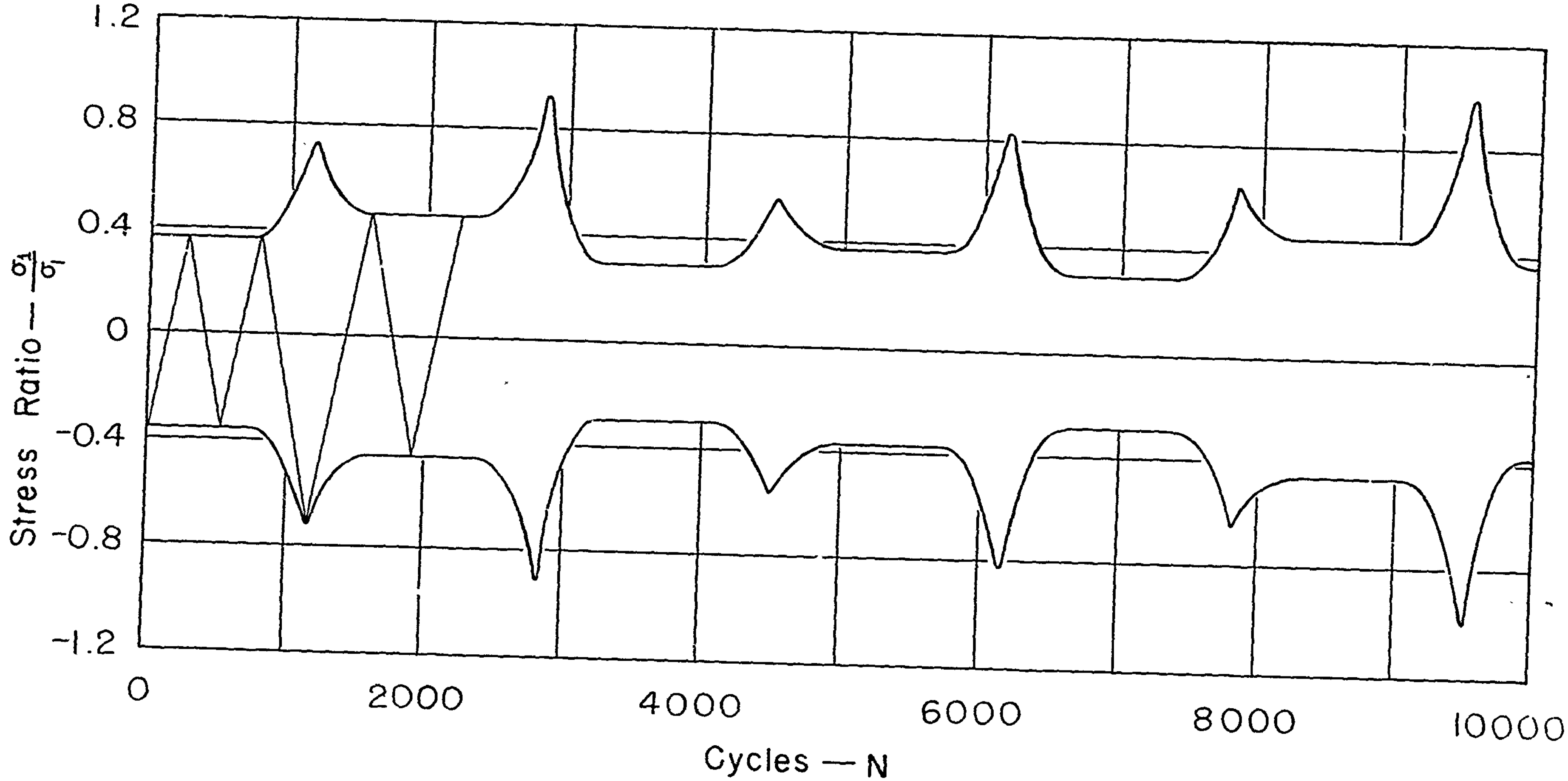


FIG. 6 PATTERN OF STRESS HISTORY OF CONTINUOUSLY VARYING STRESS AMPLITUDE EXPERIMENT, CAM 2

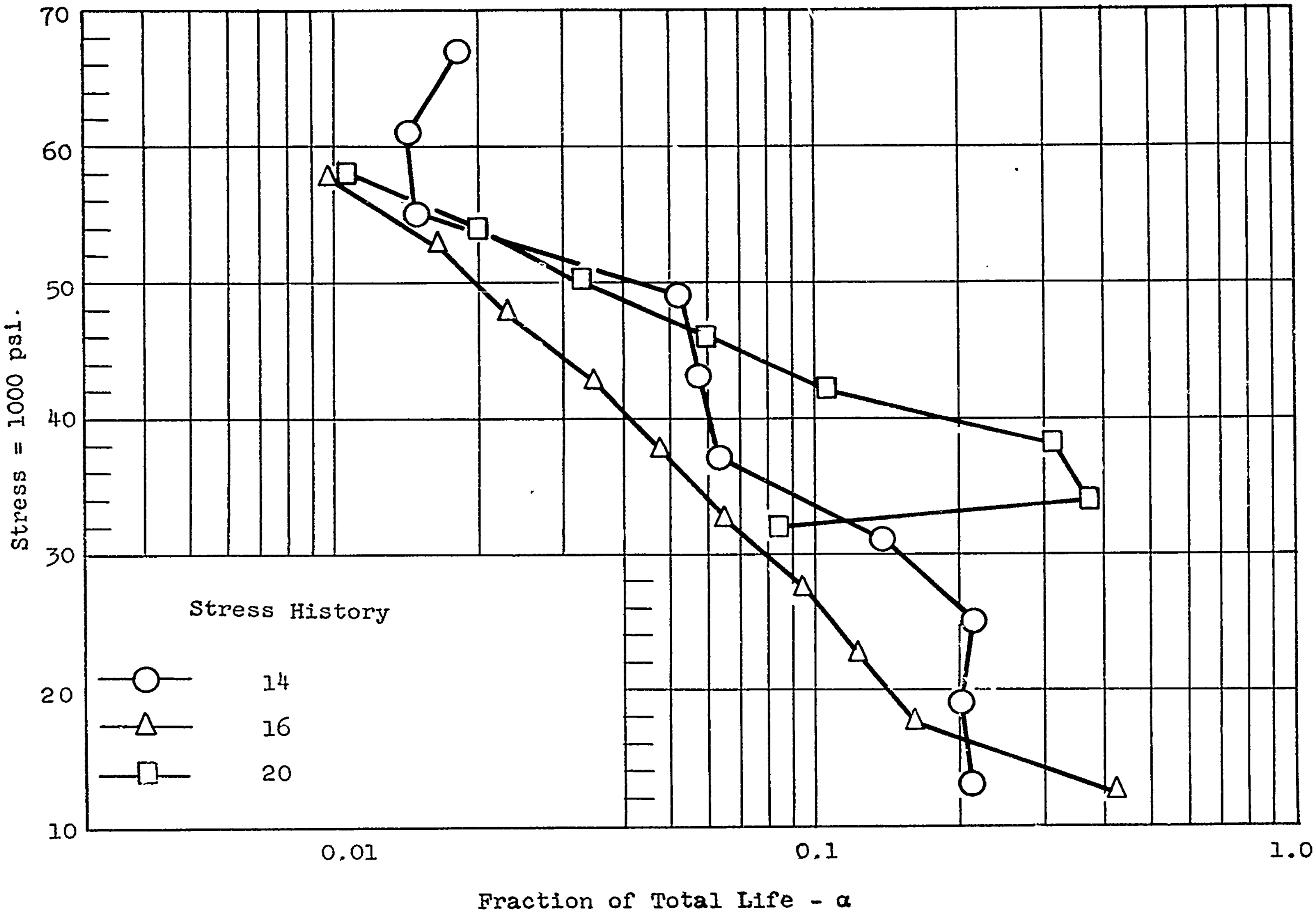


FIG. 7 FREQUENCY DISTRIBUTION DIAGRAM OF FATIGUE LIFE AT VARIOUS STRESS INTERVALS FOR CONTINUOUSLY VARYING STRESS AMPLITUDE EXPERIMENTS, 7075-T6 ALUMINUM ALLOY WIRE.

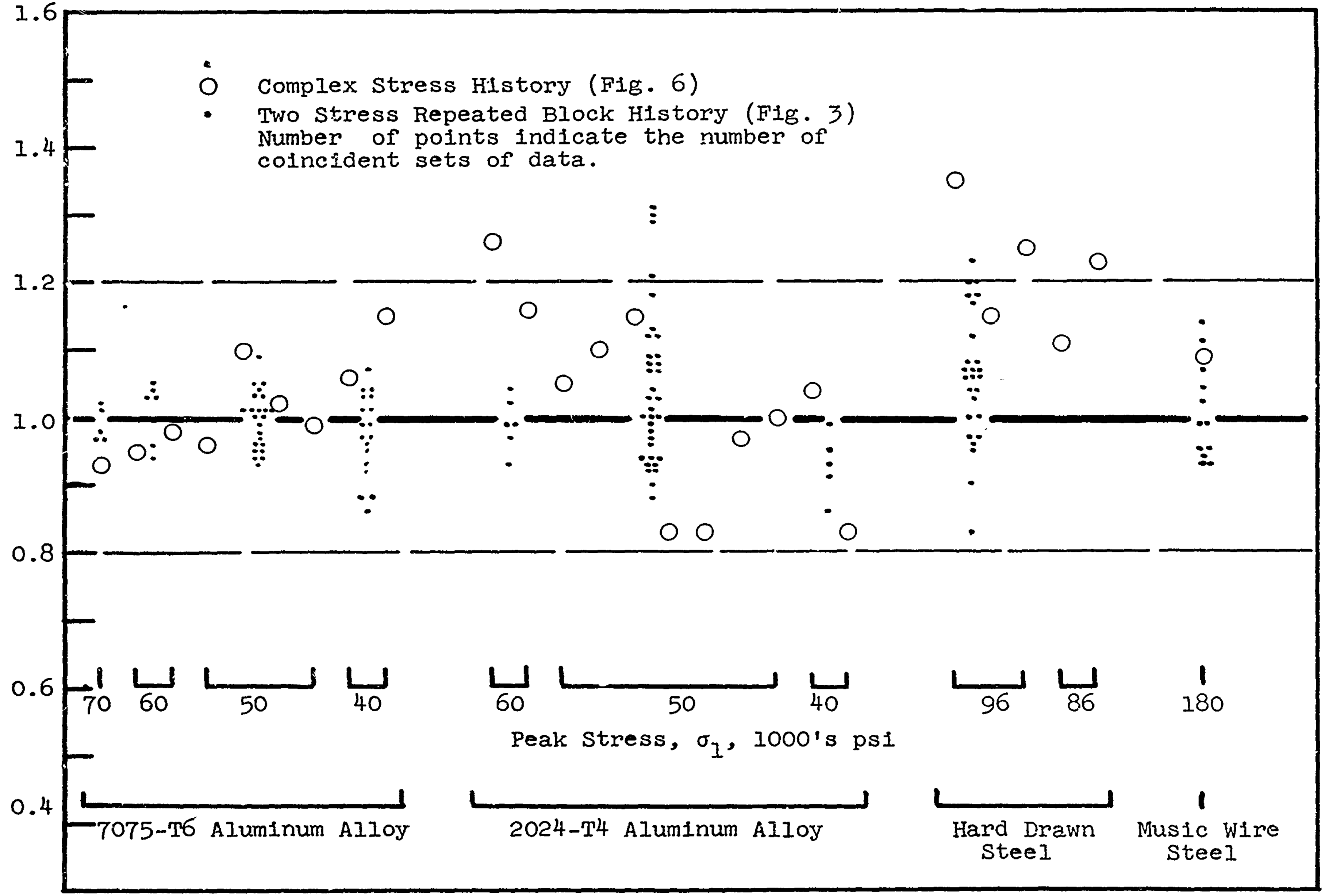


FIG. 8 COMPARISON OF ESTIMATED (EQ. 6) AND MEASURED VALUES OF FATIGUE LIFE FOR VARIOUS COMPLEX STRESS HISTORIES (11)

points are within the range of $\pm$ 10 per cent of the lif- predicted by Eq. 6. Within a range of $\pm$ 20 per cent, 22 out of the 26 points are included. Three of the points outside of this range are for a hard drawn steel wire which is known to be susceptible to strain aging; the long lives exhibited by some of the steel specimens were attributed to this phenomenon, which is not accounted for by Eq. 6. In addition to the open circles shown in Fig. 8, the two-stress repeated-block experiments are represented by small dots. In viewing the data as a whole it is encouraging that the great cluster of points fall within the region of $\pm$20 per cent which is exceptionally close when one considers the wide scatter to be expected in fatigue life of similar specimens.

It is of interest to compare Eq. 6 with the linear summation of cycle ratios hypothesis which may be rewritten in terms of the present notation as:

$$N_g = \frac{N_1}{\Sigma a_i \ (N_1/N_i)} \tag{7}$$

In this form the similarity with Eq. 6 is striking. The terms are identical except that the ratio in the denominator N_1/N_i has been replaced in Eq. 6 by the ratio $(\sigma_i/\sigma_1)^d$. Fig. 8 illustrates that Eq. 6 has achieved accuracy and reliability while maintaining simplicity of mathematical form. The advantage of Eq. 6 is evident when it is noted that cycles of stress considerably below the fatigue strength contribute damage if the previous stress history includes cycles of stress above the fatigue strength. Eq. 7 assumes that these cycles of low stress do no damage; however, the data of Fig. 4 indicate that this is not a correct assumption and that predictions based on the linear accumulation of cycle ratios may greatly overestimate the actual fatigue life.

A simple interpretation may be made by comparing the expressions in the denominators of Eqs. 6 and 7 with the conventional σ-N diagram shown in Fig. 9. The quantities N_i and N_1 represent the lives from constant amplitude experiments and correspond to points on this diagram for given stress levels. The upper portions of σ-N curves are frequently straight lines when plotted on logarithmic scales and may be represented by expressions of the form

$$({}^{\sigma_i}/_{\sigma_1})^{\delta} = {}^{N_1}/_{N_i} \tag{8}$$

The ratio of $(\sigma_i/\sigma_1)^d$ in Eq. 6 may be interpreted as defining a second straight line of slope 1/d (a modified σ-N curve) as illustrated by the dotted line in Fig. 9. This line simulates behavior of the modified structure of the material and the response to cycles

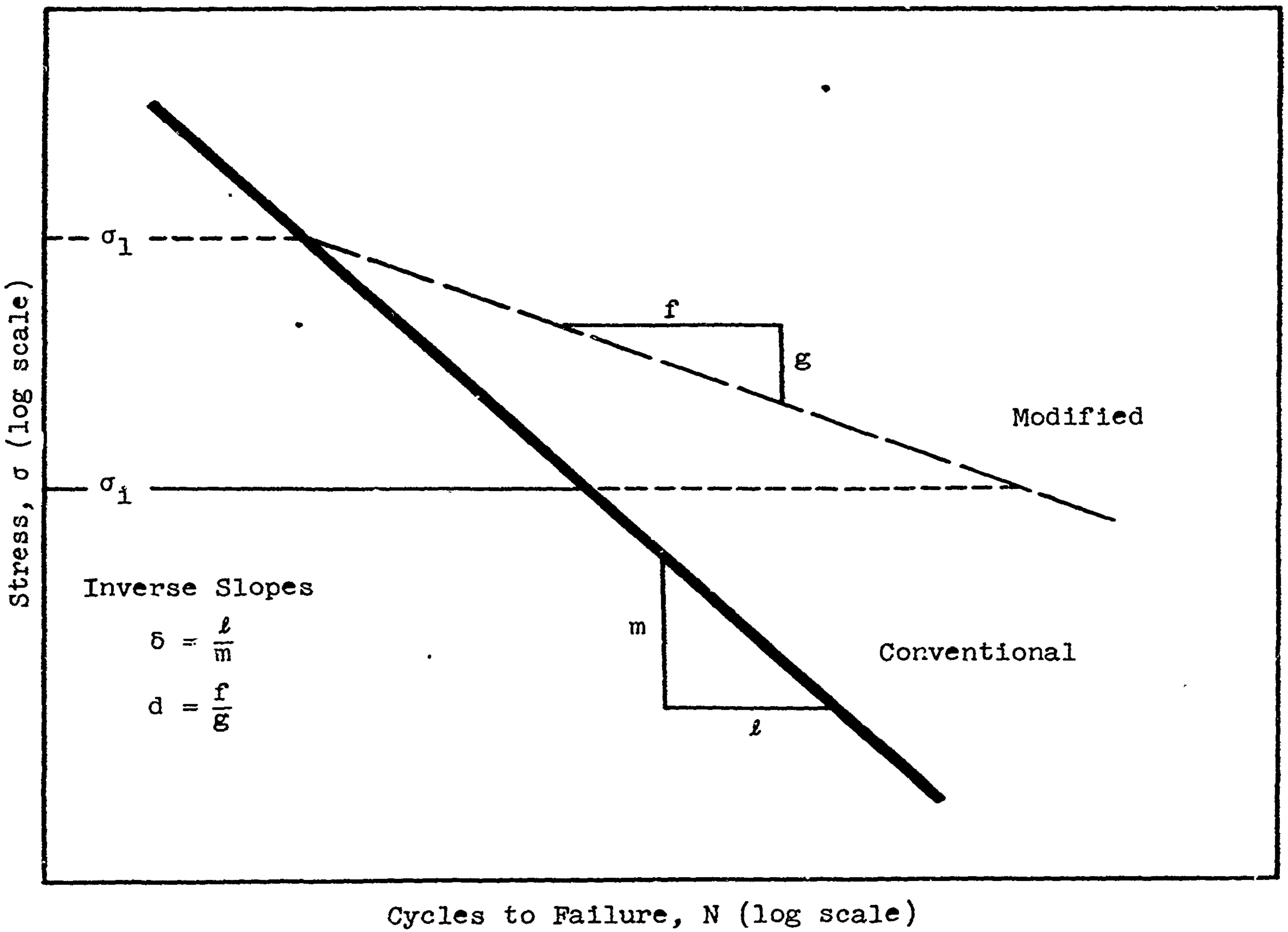


FIG. 9 σ-N DIAGRAM SHOWING THE RELATION BETWEEN THE CONVENTIONAL CURVE AND THE MODIFIED σ-N CURVE

of stress after application of peak stresses. This modified curve may be higher than, may coincide with, or may be lower than the conventional σ-N diagram, depending upon the shape of the member and the stress history.

Thus, the parameter "d" in Eq. 6 provides the necessary flexibility to estimate accurately the fatigue life associated with a random stress history without complicating the mathematical form of the equation. Appropriate values of "d" are known only for a few components; in specific cases the value of "d" may be determined from a simple two-stress repeated-block experiment of the type shown in Fig. 3 and a plot of the data as shown in Fig. 5. In some instances a value may be estimated by assuming "d" is approximately equal to the inverse slope of the σ-N curve (Fig. 9). The trends in the value of that have been observed for aluminum alloys are summarized in Table 1.

TABLE 1

Comparison of the Values of d and δ, the Inverse Slopes of the Modified and Conventional σ-N Curves, for Aluminum Alloys

Type of Specimen	Type of Loading	Stress Range Ratio, $\sigma_{min}/\sigma_{max}$	d vs δ	Reference
1. Standard small unnotched and notched rotating beam	Various multi-stress spectrums	-1	$d<\delta$	a, b, c, d
2. Unnotched rotating deflected wire struts, as drawn	Various multi-stress spectrums	-1	$d>\delta$	e
3. Axially loaded sheet specimens	Two-stress repeated-block			
(a) unnotched		-1	$d\leq\delta$	f
		0	$d\geq\delta$	f, g
(b) notched and riveted joints		0 to.5	$d>\delta$	g
4. Wing panels in bending	Various three level and multi level load spectrums		$d>\delta$	h, d, i

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In general, the mechanisms leading to the initiation and propagation of cracking in a composite structure are identical with those already discussed in connection with failure of material. The main differences are that the designer is confronted with more uncertainties in determining the critical locations and the significant ranges and states of stress to which his structure will be subjected.

In carrying over concepts of the fatigue mechanism to the behavior of a composite structure, several additional factors must be given careful consideration. In general, a composite fabricated structure will have shorter life than estimated on the basis of tests of simple specimens for the following reasons:

1. Fatigue cracks can be expected to originate at rivets, bolts, welded seams, or other discontinuities developed by methods of fastening or joining the components. These develop severe stress raising effects that are difficult to appraise in simple tests.

2. The multiplicity of stress raisers in composite structures may have the effect of multiplying together the strength reductions caused by each of two or more separate factors; for example, a rivet hole put through a wing spar in the neighborhood of a change in section such as a fillet, may effectively multiply the stress concentration factor for the hole by the stress concentration factor for the fillet.

3. Fabrication techniques often develop patterns of stresses locked into the completed structure that are not only difficult to measure but which may be altered by the first few cycles of imposed loading in service.

4. Because of fabrication, differences may exist from directional characteristics due to rolling, forging, extruding or to cold deformations in forming and fastening procedures.

5. Fretting action at joints and fasteners develops a severe reduction in strength that is difficult to simulate in a simple laboratory specimen.

6. Laboratory conditions seldom fully simulate the service environment; loading conditions, temperature and atmosphere, and the constraints from attachments cannot readily be reproduced in simple specimens. Unless operating conditions and failure are identical in the laboratory specimen with that observed in service, any comparisons of relative fatigue life are worthless.

7. Large sized components usually exhibit lower fatigue strength than small models even though geometrically similar in shape.

These factors so complicate the estimate of the response of the structure to repeated loads that an experienced engineer finds it difficult to place full confidence in design calculations. Consequently, it has become common practice to fatigue test critical components and entire structures to: (a) locate fatigue sensitive details, and (b) check the estimates of fatigue life. Constant amplitude experiments are often used in place of service simulating load tests. While many uncertainties are answered by such tests, the question of the service fatigue life remains unanswered. It is at this stage of the design that a reliable theory of cumulative fatigue damage is most useful.

As with small laboratory specimens, the response of the structure that is of primary interest occurs only after the structure has been subjected to a number of loads near peak amplitude. Available data suggest that the fatigue sensitive areas encountered in service may not coincide with fatigue sensitive areas found in constant amplitude low stress fatigue tests (12). In some cases fatigue sensitive areas found in service have correlated well with those found from constant amplitude experiments at high loads (13). It appears that this problem requires additional careful investigation. The available evidence suggests the importance of placing primary concern on "peak stress modified structures" instead of the original structure.

Attention will now be directed to presenting (by means of a practical example) a procedure for estimating the fatigue life of a structural component. The solution may be divided for convenience into three phases: (a) analysis of the service load spectrum, (b) determining the response of the member to repeated loads as measured by the σ-N diagram, and (c) prediction by means of a cumulative damage expression (Eq. 6) relating the spectrum of service loads to the conventional σ-N diagram.

Several years ago sufficient data on a fabricated steel structural component became available to the authors on each of these three phases along with actual data on service performance to provide interesting comparisons. Loads varying above zero in one direction occurred as shown schematically in Fig. 10. This or a similar sequence was repeated periodically. Laboratory fatigue data were obtained for zero to maximum loading of the structure. The load spectrum was analyzed by reducing each load to an "equivalent" zero to maximum load based on a (soderberg) mean stress-alternating stress diagram. From a plot similar to Fig. 7 it was found that a straight line adequately represented the sample of load history data. The load spectrum was tabulated as shown in Table 2, listing intervals of load amplitude and the frequency of occurrence of loads in each interval. From the load spectrum a peak value of load, T_1, was chosen*

* The peak value of load, T_1, was limited by service operation. In many cases the peak value of load is not limited. It appears that a value of peak load corresponding to a_1 from 0.001 to 0.01 may be appropriate.

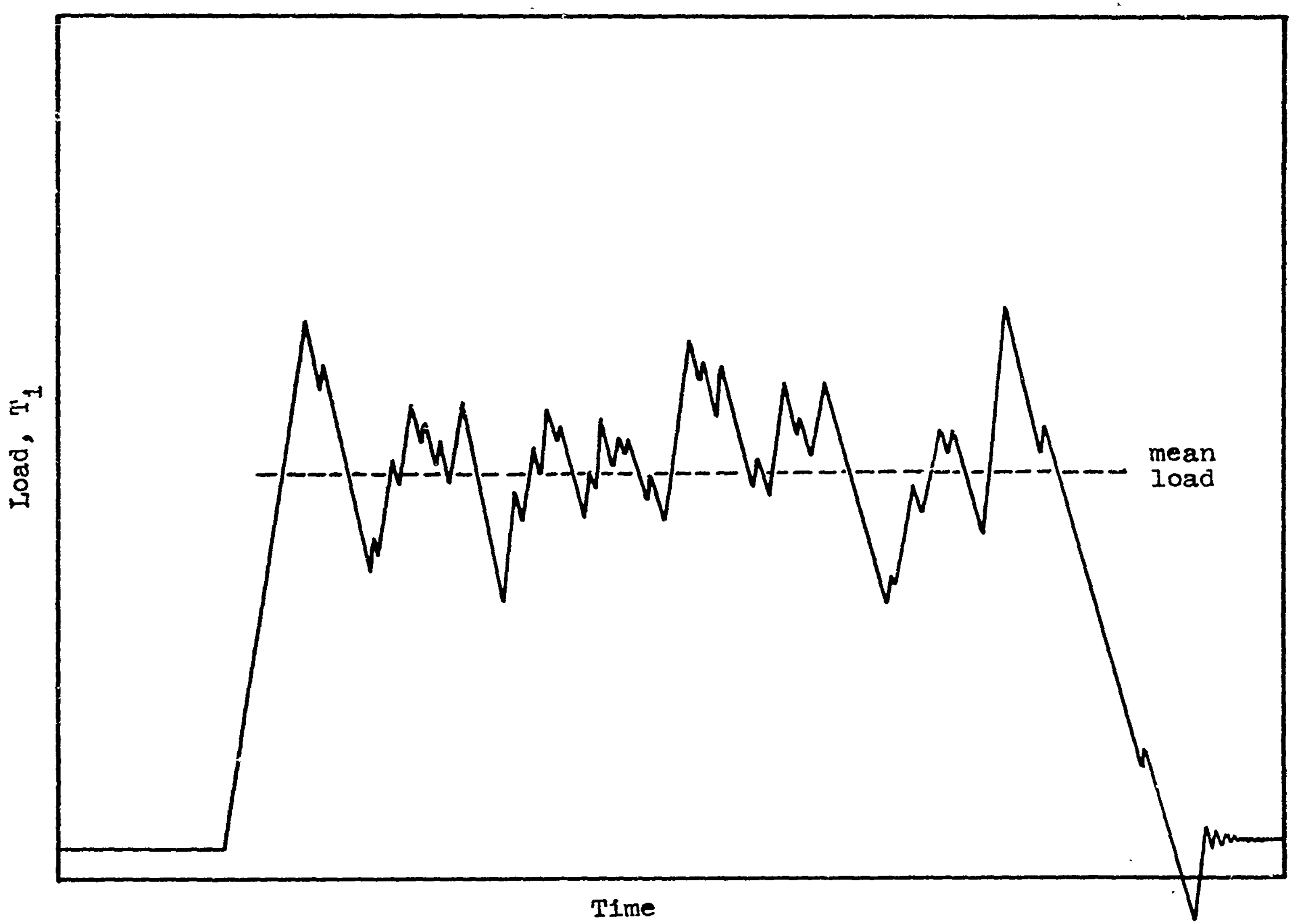

FIG. 10 SCHEMATIC LOAD HISTORY FOR ONE SEQUENCE OF OPERATION OF STRUCTURAL COMPONENT

and the life N_1 corresponding to constant amplitude test data at load T_1 was determined.

All necessary quantities thus were available except the value of d. The inverse of the slope of the σ-N diagram was 4.8 and a reasonable value of d was chosen as 4.0. Using these values in Eq. 6, the fatigue life was estimated at 5.08×10^5 cycles as shown in Table 2. The component operated in a series of approximately repeated sequences of the type shown in Fig. 10. Based on an estimate of 150 cycles per sequence and 60 sequences per day, the life in days was calculated as 56 days. Service records for several components that failed in service showed lives ranging from approximately 35 days to 70 days of operation. These components were installed consecutively in the same apparatus and thus represent average life times (not the shortest lives of a large group). Considering the many uncertainties involved in the analysis (including only a sampling of the load spectrum, incomplete σ-N data, and uncertainty as to the frequency of operation) the agreement between the estimated life and the observed lives was considered to be excellent. This example demonstrates that even with numerous uncertainties present, analysis plus the exercise of good judgment may provide estimates of life that can be accurate within a factor of less than 2.

TABLE 2

Computation of Average Fatigue Life of Structural Component Based on "Equivalent" Zero-to-a-Maximum Load Spectrum

$T_1 = 9000$ lb, $d = 4.0$

Load Increments lb	$T_{i_{mean}}$	T_i/T_1	$(T_i/T_1)^4$	a_i	$a_i(T_i/T_1)^4$
8800-8000	8400	0.93	0.76	0.004	.0030
8000-7200	7600	0.85	0.51	0.007	.0036
7200-6400	6800	0.76	0.33	0.014	.0046
6400-5600	6000	0.67	0.20	0.026	.0052
5600-4800	5200	0.58	0.11	0.048	.0053
4800-4000	4400	0.49	0.057	0.089	.0051
4000-3200	3600	0.40	0.026	0.164	.0043
3200-2400	2800	0.31	0.0094	0.307	.0029
2400-1600	2000	0.22	0.0024	0.184	.0004
1600-800	1200	0.13	0.0003	0.156	.0000
800-0	400	0.04	--	0	--
					.0344

$$\Sigma a_i \, (T_i/T_1)^d = 0.0344$$

$$N_g = \frac{N_1}{\Sigma a_i \, (T_i/T_1)^d} = \frac{N_1}{0.0344} = 29.1 \; N_1$$

$N_1 = 1.7 \times 10^4$ cycles from laboratory data.

Therefore, the predicted life, $N_g = 5.08 \times 10^5$ cycles

or 56 days at 9000 cycles per day.

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IMPORTANCE OF PROCESS CONTROL IN THE FATIGUE RESISTANCE OF STRUCTURAL MATERIALS

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ABSTRACT

Fatigue data for commercial structural alloys and for fabricated parts are almost always widely scattered and must be treated statistically for proper interpretation. The inconsistency of fatigue data is attributed, largely, to nonuniformities in the structure of the materials and in the fabricated parts as a result of improper process control. An example is given of how one process variable—top ingot discard—affects the quality and uniformity of low-carbon sheet steel. Additional examples show that variations in soldering and welding processes have pronounced effects upon the fatigue resistance of welded and of soldered joints. It is concluded that more research and development are needed in the field of process metallurgy to learn how optimum and uniform fatigue resistance can be obtained in structural alloys and in fabricated parts.

INTRODUCTION

It is a commonly accepted hypothesis nowadays that fatigue data are naturally scattered and that they must be analyzed statistically for correct evaluation of the results. This point was clearly stated by Mehl and his associates (1)[a] several years ago as follows: "Even the utmost care in the preparation of specimens and in the conduct of fatigue tests cannot minimize the scatter in fatigue life and the uncertainty in the fatigue limit beyond a point where statistical techniques are needed for proper interpretation of results."

Such nonreproducibility of fatigue data is a result, largely, of difficulties in controlling the homogeneity of materials during production. As a consequence, commercial structural materials do not have entirely uniform structures and properties. Fabricating varibles, as well as production variables, can also have marked effects on fatigue resistance. Grover, Gordon, and Jackson (2) warned of these effects as follows: "Data from laboratory tests may provide helpful guides, but not necessarily data for quantitative design, since in structural parts various fabrication effects, such as notches, cold work, a particular surface finish, heat-treating stresses, etc. —each meaning a departure from carefully polished specimens—may exist concurrently."

Experience has shown that fatigue properties are more sensitive to structural nonhomogeneity and to variations in fabricating techniques than are any other mechanical properties. For this reason, fatigue data tend to be more scattered than other types of mechanical-property data.

The great sensitivity of fatigue resistance to many types of structural, physical, and mechanical variations emphasizes the importance of good process control to minimize these variations during both the production and the fabrication of structural materials. The scatter that is almost invariably obtained in fatigue data shows that the process-control techniques employed at present are not adequate to produce materials of uniform fatigue quality.

KNOWN FACTS

As a result of previous research and testing in the field of fatigue, much is known about the effects of different process-induced materials variables. It is generally accepted, for example, that surface roughness, nonmetallic inclusions in metals, surface softening (decarburization, alcladding), hydrogen embrittlement, and stress concentrations are detrimental to fatigue strength (2, 3, 4, 5). On the other hand, strain hardening, surface hardening (carburization, nitriding), and surface compressive stresses (shot peening, coining) are generally beneficial

a. The numbers in parentheses refer to the bibliography.

to fatigue properties (2, 3, 4, 6) Residual stresses and small prestrain have inconsistent effects on fatigue life—sometimes increasing, sometimes decreasing—depending upon the magnitude and the direction of these stresses and strains (3, 7).

In a number of careful studies of the factors that cause scatter in fatigue data, it has been found that, in metals, nonmetallic inclusions have the greatest influence (5, 8, 9, 10). Epremain and Mehl (8), for example, made the following unequivocal statement: "Of the many factors that can affect the statistical behavior of fatigue properties, inclusions are the most important." Stulen (9) concluded that "In carefully prepared specimens, the origin of failure is almost always at a microscopic nonmetallic inclusion which is open to the surface or slightly subsurface." The mechanism of the effects of nonmetallic inclusions in high-strength steel were studied by Stulen, Cummings, and Schulte (5, 10). They found that in the low-stress long-life range, a single large inclusion nucleates fracture. At higher stresses, cracks nucleate at and propagate from smaller inclusions. At very high stresses, large inclusions are of little importance, and fracture is caused by the joining of numerous little cracks that are initiated and propagated simultaneously. In discussing this work (10), A. M. Askoy presented the following comparisons for air-melted and vacuum-melted 4340 steel all heat treated to 230 ksi ultimate strength level:

a. Vacuum, longitudinal—114 ksi endurance limit
b. Vacuum, transverse—105 ksi endurance limit
c. Air, longitudinal—90 ksi endurance limit

He attributed the superior endurance limits of the vacuum-melted material to its lower content of nonmetallic inclusions.

PROCESS CONTROL IN PRODUCTION

All or most of the following processes are generally employed in the production of wrought steel: melting, pouring, soaking, slabbing, cropping, reheating, hot rolling, coiling, pickling, cold rolling, annealing, and heat treating. Most of the research that has been carried out on the control of the properties of steel has been devoted to the first and the last processes—melting and heat treating. Experience indicates that the properties of steel are influenced by small variations in the thermal and mechanical processes between the melting furnace and final heat treatment. Although the intermediate processes are subject to reasonably close control, very little quantitative information is available on the effects that controlled variations in these processes have on the final properties of steel.

In order to provide such information, Southern Research Institute, under the sponsorship of a large steel company, has initiated an investigation of the effects

of a number of intermediate process variables on the quality of low-carbon sheet steel. Although notched tensile strength rather than fatigue strength is used as a measure of quality in this work, some of the initial results illustrate the need for a knowledge of process control if both optimum efficiency and optimum quality are to be attained. Notched tensile strength is used because under certain conditions it has been found to provide a quantitative measure of both strength and formability.

It had been standard practice in many steel plants to crop off about 30% or more of rimmed ingots and to use only the bottom 70% or less for their high-quality products. The reasoning is that the top of the ingot is low in quality due to shrinkage and gas porosity as well as to segregated concentrations of sulfur, phosphorus, and inclusions. Since no quantitative data were available to justify this procedure, forty samples of hot-rolled strips were obtained. Twenty of the samples represented ten different ingots from a production run in which the top 30% of the ingots had been discarded; the other twenty samples represented ten ingots from a production run in which only 8% of each ingot had been discarded. Visual comparisons of the notched tensile properties of the two groups of samples are given in Figure 1, which shows the normal frequency distribution curves for these properties. Also shown in Figure 1 is the T value for the data, which is a statistical measure of the probability that there is a significant difference between the two groups of data—the higher the T value, the greater the probability of a significant difference. The data in Figure 1 show that the different degrees of top ingot discard had no significant effect on the properties of the hot-rolled strip. These experiments show that no improvement in the quality of low-carbon sheet steel is obtained by discarding 30% rather than 8% of the ingot.

Further investigation, however, showed that it would be wrong to conclude that reasonable uniformity is obtained in the properties of the sheet steel regardless of the amount of top ingot discard. Figure 2 shows a comparison of the distribution of the notched ultimate strengths of the heads and tails of 20 different hot-rolled-strip coils. The head corresponds to the top of the ingots after 30% top ingot discard, and the tail to the bottom of the ingots from which the coils were produced. The tail samples had significantly higher strength than the heads as shown visually and by the large T value in Figure 2. It is not known, as yet, whether the inferior strength of the head is a result of nonhomogeneity that could be eliminated by a greater amount of top ingot discard, or whether some other process variables in the reheating, hot-rolling, and coiling operations are responsible for the difference.

This information on low-carbon steel sheet is presented primarily to illustrate the need for a better knowledge of the effects of processing variables on the quality of structural materials. Since this type of alloy is produced in greater quantities than any other metal, it is likely that process control for its production

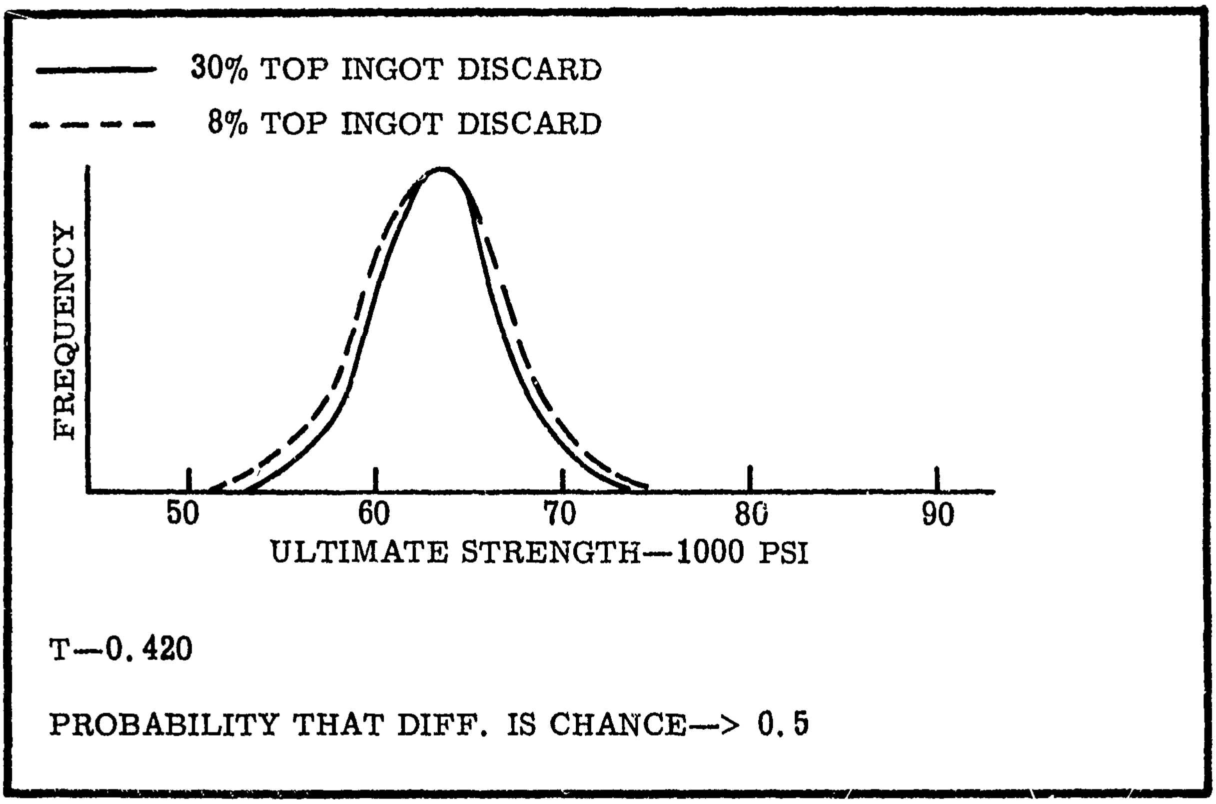

Figure 1. Normal frequency distribution curves for notched tensile strength of hot-rolled low-carbon steel strip with different amounts of top ingot discard.

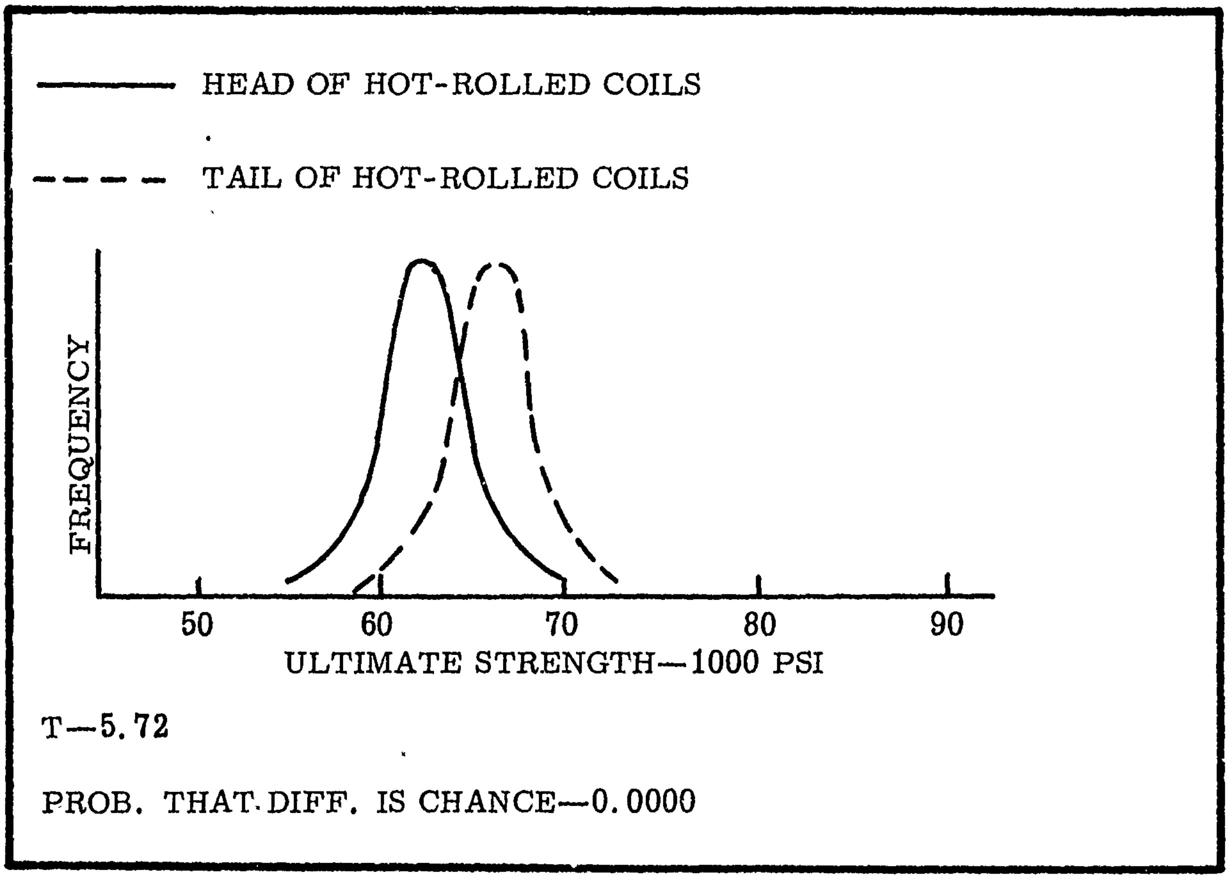

Figure 2. Normal frequency distribution curves for notched tensile strength of hot-rolled low-carbon steel strip showing difference between heads and tails of coils after 30% top ingot discard.

is as highly developed as for any metal. If other production processes intended to improve the uniformity of materials are as ineffective as the concept of 30% top ingot discard discussed above, it is no wonder that fatigue data are widely scattered and inconsistent.

PROCESS CONTROL IN FABRICATION

The degree and type of control applied in each of the many processes by which structural materials are fabricated—machining, welding, soldering, brazing, riveting, drawing, pressing, spinning—have a pronounced effect upon the fatigue properties of the finished part. Two examples—one involving soldered joints and the other involving welded joints—will be used to illustrate this fact.

In an investigation of the fatigue characteristics of soldered joints being carried out for Army Ballistic Missile Agency at Southern Research Institute, it was desired to test straight soldered connections of two copper wires, the wires being spaced about 1/4-in. apart by the solder. Various methods for producing the soldered joints were investigated. Conventional hand-soldering methods were entirely unsatisfactory for producing test samples because of the extremely inconsistent properties of such joints. In the first attempt to make more reproducible joints, two fluxed copper wires were placed in a grooved mold with the ends spaced 1/4 to 3/8 in. apart. A slug of solid solder alloy was placed in the groove between the copper wires. After the mold was closed, the entire assembly was heated to 50° F above the liquidus temperature of the solder and then cooled to room temperature while slight inward pressure was applied to the two wires. The open circles in Figure 3 show the fatigue life as a function of bending amplitude of joints made in this manner with lead-tin solder. The data are extremely scattered, the number of cycles to failure varying by as much as a factor of 100 at constant amplitude.

Further studies with the same solder-joint mold showed that marked improvements in both the level and the uniformity of the fatigue data could be obtained with relatively minor variations in the soldering process. Optimum results were obtained when the flux was eliminated and the ends of the wires were pretinned—coated with a thin layer of solder by dipping into a molten bath—before the molding operation. It was also found that thorough cleaning of the mold, the wires, and the solder before each soldering operation is essential. Further beneficial effects were obtained by orienting the mold in the vertical rather than the horizontal position and by reducing the maximum temperature to the liquidus temperature of the solder. The improvement in fatigue properties obtained with these processing techniques are illustrated in Figure 3 by the black-diamond points. The improved soldering process resulted, roughly, in a three-fold improvement in endurance limit and in a reduction in the maximum scatter of the data to a factor of ten at constant amplitude.

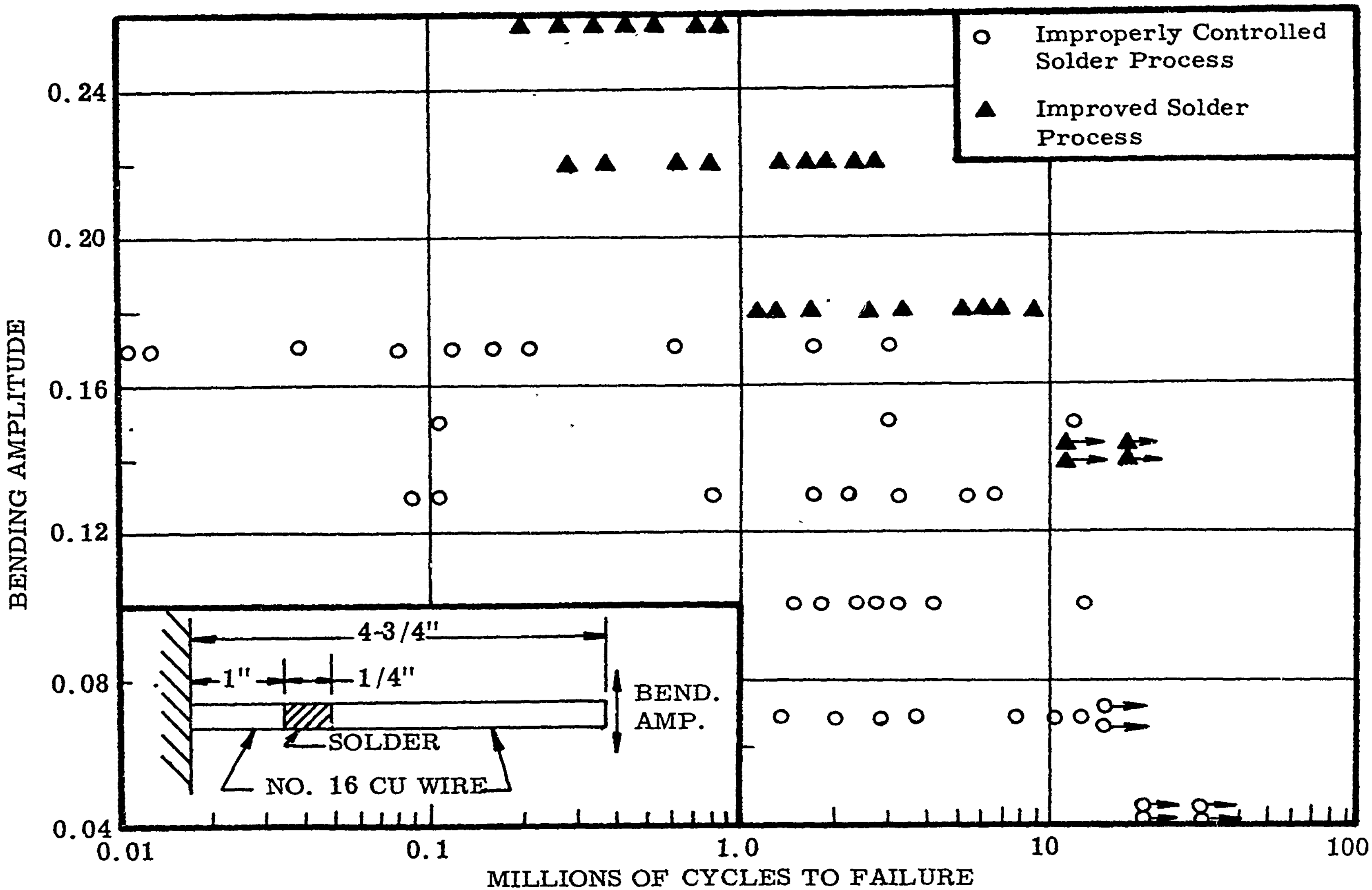

Figure 3. Improvement in the solder process increases fatigue life in lead-tin solder joints.

Figure 4 presents some fatigue data for butt-welded 5052-H32 aluminum-alloy sheet as compared with the unwelded material. These data are rather inconclusive regarding the merits of the hydrogen welding atmosphere as opposed to the inert atmosphere, of the 4043 welding rod as opposed to the 5053 rod, and of the weld bead ground flush with the surface of the sheet as opposed to the in-tact weld bead. They do show, however, the importance of the use of the proper welding process if optimum fatigue strength is desired. The optimum welding process, in the data illustrated, resulted in a decrease of less than 10% in the endurance limit below that of the welded material, whereas the most detrimental process caused a decrease of almost 50% in the endurance limit.

CONCLUSIONS

1. Even with the most carefully prepared specimens and controlled tests, fatigue data for commercial materials are widely scattered, largely as a result of nonhomogeneity of the materials.

2. The inconsistency of fatigue resistance is even more serious in fabricated parts because improper control of fabrication processes can result in further scattering of fatigue data.

3. Proper control of the production processes and of the fabrication processes is required to obtain homogeneous structural materials and fabricated parts with optimum fatigue properties.

4. Considerably more research and development are needed on many production and fabricating processes to determine how these processes should be controlled to obtain optimum and uniform fatigue resistance.

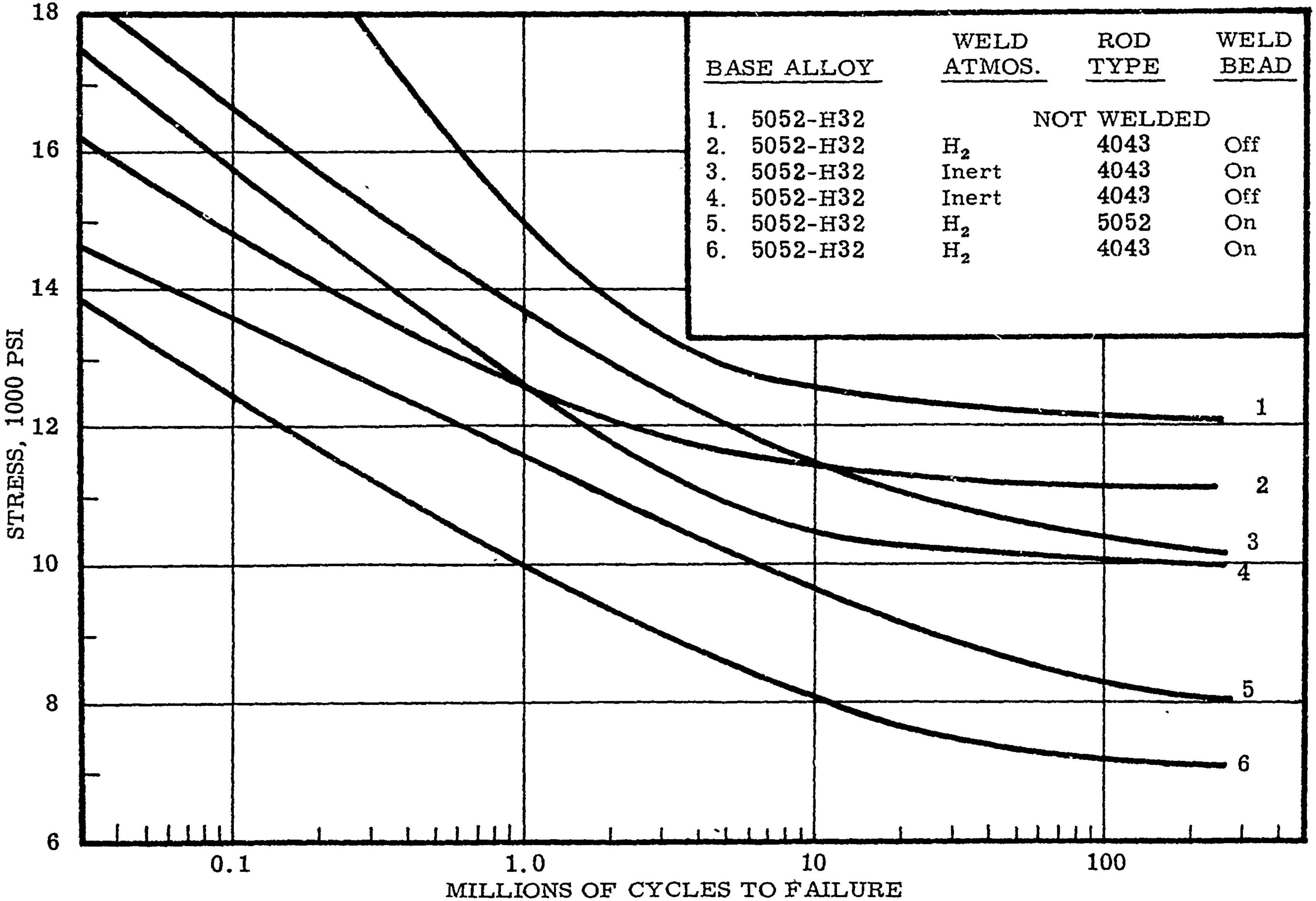

Figure 4. Variations in welding process produce changes in fatigue strength of welded 5052-H32 aluminum-alloy sheet.

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OPTIMUM SELECTION OF MATERIALS

By

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ABSTRACT

Under simple static loads, minimum structural weight can generally be attained by choosing, from the available materials, that one that has the highest strength-weight ratio. When maximum service life is desired, however, it is not always sufficient to choose the material that shows the highest fatigue strength in a simple laboratory test.

In a service situation, the factors that determine the magnitude of peak stresses often have greater influence on service life than the relative behavior of materials under equal stresses. Among these factors are the kind and location of stress raisers, the ability of the material to relieve peak stresses, its ability to develop and retain beneficial residual stresses, and damping characteristics.

Ultimately, it may be possible to account for all these factors in the design calculations. At present, however, this is generally not possible; as a result, a great deal of effort is called for to devise and conduct tests in such a manner that the pertinent conditions of service are fully simulated.

INTRODUCTION

Under simple static loads, minimum structural weight can generally be attained by choosing, from the available materials, that one that has the highest strength-weight ratio. The strength-weight ratio corresponding to a given loading condition can be determined with sufficient accuracy from relatively simple formulas.

When fatigue is critical, however, the actual performance of a part in service is a complex result of a multitude of factors, few of which are amenable, at the present state of the art, to mathematical treatment. It has generally been held that results of laboratory fatigue tests are of extremely limited utility for direct use in design calculations. Instead, they have been used to establish design limits and to make selections among competitive materials.

Up to a point, selection of materials on the basis of today's laboratory fatigue tests is a satisfactory procedure. However, as performance requirements continue to increase and safety margins are reduced, we find that what once was a satisfactory design will no longer suffice, and nothing but the best possible can be considered adequate. It behooves us, then, to begin with the selection of a material that is the best possible for the job at hand.

This is virgin ground. A review of the situation reveals far more questions than answers. For many years it has seemed that fundamental studies in the true nature of fatigue have lagged behind the empirical approach aimed at defining the

engineering parameters. Now it would appear, the "shoe is on the other foot." In spite of reams of data on an incredibly wide variety of materials and ramifications of each, we know very little about how to predict relative performances among closely competitive materials on the basis of simplified laboratory tests.

The only complete measure of performance in actual service that we now have at our disposal is performance in actual service. This has been and still is the basis for many engineering designs. But in an area where designs are well on their way toward obsolescence by the time they are first put into service, the inadequacies of this procedure are obvious.

It is essential that we learn how to simulate service, whether it be done by specialized testing or by understanding and utilizing the necessary conversions from results of the established tests. This paper is an attempt to delineate some of the problems we face and some of the factors we must consider, and to offer an occasional suggestion on methods for handling them.

THE ORIGIN OF FATIGUE STRESSES

It may seem trite, in a paper purportedly dealing with problems of engineering design, to raise a question regarding the source of a stress. However, it is a not uncommon experience to measure, calculate, or otherwise determine peak stresses in a part, and impose limitations intended to keep these stresses below the lowest level that can precipitate a fatigue failure, only to have the part fail in service by fatigue.

There is an important principle - admittedly a truism, but often overlooked - that applies to this situation: no part can fail in fatigue at a stress lower than that necessary to cause fatigue failure in the material of which it is made. From this principle may be extracted the equally obvious conclusion that if fatigue failure occurs, then by some means the stresses were generally or locally increased to the level necessary to cause that failure.

The material used may have a significant effect on the magnitude of the peak stresses developed under a given loading situation, but the possible effects are conditioned by the manner in which load is introduced. Consequently, the first job is to arrive at a satisfactory analysis of the loading and general stress paths.

To amplify this point, consider first a bar of uniform cross-section subjected to an axial tensile loading. The average stress can have but one value, which is the load divided by the area. This value is independent of the material, while internal variations from the average value are dependent entirely upon the material's internal structure and residual-stress condition. If, on the other hand, the bar is stretched so that its length is increased by, say, two-tenths of one percent, then the average stress is dependent upon the gross stress-strain characteristics of the material, and variations are dependent upon internal conditions.

Now suppose the bar is in pure bending. The stress-strain characteristics of the material will influence both average and peak stress levels regardless of whether load or deflection is the fixed independent variable. It is worth noting, however, that if load is fixed, an increase in section size will tend to lower the stress, whereas if deflection is fixed, the stress will be reduced by decreasing the section thickness.

Perhaps the most frequent situation in complex parts is that in which neither load nor deflection is fixed, but each influences the other. As an example, consider an aircraft wheel with the tire pressing outward against the rim flange. If there is considerable "give" in the flange - whether by virtue of shape, low modulus of elasticity, or low proportional limit - the load will tend to redistribute and spread over a wider contact area. If the flange is essentially rigid, the tendency will be for the load to remain concentrated at the point of initial contact.

While a test program aimed at optimum material selection may not achieve a complete separation of design influences from material influences, it is essential that the difference be recognized so that the suitability of the program to screen for any given area of application can be recognized.

Some characteristics of materials that should influence selection are dealt with in the following sections.

NOTCH SENSITIVITY AND ACCOMMODATION

The quantity - or quality - known as "notch sensitivity," while useful in some analyses, is generally overused and frequently misused or used in misleading fashion. A very low "notch sensitivity" may be recorded for a material for no other reason than that it was never possible to test it in an un-notched condition. That is, the presumed "un-notched" samples contained internal or surface stress raisers that could not be eliminated. If their effect is of such a nature as not to be additive when the material is deliberately notched, it is to be expected that there will be little change in fatigue strength resulting from the added notch. But it would be completely erroneous to infer that the material was insensitive to notches.

It is probably safe to say that, at sufficiently low stresses, all materials are fully sensitive to notches, with the only limitation currently suspected being that imposed when the notch dimensions are reduced to the same order of magnitude as the microstructural dimensions of the material. Experience with field failures also leads one to the conclusion that they are almost invariably precipitated by a notch of some kind. (The term "notch" is here meant to include any stress raiser supplemental to the load itself.)

If these two statements are accepted, then it follows that the quantity of real significance is the absolute fatigue strength in the presence of the notch, regardless of how this relates to strength without the notch. The "notched fatigue strength" of a material is influenced by a very important, but completely undefined quality I shall call "accommodation." Perhaps the most important single goal we might set in the area of material selection is the attainment of a full understanding of accommodation and the development of some quantitative measure for it.

Accommodation can be described in terms of its effects and we can go so far as to state, in general terms, certain characteristics of a material that would lead to good accommodation. But a lot more definition is needed.

In crude terms, accommodation is simply the ability to "roll with the punch." In a military situation, with lines drawn up face to face, if the enemy puts forth a concentrated thrust at one point, it is generally good tactics to withdraw and allow a partial penetration of the lines. This exposes his forces on three sides to a firepower that can achieve high total effect with relatively low unit intensity.

In a structural material that has good accommodation properties, the same kind of resistance is developed to loads. If a stress tends to be high at one point, the material yields and permits a spreading of the load over a broader area. On this broadened base, the same amount of load can be resisted by a lowered value of maximum unit stress.

What gives a material this property of accommodation? Clearly, it must possess ductility. Moreover, it must at some points exceed its elastic limit if redistribution is to occur under an unchanging form of load application. As a boundary condition, the fatigue strength must be high enough to permit these characteristics to exert their influence.

It remains to determine how much ductility is needed, and at what stress levels the ductility is actually useful. It is doubtful that the value of residual elongation - or reduction of area - after tensile fracture has any real significance here. Perhaps the ratio of yield strength to tensile strength can provide some kind of "modulus of accommodation." At present, however, it would seem that the best way to make this quantity a part of the selection criterion is to conduct tests in such a manner that the res .s are dependent upon it. This will involve reproducing in some fashion both the stress raisers encountered in service, and the general distribution of high- and low-stressed material under or around these stress raisers.

RESIDUAL STRESSES

Residual stresses, whether deliberately or accidentally produced, will significantly influence behavior under fatigue loading. Thus the ability to develop and retain such stresses is an important characteristic of a material. Capacity for development of beneficial stresses may well be enhanced by the same characteristics that provide accommodation, since localized yielding is a prerequisite.

Capacity to retain a beneficial residual stress under service conditions is obviously as important a characteristic as the ability to develop it in the first place. This presumably sets some kind of upper limit on the amount of plastic-flow tendency that should be present; again the defining quantities for this limit are still unspecified. Until they can be specified, it appears the selection testing will have to be done in such a way that service residual-stress effects are essentially duplicated.

With the number of different techniques available for inducing residual surface compressive stresses, the method itself becomes a subject for optimum selection along with the materials in question. The service requirements must be carefully appraised, and the situation duplicated as nearly as possible in all its significant facets. Presence of a mean load, for example, could be expected to be of great significance, as it could decidedly affect a given material's capacity to retain residual stresses. Operating temperatures too will have an obvious influence, and will have to be accounted for.

Clearly, it is going to require considerable ingenuity to devise screening tests that truly represent relative potential serviceabilities, yet are not prohibitive in size and complexity.

VIBRATIONS AND DAMPING CAPACITY

A rather special fatigue situation occurs when the stresses are present because of forced vibration of the part. Not only the magnitude of the driving force, but also its frequency and the geometry of the part will affect the stress level. If the driving force is not constant, but of an intermittent nature, it is clear that the damping capacity of the material can influence the number of cycles of near-maximum stress that are experienced.

It is entirely possible, in this situation, that a material of relatively low fatigue strength but high damping capacity would outlast one of higher strength but less damping capacity. If this is so, then a comparison of fatigue strengths determined in the usual fashion will represent a very poor basis for making a selection.

It follows once again that a test intended to be the basis for arriving at an optimum selection of material must incorporate all those service factors that cannot be handled by analysis.

MATERIAL SAMPLING

Up to this point, the emphasis herein has been on design of a test to provide an adequate simulation of service conditions. All the effort expended in that direction, however, will be wasted if the test is not applied to material representative of what will be put into service. There is no intent here to imply that anyone would test steel bars in order to appraise a magnesium alloy - there are far more subtle effects that can enter to make a sample completely unrepresentative of the actual part.

Metal fabricating techniques are such that size and shape influence properties. In a forging, for example, the lines of flow are presumed to exert an influence on fatigue properties. Since it is seldom possible to make stress flow follow the identical paths of material flow, and since flow characteristics vary from shape to shape, it would seem the only sure way to obtain specimens whose properties represent those of the actual part is to take the specimens from the critical area of the part. Furthermore, not just the location, but the orientation as well, must be maintained.

Castings are subject to similar variations from part to part and from location to location; the same sampling requirements will apply. Other metal forms are perhaps less subject to variation, but in every case steps must be taken to ensure that the material tested truly represents the material that is to be put into service.

CONCLUSIONS

The need to achieve a truly optimum material selection for fatigue service becomes increasingly urgent with each new development in service requirements. Designs become more and more precise, and once-comfortable margins of safety become thinner and thinner cushions against failure.

There are a multitude of service factors that are not now amenable to precise analysis, and effects of loading, geometry, and material are often inseparably intertwined. Under such circumstances, we are faced with two major tasks.

Methods of analysis must be improved so that increasing numbers of the service factors can be handled by analytical means. In the meantime, all such factors as cannot be so handled will have to be directly represented in the tests that are to form a basis for selection. A good prediction of service capabilities will remain a chimera as long as any of the influential conditions are not accounted for by one means or the other.

With respect specifically to the experimental phases of selection, it will be an interesting test of our ingenuity to see how well we can represent all the significant factors while avoiding the prohibitively costly expedient of actually duplicating the full service environment.

EFFECT OF MATERIAL PROPERTY VARIATIONS
ON FATIGUE

By

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ABSTRACT

This paper discusses the effect of several material properties on the fatigue strength of alloys. Metal fatigue depends not only on the "bulk" properties of the metal but also on the variation of properties at the microscopic and submicroscopic levels. The variations of the fatigue strength as a function of the tensile strength are shown to be primarily dependent on the size and distributions of the largest inhomogeneities, at least in the long-life range of stresses. Experimental heats of special steels have been produced that show superior fatigue strengths. Possible relations between the fatigue strength of an alloy and its microstructure, chemistry, heat-treatment and ductility or toughness are discussed.

Some observations on crack propagation and the critical crack size are presented. A relation between the rate of crack propagation, the critical crack size and certain other material properties is suggested.

INTRODUCTION

The fatigue strength or the fatigue life of a manufactured part in service depends on its design details, the manufacturing processes employed, inspection procedures, service stresses and environments,

maintenance of the part in service, and the material properties. The satisfactory service life of a part is like a chain, it being no better than its weakest link. For example, the lack of proper attention to design details can readily mask the advantages of superior material properties and manufacturing processes. The use of metals having higher fatigue strengths may be overshadowed by weaknesses in other links in the chain. In this paper only the properties of structural metals in their relation to the fatigue strength will be considered.

One of the most outstanding and peculiar facts in the fatigue of metals has been the singular lack of success of past investigators in obtaining any moderately precise and general correlation between the fatigue strength of a metal and its other mechanical properties. Even the cause or causes for this lack of correlation have not been entirely understood. As recently as 1953, Grover et al. (1) said ". . . . attempts to correlate fatigue life with these (mechanical) properties have failed to show any correlation so far". Indeed there is the basic question whether fatigue strength is a completely independent property of a material unrelated to its other physical properties. However, within recent years some facts have been uncovered that may be significant in establishing some of the links between the fatigue properties and other properties. It is the purpose of this paper to discuss some of these trends, although it should be recognized that only a start in the understanding of this general problem has been made.

One characteristic of the fatigue of metals is the relatively large amount of scatter in the fatigue lives of supposedly identical parts or specimens tested at one stress level. This scatter occurs even when extreme care is used in the melting and manufacturing processes and in the preparation and testing of the specimens. Because of the lack of appreciation of this fact investigators have not always used a sufficient number of specimens in their tests so that positive and quantitative statements on observed trends could be made. Furthermore, the effects of important variables on fatigue were sometimes masked by this scatter in fatigue life. However, within the last decade there has been an increasing number of investigators (for example, 2-8) who have planned, conducted and evaluated fatigue programs using statistical procedures. Statistical techniques have been found to be mandatory in determining fundamental relations governing fatigue failures such as cumulative damage, crack propagation, nucleation, etc. Some effects such as notch sensitivity and size effect can be explained, at least in part, by statistical effects (6). The lack of appreciation of the statistical nature of fatigue has probably been one of the deterrents in establishing some of the fundamental relations in fatigue. To promote better understanding of this basic characteristic of fatigue, the ASTM (9) recently published a guide for planning, conducting and analyzing fatigue tests using statistical techniques.

One major difficulty that arises when the fatigue strength of a metal is correlated with any other single property of the metal is that the fatigue strength may also be dependent on several other properties. Unfortunately, because of this multi-variable aspect of fatigue and the difficulty of changing one bulk property of the metal without affecting others, a direct comparison of the fatigue strength with another variable does not usually show a high degree of correlation.

THE RELATION OF FATIGUE LIMIT TO TENSILE STRENGTH

For many years, investigators have recognized that there is a vague trend of the fatigue limit or long-life fatigue strengths of alloys of a certain type with their corresponding tensile strengths. For example, in figure 1 the fatigue strengths (500×10^6 cycles) of various wrought aluminum alloys have been plotted as a function of the corresponding tensile strengths. In this plot the change in tensile strength was obtained by changes in the alloy content as well as by changes in the heat treatment and degree of work-hardening. It is known that when the tensile strength of an aluminum alloy is greater than about 40,000 psi, there is relatively little increase in its long-life fatigue strength.

When the fatigue strengths (10^7 cycles) of a given type of steel are plotted against the various tensile strengths obtained by varying the tempering temperature, the resulting curve is usually one of the three shapes (marked A, B and C) in figure 2. In all cases, the ratio of the fatigue limit to the ultimate tensile strength decreases with increasing tensile strength. In many alloys, the maximum fatigue limit occurs at some intermediate tensile strength and then decreases with further increase in tensile strength as shown by curve C.

Figure 3 shows the average fatigue strength of one heat of air-melted electric-furnace SAE 4340 as a function of its tensile strength. A large number of rotating-beam specimens was used to establish this curve. All failures at the higher hardness levels were found to have originated at small inclusions. By changing the melting practice (air-melted) in one special heat, it has been found possible to increase the fatigue limit at the highest hardness level by 54%. This has been plotted as point A in figure 3. The change in the size and type of inclusion was found to be responsible for this substantial improvement in fatigue strength. This improvement was obtained by the air-melt electric-furnace process.

Here, then, is a case where a substantial difference in fatigue strength was found between two heats of steels having the same chemistry and hardness. It will be shown that this difference is caused by a change in a microscopic characteristic that is not readily detectable by any of the conventional bulk properties of the material. This example tends to minimize the possibility of correlating the fatigue strength of a metal with its tensile strength alone.

INTERNAL STRESS RAISERS

There are many types of notches and stress raisers in a manufactured part that may be present and decrease the ability of the part to resist fatigue loads. The first type of notch is that which may be present in the design itself. Holes, sharp fillets, rapid changes in section, threads, and keyways are well-known examples of design details that can cause a substantial loss in fatigue strength. Manufacturing defects such as rough machined surfaces, decarburization, inspector's stamp, lack of blending of sharp corners, improper heat-treatment, assembly or residual stresses, overheating in grinding and scratches are some of the many conditions that can cause large losses of fatigue strength in a manufactured part. In the service and maintenance of a part, there are many additional factors that can cause loss of strength such as erosion, corrosion, improper assembly, fretting, etc.

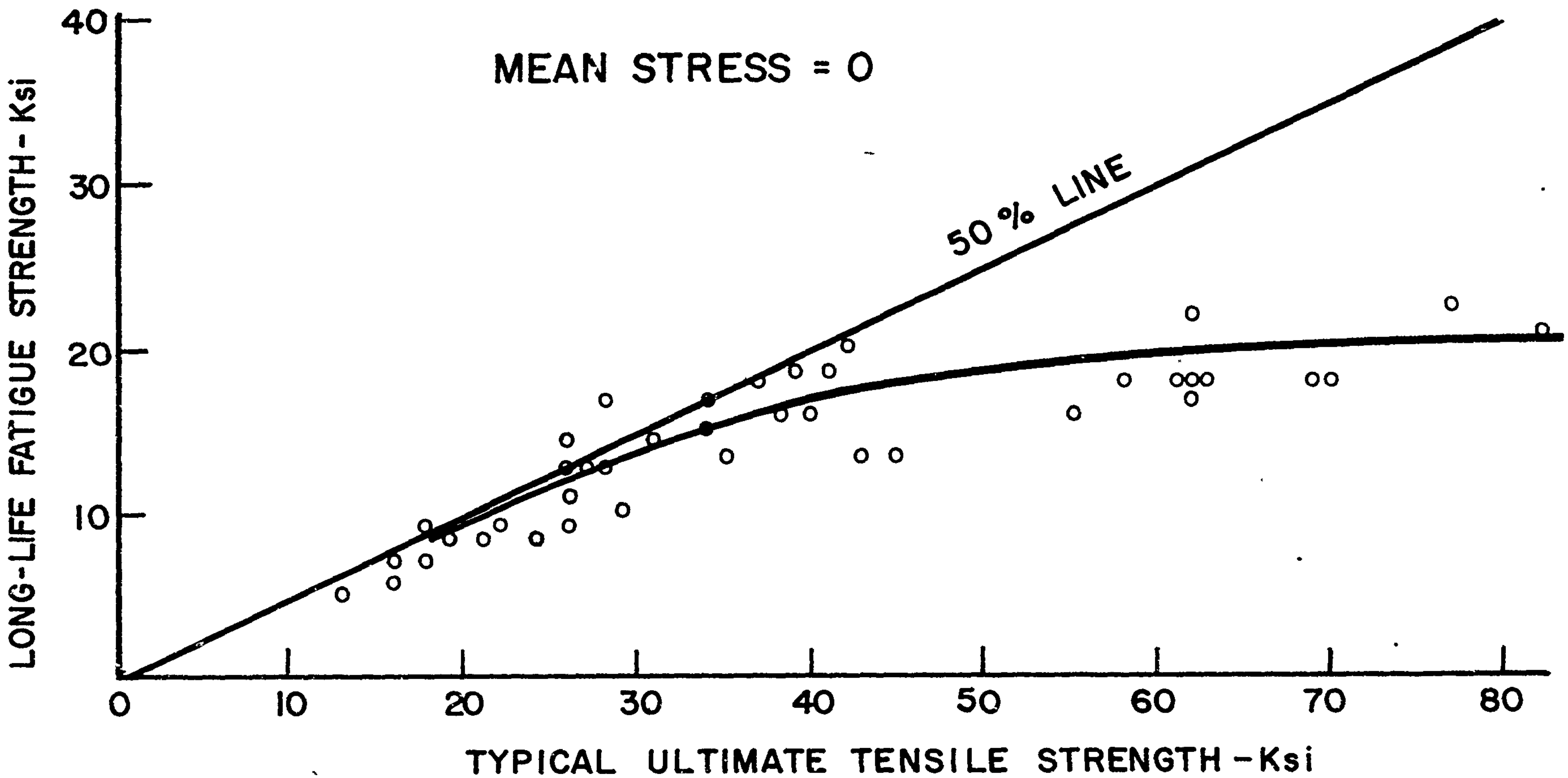


FIG. 1

LONG-LIFE FATIGUE STRENGTHS (500 X 10^7 CYCLES) OF WROUGHT ALUMINUM ALLOYS VERSUS THE ULTIMATE TENSILE STRENGTHS

(REYNOLDS METAL CO. DATA)

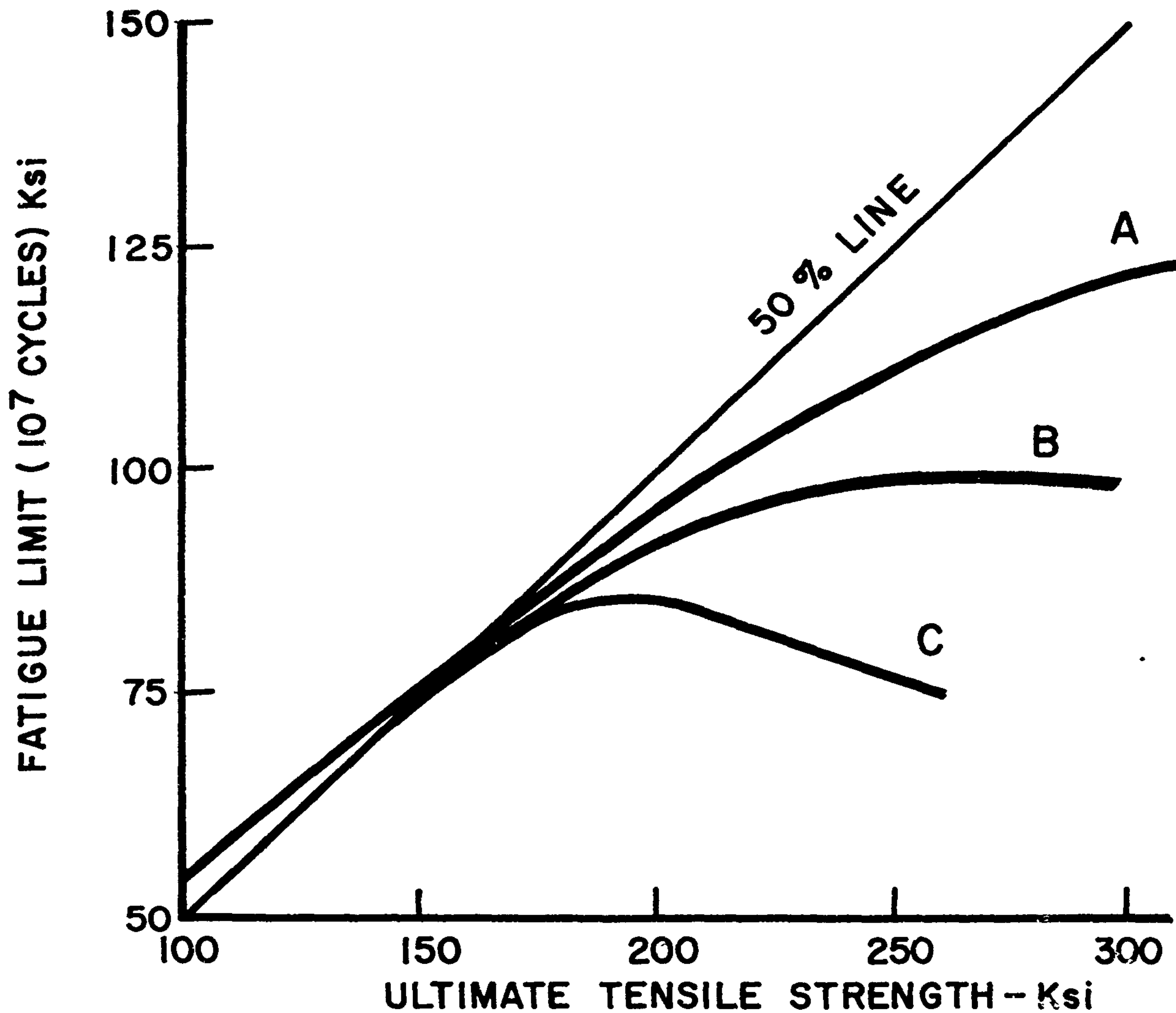


FIG. 2

TYPICAL SHAPE CURVES OF THE FATIGUE LIMITS OF ALLOY STEELS AS FUNCTIONS OF THEIR TENSILE STRENGTHS

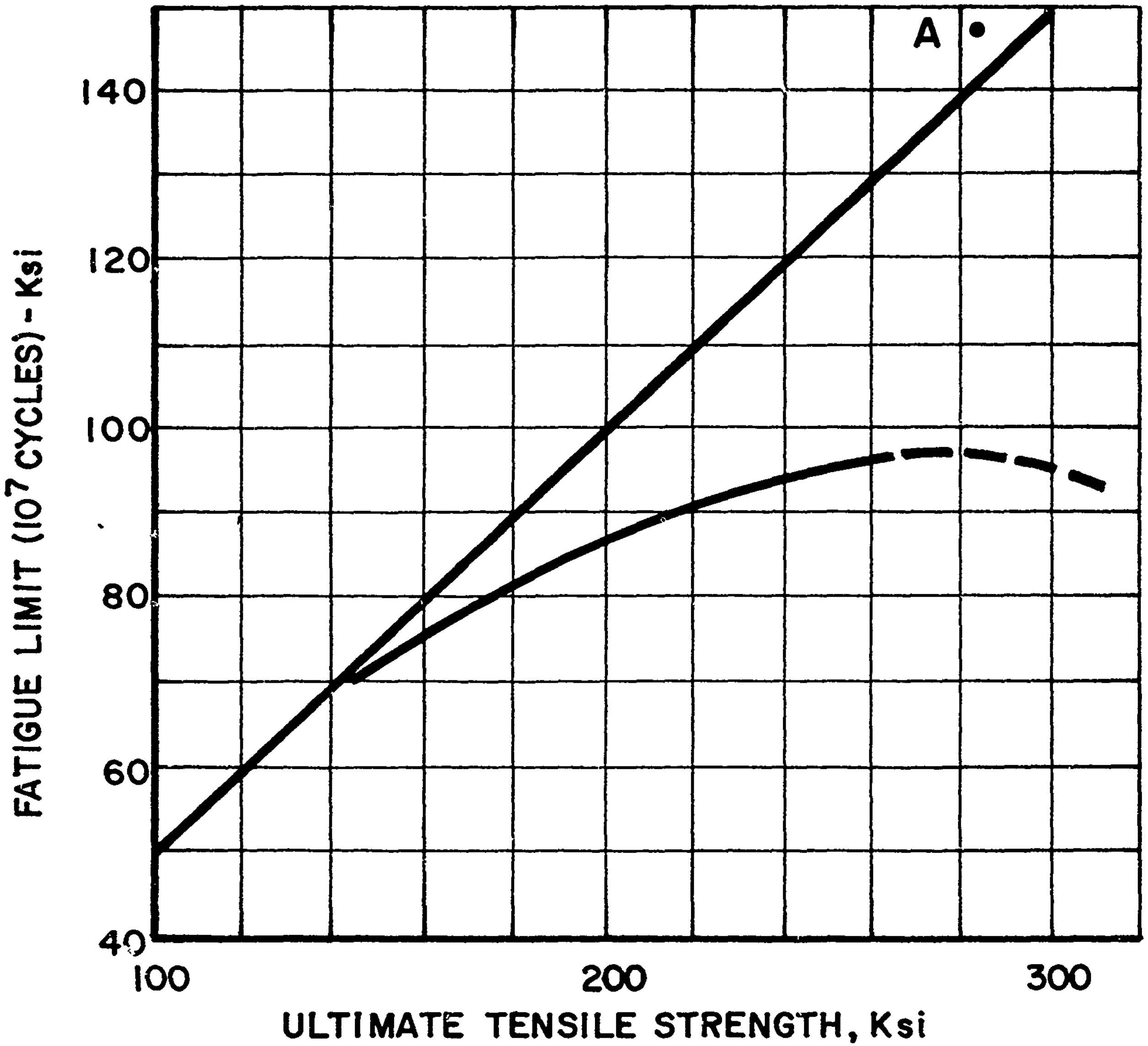


FIG. 3

FATIGUE LIMIT (10^7 CYCLES) VS ULTIMATE TENSILE STRENGTH FOR ONE HEAT OF SAE 4340 STEEL

The above list of conditions although not complete has been presented simply to show that the fatigue strength of a manufactured part is dependent on many external influences, many of which are not directly related to the fatigue properties of the material itself. It is for this reason that a fatigue failure in service is often found to be caused by factors largely unrelated to the material properties.

The large degree of scatter in the fatigue strengths or fatigue lives of supposedly identical parts tested under constant conditions is partly caused by the variability of these external influences. As mentioned previously, when these external sources of variability are completely eliminated so that the plain material is being tested in fatigue alone, a surprising amount of scatter is found to exist in the material itself. For example, in high-quality aircraft steels, the standard deviations of the fatigue limit may range from several percent of the fatigue limit to over ten percent. The largest scatter is associated with the highest hardnesses. On the other hand, the conventional static mechanical properties of these materials may be quite uniform.

This has led past investigators (10) to suspect that there is some source or sources of variation in fatigue strength from one microscopic point to another within the material itself. The variability may be traced to many types of inhomogeneities or discontinuities that exist in engineering polycrystalline metals. Each type of inhomogeneity may be considered as a population having some prescribed distribution and having its own effect on the overall fatigue strength of the metal. If the dominant type of inhomogeneity were eliminated, it would be found that another, less dominant, population of inhomogeneity would then govern the magnitude and variability of the fatigue strength of the metal, and so on ad infinitum. Some of the known and suspected types or sources of inhomogeneities are listed below:-

(1) Metallic and non-metallic inclusions
(2) Grain orientation
(3) Metallurgical Structure
(4) Segregation of alloys
(5) Microstresses
(6) Shape and size of the precipitate
(7) Density of dislocations, vacancies, etc.
(8) Imperfections such as microcracks, blowholes, seams, etc.

It is the variability of the material from one microscopic or submicroscopic point to another that is responsible for the large variability in the fatigue strength or life of the material. When metal is tested at stresses corresponding to long life, the weakest microscopic spot subjected to the highest stresses is the first one to fail. It is the variability of these extreme inhomogeneities that causes the large variability in the long-life fatigue strength. As discussed by Peterson (10) and others, there is an increasing number of lesser inhomogeneities where cracks initiate as the stress is increased. The effect of certain types of inhomogeneities will be discussed in the following paragraphs.

NON-METALLIC INCLUSIONS

For many years, it has been recognized that imperfections in metals such as microcracks, segregation, blowholes and inclusions have a deleterious effect on

their fatigue strengths. This is particularly true of alloy steels heat treated to the higher hardnesses. Within the last few years, several investigators (11-17) have studied the effect of inclusions on the fatigue strength of steels.

Early in 1953, the WADC Materials Laboratory[a/] (14,18,19) sponsored a research program to determine the variability of fatigue strengths of high-strength aircraft-quality steels and to investigate the sources of this variability. Rotating-beam specimens were tested in this program in sufficient numbers for each heat of steel so that trends could be definitely identified and measured. (Often in previous work of this kind insufficient specimens were tested so that trends could not be quantitatively evaluated.)

During this program, several facts concerning the fatigue of high-strength steels were uncovered. The more pertinent facts are summarized below:-

(1) The largest variability in fatigue life or strength occurs near or slightly above the fatigue limit. At higher overstresses, this variability decreases with increasing test stress. (This characteristic had been observed previously by many investigators.)

(2) As the tensile strength is increased above about 140,000 psi by change in the tempering temperature, the variability in the fatigue strength increases. At the highest strength level (280/300,000 psi) this variability increased threefold.

(3) Near and slightly above the so-called fatigue limit, the crack starts at a single nucleus and propagates to complete failure. At stresses above 130% of the fatigue limit, cracks nucleate at increasing numbers of nuclei and join to form the final fracture surface.

(4) It was found early in this investigation that all nuclei of cracks were situated at microscopic spheroidal silicate inclusions that were either open to the surface or slightly below the surface. The silicate type of inclusion is common to the basic electric-furnace steels. In one heat of 4340 steel, the average size of these nucleating inclusions was only 0.0015 inches and the maximum did not exceed 0.0030 inches. However, it was determined that these microscopic inclusions are a dominant factor in fatigue.

(5) By statistical analyses two important relations between the inclusion sizes and fatigue strengths were established as follows:

    (a) The long-life fatigue strength is directly related to the inclusion size; the largest sizes correspond to the lowest fatigue strengths or lives, and the smallest sizes correspond to the highest strengths or lives.

    (b) The reduction in fatigue strength associated with a given inclusion size increases with the hardness level of the steel.

a/ This work was sponsored by the Creep and Fatigue Section of the Materials Laboratory, headed by W. Trapp.

Figure 4 is a typical plot of probability S-N curves obtained by testing smooth rotating-beam specimens of one heat of air-melted electric-furnace 4340 steel. The specimens in this particular heat were heat-treated to an ultimate tensile strength of about 190 ksi. It is apparent from these curves that the greatest variability or scatter in life or strength occurs at or near the fatigue limit and that the variability decreases as the stress is increased. In this program, this steel was heat treated to four hardness levels and tested in fatigue. It was observed that the variability in strength near the fatigue limit increased rapidly as the tensile strength was increased above 140 ksi. Figure 5 is a plot of this variability (in terms of standard deviations) against the tensile strength. Figure 6 is an enlarged view of a typical fractured surface showing the inclusion at the origin of the failure.

In the above test program, the sizes of the nucleating inclusions were measured on hundreds of specimens and correlated with the fatigue life displayed by each specimen (17). The analysis of these data is summarized in figure 7 which presents the "notch reduction factor" of this type of inclusion as a function of its size and the hardness level of the steel. These results are rather surprising in that they are not in agreement with the classical theory of elasticity which indicates that the notch reduction factor of these inclusions should be independent of the inclusion size and the hardness level.

In figure 3, it is seen that inclusions below about 0.0003 inch in diameter have little effect on the fatigue limit and that small inclusions (up to 0.0030 inches) are not important at tensile strengths below about 140,000 psi.

Another fatigue investigation on three heats of 52100 steel heat treated to C63 Rockwell hardness corroborated these findings. In this investigation, the Prot accelerated test method was employed, using a very low rate of loading (0.007 psi per cycle). The Prot failure stress was plotted as a function of the size of the nucleating inclusion in figure 8. Here it is seen that if the maximum inclusion is reduced from 0.0020 inches to less than 0.0003 inches, the long-time fatigue strength increased from about 110,000 psi to over 160,000 psi; a substantial increase.

Several hypotheses have been suggested to explain this marked increase in the sensitivity of steels to inclusions when the hardness level is increased. One possibility is that the ductility or plasticity of the steel decreases, and this lower ductility allows less re-adjustment of the high stress field about the inclusion and less work hardening in fatigue so that a crack will initiate more rapidly prior to establishment of a more resistant work-hardened state in the metal about the inclusion. Another possibility is that high residual tensile stresses are set-up around the inclusion during the quenching operation since the coefficient of thermal expansion of steel is greater than that of silicates. Since the tempering temperature is lower at the higher hardnesses, there is less relaxation of these residual stresses. These residual stresses have been estimated to exceed 100,000 psi.

Encouraged by these foregoing results, the WADC Materials Laboratory sponsored a program at Armour Research Foundation to produce and test various air-melted heats of 4340 steel that were free of the silicate type of inclusion (20). In one experimental heat they found one of the highest fatigue strengths that has been recorded to date. In this heat the silicon was eliminated and the final deoxidation was done with aluminum. The maximum inclusion size was less than

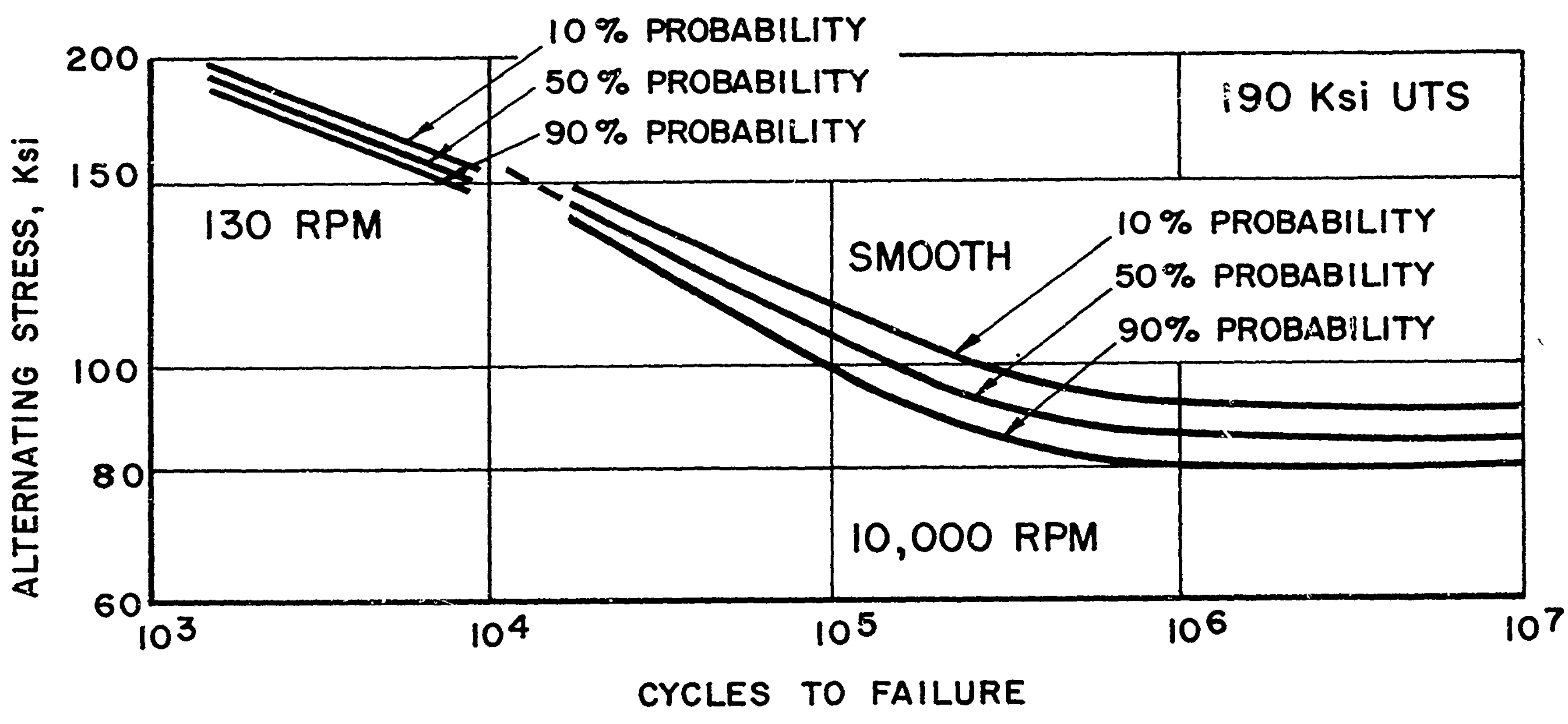


FIG. 4

S-N CURVES OF CONSTANT PROBABILITY OF SURVIVAL OF STRESS AT CONSTANT LIFE

SAE 4340 STEEL R.R. MOORE ROTATING BEAM TESTING

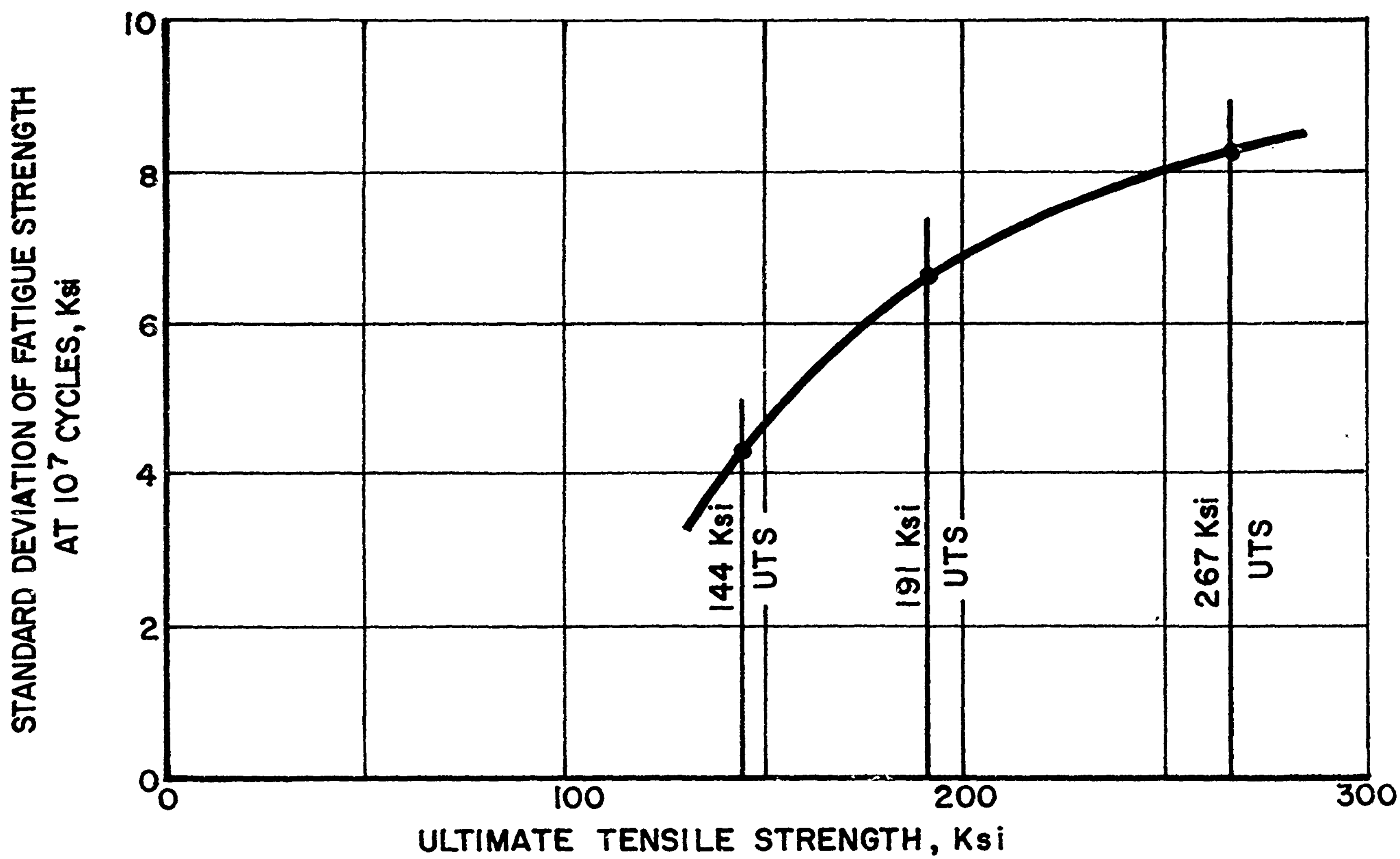


FIG. 5

STANDARD DEVIATION OF FATIGUE LIMIT (10^7 CYCLES)
VS ULTIMATE TENSILE STRENGTH FOR SAE 4340 STEEL

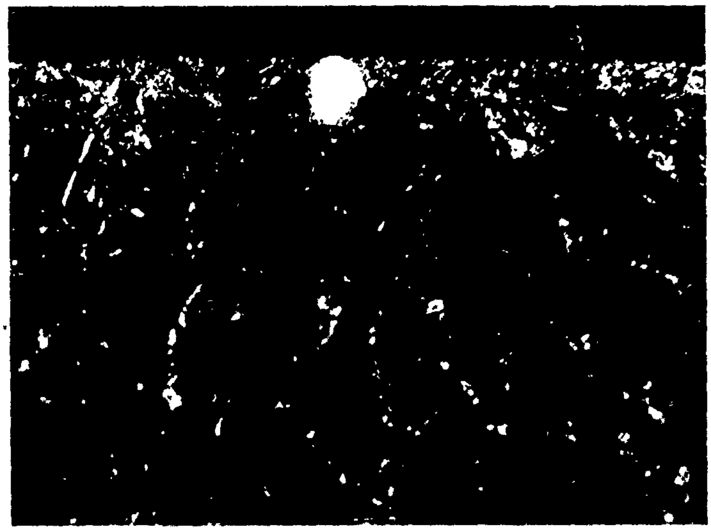


FIG. 6

INCLUSION OBSERVED AT ORIGIN OF FAILURE OF A

SPECIMEN OF SAE 4340 STEEL

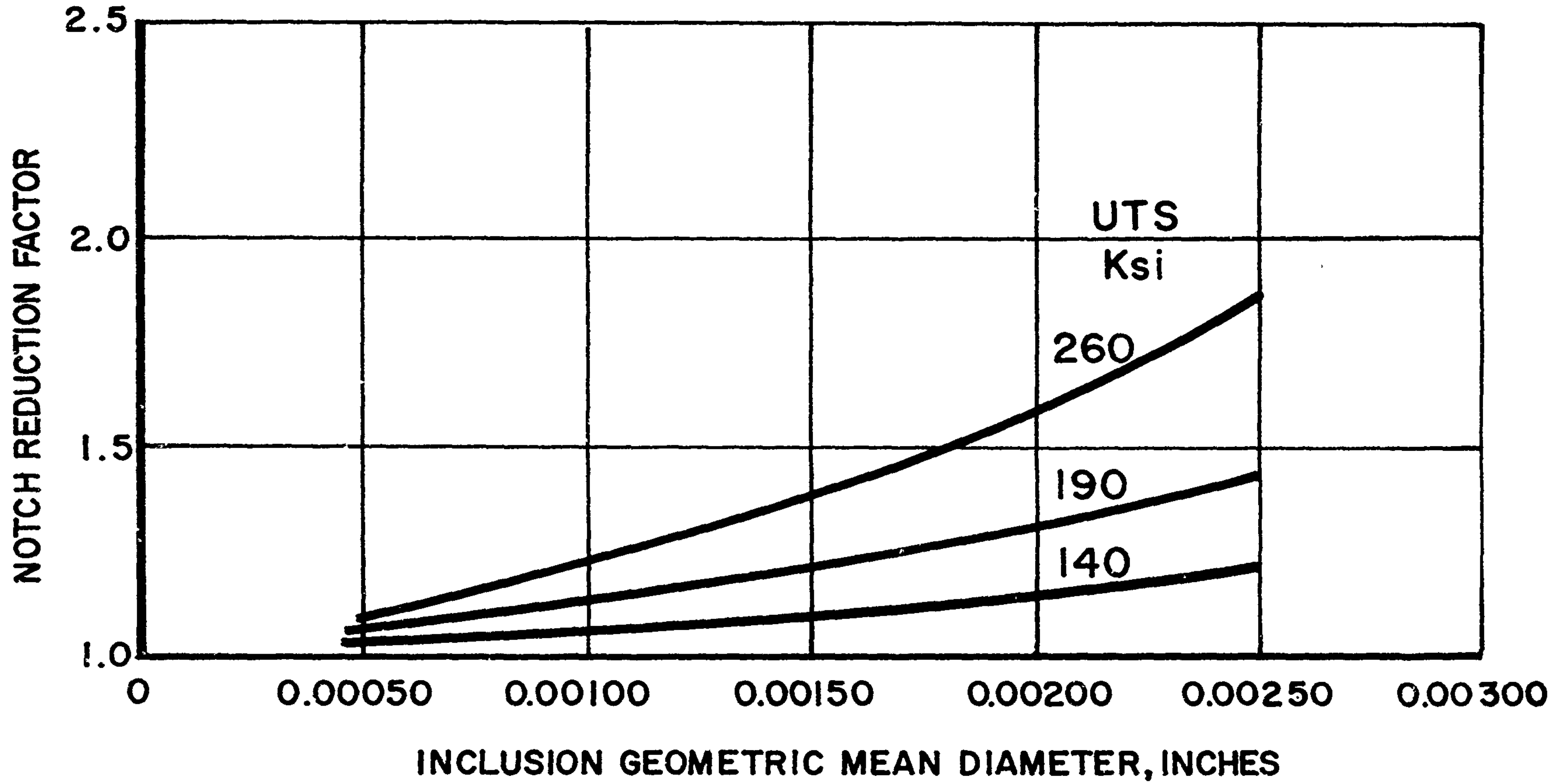


FIG. 7

TENTATIVE NOTCH REDUCTION FACTORS OF NON-METALLIC INCLUSIONS IN SAE 4340 STEEL AT 10^7 CYCLES

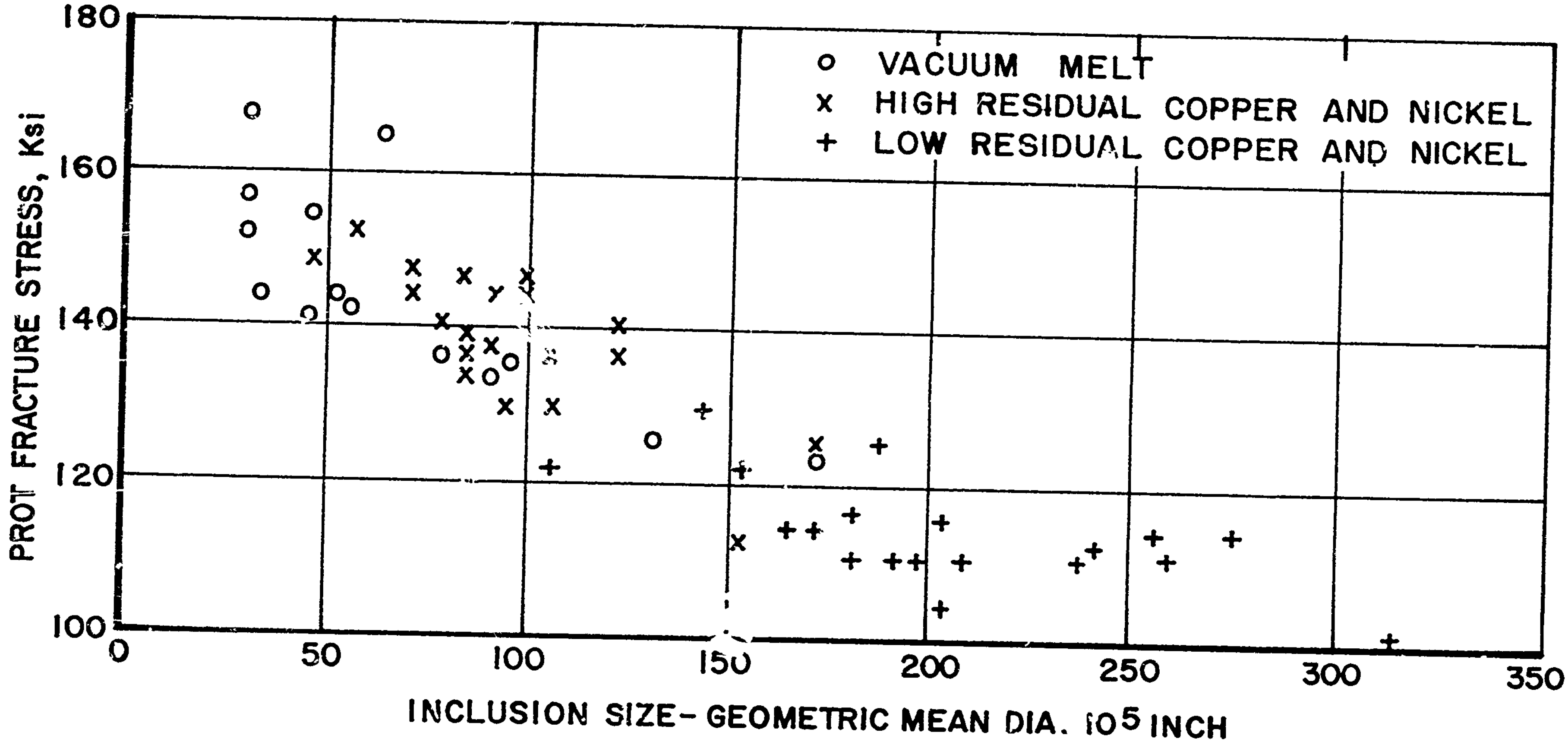


FIG. 8

PROT FRACTURE STRESS VS INCLUSION SIZE FOR 52100 STEEL

0.0007 inches. These strengths are compared with those of a high-quality 4340 steel made in the conventional manner, in the following table:

Material	Method of Manufacture	Ultimate Tensile Strength-ksi	Average Fatigue Strength-ksi
AISI 4340	Special Air-Melt to Reduce Inclusion Size	282	148
AISI 4340	Conventional Electric-Furnace Air-Melt	268[a/]	96

(a/- Had this UTS been equal to 282 ksi, the fatigue strength would be estimated to be about 94 ksi.)

The characteristic curves (A, B & C) of figure 3 showing the typical shapes of fatigue versus tensile strength relations may be explained on the basis of the inclusion notch reduction factors shown in figure 7 and on the basis of one assumption. Since investigations described in previous paragraphs have shown that the fatigue limits of high-strength steels that are practically free of inclusions exceed 50% of the ultimate tensile strength at the highest hardness levels, it is assumed that an "inclusion-free" steel will have a fatigue limit equal to 55% of its ultimate strength over its practical range of hardness. Now referring to figure 7, it is seen that if the maximum size of the inclusions does not exceed 0.001 inch, the fatigue limit at 260,000 psi tensile strength would be 0.55 x 260,000/1.20 = 119,000 psi. If, however, the maximum inclusion size were 0.0025 inch, the fatigue limit would be 0.55 x 260,000/1.65 = 87,000 psi. By this method of analysis, it is possible to approximate the shapes of the curves shown in figure 2.

It is suggested that the standard plot of fatigue strength versus tensile strength of a given material might serve as a "homogeneity index" (provided of course the ultimate tensile strength can be changed by heat-treating procedures). At least, in steels this plot might be used to determine whether the steel is homogeneous or whether high microstresses exist after heat-treatment.

On comparing figure 1 for aluminum alloys with figure 2 for steels, it is seen that there is a definite similarity in the shapes of the curves of these two metals. For this reason, as well as because of the similar degree of scatter in fatigue life of high-strength aluminum alloys, it is suspected that there are inhomogeneities or high micro-stresses in the high-strength aluminum alloys that cause this flatness in the curve of figure 1. The authors of reference (21) suggest that the fatigue nuclei in aluminum alloys are not actually foreign particles but may be a constituent formed from elements that have low solubility in aluminum. In the case of 2024 and 7075 alloys, they suggest that the hard particles of an Fe-Al-Si ternary constituent which are present act as nuclei in fatigue. The precipitates themselves may be the source of the fatigue nuclei. Certainly additional investigations are required to answer these questions.

EFFECT OF MICROSTRUCTURE

Although the effect of the type of microstructure on fatigue has been reported from time to time in the past by a few investigators, the results have

been confused by changes in the tensile strength when the type of microstructure is changed. For example, a change in the microstructure (ferrite, pearlite, martensite, etc.) in a steel of a given composition is usually accompanied by a significant change in the tensile strength. Cazaud (22) in his book on fatigue has observed general trends of the fatigue strength ratios for various microstructures in steels. In order to separate these two effects (microstructure versus tensile strength), it is necessary to change the microstructure without changing the tensile strength.

One investigation of this point was undertaken by Dieter and Mehl (21) who examined the effect of two different microstructures on the fatigue strength of the same steel. By using different heat treatment procedures they produced a coarse spheroidized structure and a coarse pearlitic structure from the same heat of plain carbon eutectoid steel. Both structures had the same tensile strength but different ductilities and fatigue limits. The globular spheroidized structure had the higher fatigue limit and ductility. This can be explained on the basis that the plate-like pearlitic structure presents a greater stress concentration than the spheroidized structure. Although the spheroidized structure had the higher fatigue limit, it also showed about twice the scatter (in terms of standard deviation).

There are other types of inhomogeneities in the microstructure of metals that are suspected of causing a loss in the fatigue strengths and of creating the observed variability. Segregated carbides are known to be deleterious to the fatigue strength of high carbon steels. Segregation of other alloying elements is another source of loss and variability. The anisotropy and orientation of the crystalline structures as well as the various shapes and sizes of the grains and subgrains are suspected to be additional sources.

As implied above, there are two aspects of inhomogeneities in metals; (1) the notch reduction factors of each type and (2) the variability in this factor for a given type. To reduce the loss in fatigue strength, either the inhomogeneities must be eliminated or their sizes must be reduced. Since it is often impractical to eliminate the inhomogeneity, a more practical approach is to reduce the size of the largest inhomogeneities.

A survey of the literature indicates a paucity of investigations directed towards obtaining solutions to this major problem. This is particularly so when it is realized that statistical techniques must be used to plan, conduct and analyze experiments in this field.

EFFECT OF MICROSTRESSES

Most of the high-strength engineering alloys obtain their optimum strengths by heat-treatment. Usually the strength properties are obtained by the precipitation of hard constituents (cementite in steel, copper in dural, etc.) from the high temperature solid solution. During the cooling of the metal in the heat treatment and during the precipitation of the hard particles, macrostresses and tessellated microstresses are induced. The macrostress distribution (which is usually beneficial) is caused by the difference in the cooling rates at the surface and at the interior of the metal. This macrostress distribution can be measured by machining metal from the part and observing the corresponding changes in the dimensions.

However, microstresses are induced by the fact that the metal crystal has different coefficients of thermal expansion in the different metallographic directions. Also the precipitated particles create a dimensional misregistry with the adjacent matrix metal on an atomic scale. These stresses vary from tension to compression from one microscopic point to the next in a metal. For this reason, they cannot be measured directly. It is believed that the microstresses around the non-metallic inclusions and submicrostresses around the precipitates have dominant deleterious effects on the fatigue strength and on the ductility of alloys.

If this hypothesis is correct, one would expect that the fatigue strength of steel could be substantially improved by changing its chemistry so as to require a high tempering temperature to produce a given hardness. The high tempering temperature would permit a greater stress-relief of the microstresses induced during the quenching operation.

During recent years several companies have developed new high-strength steels that have superior properties at tensile strengths of the order of 280-300 ksi. In every case the tempering temperature has been increased considerably above that used in former grades of steel for this tensile strength. The highest fatigue strengths have been obtained in those steels requiring the highest tempering temperature.

By multiple correlation studies on various high strength steels, it has been determined that, of the common elements used in steel manufacture, molybdenum has, by far, the greatest effect in raising the tempering temperature. Several steels of high moly content were designed and tested at our laboratory. Unfortunately the first heats of these steels were very "dirty" (i.e., they had large non-metallic inclusions). However, they did show a much greater fatigue strength than a conventional steel heat treated to the same hardness. These results are shown below:-

Steel	Tempering Temperature-°F	UTS-ksi	Average Fatigue Strength (10^7)
4350	400	314	94
Special High Moly Steel	1100	300	146

Higher fatigue strengths in steels are to be expected if the tempering temperatures can be further increased.

RELATION OF FATIGUE TO DUCTILITY

Since the fatigue strength of a metal may depend on a number of independent factors and since it is difficult in experiments to change one variable at a time, it is not possible to obtain a precise correlation between it and any other single bulk property of the metal.

The static ductility of a metal found in a conventional tensile test is a bulk property of the metal and depends in part on the strength level, microstructure, cleanliness and microstresses in the metal. However, these factors affect the ductility in a degree and manner that are different from their effects on

fatigue. The long-life fatigue strength is dependent on the one single microscopic spot that is the weakest in the region of the highest stresses. A relatively small change in the maximum sizes of the inclusions in steels that are otherwise identical would cause a substantial difference in their fatigue strengths but would probably show relatively small changes in the static properties including ductility. However, it would appear reasonable to suspect that, if the type of microstructure, grain size, inclusions, etc., in a metal were held constant, then ductility would correlate with the long-life fatigue strength.

Some authors (5, 22, 23) have noted that there appears to be a general correlation between the long-life fatigue strength of a metal with its ductility for a specified tensile strength. Cazaud (22) says ". . . steels which give a high percentage reduction of area at fracture in a tensile test also give a high endurance ratio". The author of reference (22) was discussing ratio of fatigue strength at long life to ultimate tensile strength. Evans et al. (23) have the same opinion. They say, discussing tests on wrought steels, - ". . . the fatigue properties must be somewhat dependent upon the ductility because, although the (static) strength properties are essentially the same (longitudinal and transverse), a pronounced difference is evident in the reduction of area values. In fact, the fatigue properties are rated in the same order as the ductility". Ransom and Mehl (5) speculated that the reduction of area in tension tests of transverse specimens <u>might</u> be used as a quality index for transverse fatigue properties.

Ductility may be defined in several ways for example such as the percent elongation at failure, the reduction of area, or the strain-hardening exponent. The strain hardening exponent is a measure of ductility since it is equal to the true plastic strain at the stress corresponding to tensile instability or the maximum load. Another index of ductility is one minus the yield to tensile-strength ratio since this ratio is related to the strain-hardening exponent.

This suggested general trend might be more specifically stated as follows:

> As a general rule, the fatigue strength of a given alloy <u>at a specified ultimate tensile strength</u> tends to increase with corresponding increases in its ductility (as measured by the reduction of area) obtained by changes in the melting, the reduction, or the heat treatment procedures.

Since fatigue is dependent on many factors other than ductility, the above statement must be considered to be a probabilistic or statistical statement. In certain tests wherein the ductility is not changed appreciably, other factors might be more dominant thereby seemingly invalidating the above statement.

Some data that support this proposed rule have been tabulated in table 1. A review of these data indicates that the reduction of area correlates better with the fatigue limit than the percent elongation. Evans and his co-workers (23) found that wrought steels always displayed higher fatigue strengths and higher reductions of areas when compared to the same steels in the cast form. Only the average values of these cast and wrought steels have been tabulated in table 1 to show this effect. Cazaud (22) in his book has compiled fatigue data on steels having different microstructures. In general, his data show that, if steels are heat treated to a given tensile strength and have the same microstructure, there is a correlation between the reduction of area and the fatigue limit.

regardless of the chemical composition.

The long-life fatigue strengths (500×10^6 cycles) of annealed and work-hardened aluminum alloys have been plotted in figure 9 as a function of the elongation. Each of these curves correspond to a small range of tensile strengths. Since a given alloy can be work-hardened to various tensile strengths, each alloy has been plotted as several points depending on its hardness. If the tensile strength of an aluminum alloy is increased by cold-working, its fatigue strength will increase only slightly and its ductility will decrease. No relation between ductility and fatigue strength was found for the heat-treated aluminum alloys, probably because the range of hardnesses of such alloys is too small for a trend to be established.

NOTCH SENSITIVITY

Although a large volume of data on the notch fatigue strength and notch sensitivity of materials has been published, much confusion still exists on the interpretation of these terms and their relation to the fatigue quality of a material. For example, it would seem natural to judge a material to be excellent if its notch sensitivity is very low, yet many materials that have low notch sensitivities also have low fatigue strengths. Cast iron has a low notch sensitivity, but also has a relatively low fatigue strength.

It is suspected that, in many notched fatigue tests reported in the past, there were high residual stresses and work-hardening at the bases of the notches, introduced by the machining processes. In these cases, the fatigue strength of notched specimens is confused with these additional, extraneous, variables. Only when the notches are carefully machined and stress-relieved is the true notched fatigue strength obtained. Materials that have a high strain-hardening index can be inadvertently work-hardened considerably at the base of a sharp notch during machining, which also induces high residual compressive stresses. An example of this is austenitic stainless steel where large increases in their notched strengths have been found by work-hardening the roots of the notches. Under this condition, the notch sensitivity would appear to be low. However, when these notched specimens are properly stress-relieved, much lower notched fatigue strengths are observed.

Therefore, the proper definition of notch sensitivity should involve the notch fatigue strength of the material wherein the notch root is known to be free of work-hardening and residual stresses.

Another fact that confuses the meaning of notch fatigue strength and notch sensitivity is that British investigators (24, 25, 26) have discovered non-propagating cracks to form and stop their growth in notched specimens at stresses below the nominal notched fatigue limits. If crack formation is the criterion of failure, then in these cases the fatigue limit is considerably below that corresponding to complete failure.

The fatigue notch sensitivity of a metal may be interpreted in a different way. This is shown by the results on two heats of 4340 steel (18); one was an air-melted aircraft-quality heat and the other a vacuum-melt heat. Both sets of specimens were heat-treated to about 190 ksi tensile strength and carefully stress-relieved in an inert atmosphere after the final machining operation. Rotating-beam fatigue tests were conducted and analyzed using statistical

Table 1

Relation of Ductility and Fatigue Strengths

Material	UTS ksi	YP ksi	Elong. % in 2"	R.A. %	(10^7 Cycles) Average Fatigue Strength - ksi	Remarks	Ref.
Spheroidized Steel	92.7	71.1	28.9	57.7	41.5		(21)
Pearlitic Steel	98.2	35.5	17.8	25.8	34.0		(21)
4340- Air-melt	191.7	184.0	15.3	50.1	84.5		(18)
4340- Vacuum-melt	195.2	185.0	16.2	57.9	100		(18)
4340 Consumable Electrode	297.6	224.0	11.4	36.4	125	By Prot Method	(18)
Tricent Consumable Electrode	304.6	251.2	10.0	28.5	117	By Prot Method	(18)
Super Hy-Tuf Consumable Electrode	313.2	267.3	8.2	26.2	119	By Prot Method	(18)
Tricent - Electric Furnace	294.0	229.0	11.0	37.2	116	By Prot Method	(18)
4350 - Electric Furnace	314	310	8.3	6.1	94	Conventional Test (10^7)	(18)
Ave. of 5 Cast Steels (Normalized & Tempered)	108.6	78.0	20.0	51.5	50.8	From Evans et al.	(23)
Ave. of These-Wrought (Normalized & Tempered)	107.2	76.0	24.2	61.1	60.8	From Evans et al.	(23)
Ave. of 4 Cast Steels (Quenched & Tempered)	143.6	129.7	14.7	35.9	65.6	From Evans et al.	(23)
Ave. of These-Wrought (Quenched & Tempered)	143.7	130.6	19.9	60.7	82.5	From Evans et al.	(23)

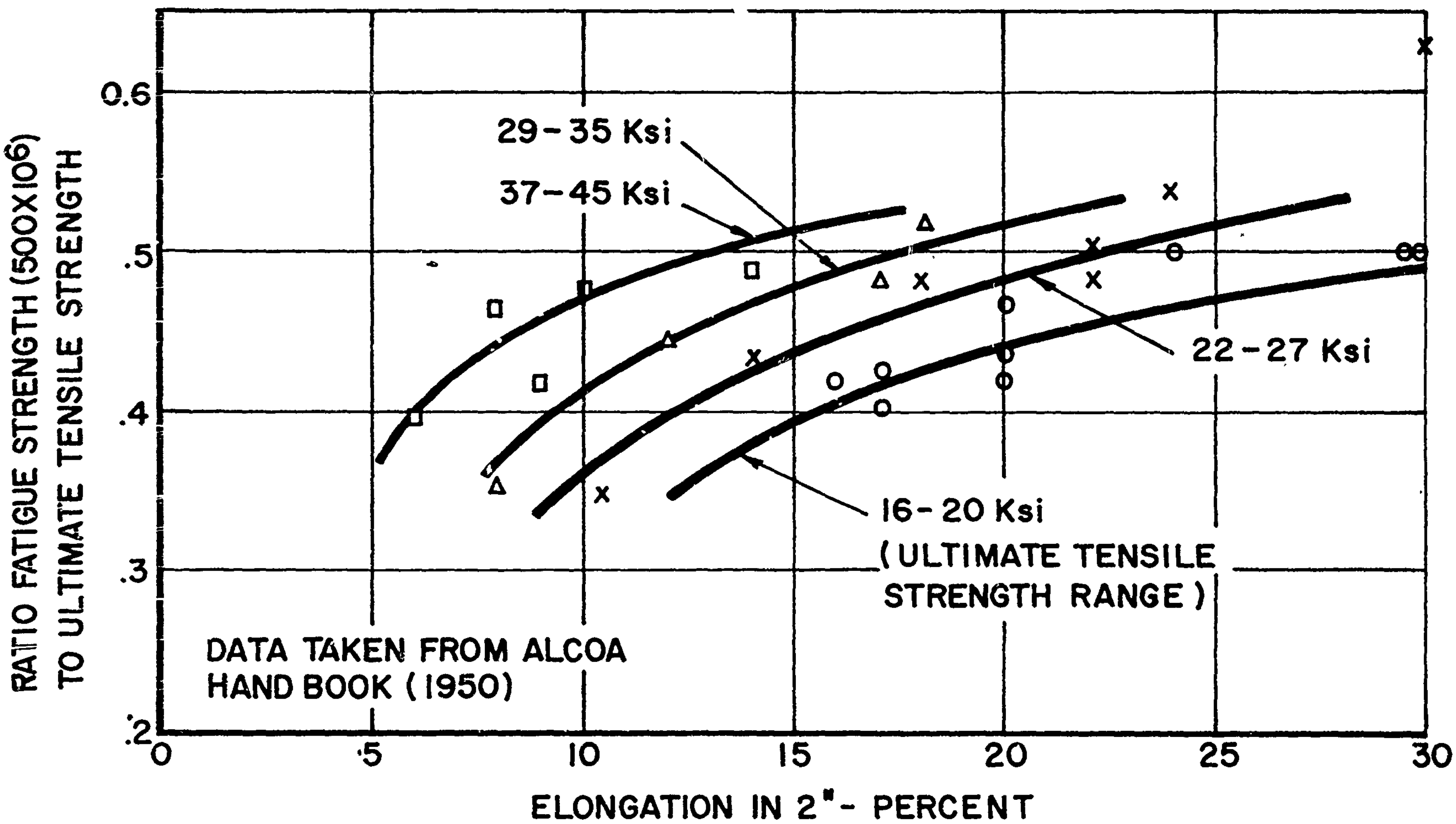


FIG. 9

FATIGUE STENGTH TO TENSILE STRENGTH RATIO AS A FUNCTION OF ELONGATION FOR SOFT AND HARD WORKED ALUMINUM ALLOYS

procedures. The results are shown in the table below:

Material	Average Smooth Fatigue Strength (10^7)	Average Notched[a/] Fatigue Strength (10^7)
Air-Melted 4340	84.5	41
Vacuum-Melted 4340	100	40

a/ The geometric stress concentration factor of the notch was 2.6.

From these values it is seen that although the smooth value of vacuum-melted heat is substantially higher, there is no significant difference in the notched strengths.

Here, then, is a case where a superior material showed no improvement in its notched condition and where its notch sensitivity increased. This suggests the following explanation. In either heat of steel the amount of material subjected to high stress at the base of a sharp notch is very small compared to that of a smooth specimen. Therefore, we are testing relatively clean material at the base of the notch in either case and the difference in performance appears only when sufficient volume is tested.

From this one would expect that notch sensitivity is a probabilistic phenomenon related to the density and influence of the inhomogeneities. As suggested by Schuette (27), we should assume the notch fatigue strength to be the datum from which to judge the smooth fatigue strength. Perhaps we should speak of the "un-notched sensitivity" of a material rather than its notch sensitivity. This philosophy leads to an interpretation of notch sensitivity that is opposite to the conventional interpretation.

Further, this explanation would imply that notch sensitivity is dependent on the size of the notch. A large notch that is geometrically similar to a small notch is expected to have a lower fatigue strength.

Because of the continued confusion in the interpretation of notch sensitivity, it is suggested that future work should be directed towards a better understanding of this term.

THE λ -PARAMETER

A new, simple, relationship that predicts the behavior of metal under biaxial fatigue stresses was independently discovered at about the same time in three different laboratories, two in this country (28 and 29) and one in Japan (30). In this formula, there appears a new parameter which is basic in fatigue phenomena. A brief discussion of this new relationship follows.

If metal is subjected to a set of biaxial or triaxial alternating stresses where the maximum principal alternating stress is designated by p_1 and the algebraically-least alternating principal stress is p_3, then the metal behaves as though it were subjected to an effective uniaxial stress, s, equal to:-

$$s = p_1 - \lambda p_3 \qquad (1)$$

where λ is a material constant. If there are biaxial in-phase stresses, then $p_3 = 0$ (it is the minimum of the three principal stresses) and the effective stress is simply equal to the larger of the two stresses. If the biaxial stresses (see figure 10) are out-of-phase, that is, one is positive when the other is negative then the effective stress is equal to the maximum plus a fractional part (λ) of the other (the plus is used since there are two negatives). This relation is valid at the "endurance-limits" of the material.

Findley and co-workers in a series of reports (29, 31, 32) compared this relation with many others and found that it best fitted the experimental data.

The material constant, λ , may be determined by noting that an alternating pure shear stress is equivalent to two out-of-phase principal stresses equal to the shear stress (designated by τ_e). The effective uniaxial stress for a pure shear test is obtained by substituting τ_e and $-\tau_e$ in the above relation and noting that this must be equal to the uniaxial fatigue limit. On this basis, then:-

$$S_e = \tau_e + \lambda \tau_e$$

which is solved for λ , or:-

$$\lambda = \frac{S_e}{\tau_e} - 1 \qquad (2)$$

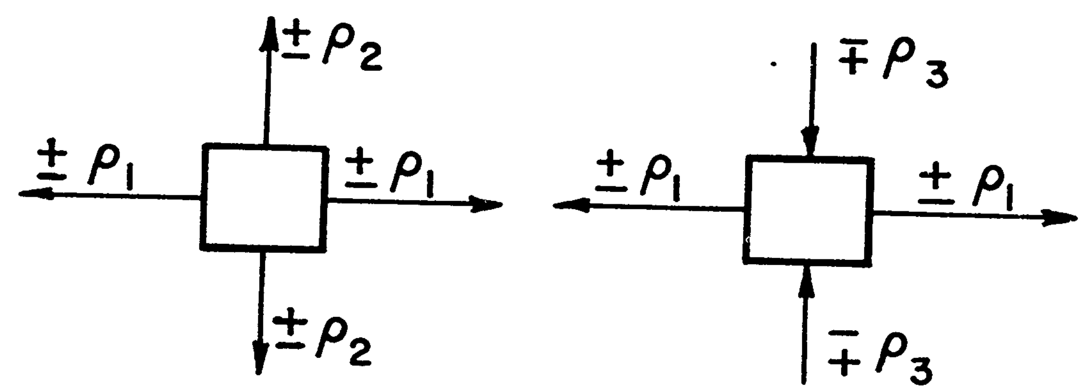
That is, this parameter may be determined by a uniaxial fatigue and a shear fatigue test since it is equal to the quotient of the fatigue limit under uniaxial stress and the fatigue limit in shear, minus one. Theoretical considerations and experimental tests demonstrate that its value lies between zero and unity.

Experimental data show that the value of this constant is near zero for low-quality brittle materials, and that values in the vicinity of 0.6 to 0.7 are common to high-quality wrought engineering alloys. This trend suggests that λ might be used as a "fatigue quality index". A chart suggesting the relation between λ and the fatigue quality is presented below:

Fatigue Quality	λ - Range
Very Poor	0 to 0.2
Poor	0.3 to 0.5
Good	0.6 to 0.7
Excellent	Greater than 0.7

Most wrought engineering alloys are anisotropic in regard to their fatigue strengths. That is, the fatigue strength depends on the orientation of the applied stress. In rolled plates and sheets, for example, there are three major axes of anisotropy; one in the direction of rolling and another two that are perpendicular to this direction. The fatigue strengths in these two latter directions are usually less than in the direction of rolling. A limited amount of data indicates that the fatigue strength in the "short perpendicular" direction may, in some cases, be very low. It is for the above reason that the value of λ ,

$$S = \rho_1 - \lambda \rho_3$$



$\rho_3 = 0$ $\rho_2 = 0$

IN - PHASE BIAXIAL PRINCIPAL STRESSES

OUT-OF-PHASE BIAXIAL PRINCIPAL STRESSES

FIG. 10

BIAXIAL STRESS SYSTEMS

must be known for each principal direction of anisotropy. A material may rate high along the major axis but be inferior in the other directions. A discussion of the sources and causes of anisotropy in wrought steels is presented in reference (13).

There are several fatigue characteristics that correlate with this new parameter. For example, the slope of the Goodman diagram is inversely related to the value of λ. The notch-sensitivity of a metal is directly proportional to this parameter.

THE FATIGUE PROCESS AND CRACK PROPAGATION

The conventional S-N curve obtained by testing supposedly similar parts or laboratory specimens in fatigue is usually associated with the complete fracture of the parts or specimens. In some cases, the failure is defined as that point in the fatigue process when the crack or cracks have become visible or have grown to some measurable size. The conventional S-N curve usually is the representation of two or more distinct stages of fatigue; one is the initiation or nucleation stage and another is the crack propagation stage.

It has been found that at moderate stresses the nucleation stage prior to when the cracks are first formed occurs at a relatively low cycle-ratio or a small percentage of the fatigue life; the actual amount being dependent on the magnification employed in detecting the first crack (19, 33-39). There is evidence by electron microscope techniques that micro-cracks have formed at relatively few cycles (35). When smooth parts or specimens are tested at stress levels near the long-life part of the S-N curve, there are relatively few cracks that initiate and there is considerable difficulty in detecting when and where they are first formed.

The nucleation stage as well as the stage when the crack is very small is characterized by a relatively high degree of variability as measured in cycles. Recent work (19, 36, 37) has established that the variability in the individual nuclei or origins of fatigue cracks accounts for the largest part of the variability in the total number of cycles to failure for stresses at, near, or moderately above the long-life fatigue strength. After a crack has attained a certain small size, its frontal length encloses many grains so that its propagation then becomes more uniform. For this reason the variability in the number of cycles in the final propagation stage among similar specimens or parts tested at the same stress is small compared to the variability of the initial stages.

There is also another, intermediate, stage in the fatigue process of smooth specimens that occurs after a micro-crack first forms. This stage might be appropriately called the "hesitation" stage since in the beginning of the crack propagation, the initial crack often hesitates for relatively large numbers of cycles before restarting its growth. Hunter and Fricke discuss this peculiarity in reference (37). Apparently the crack front runs into "road-blocks" in the form of harder or stronger grains or subgrains necessitating a longer time for the crack to traverse it or to circumvent it by changing its path to adjacent weaker material. Hempel (35) discussing the start of the crack from slip bands states "The direction of the propagation will be influenced above all by defects of the most diversified types, as well as by the orientation of the crystallites as to the direction of loading and, in addition, by the slip traces". Non-propagating cracks have been found at the roots of notched specimens stressed

below their endurance limit (34). Perhaps, these cracks, having less degree of freedom of motion compared to those of a smooth specimen, run into road-blocks that they cannot readily circumvent; hence they stop growing. In reference (19), some non-propagating cracks were observed in smooth specimens that were tested at a stress equal to the average fatigue limit. These grew in lengths not greater than about 0.003 in 10^6 cycles, and failed to grow larger after 10^7 cycles.

Table 2

Non-Propagating Cracks (reference 19) in SAE 4340 Steel, 190 ksi UTS, Stressed at ±85 ksi, at 10,000 RPM

Kilocycles	Crack Length, Inches			
0	0.0004	0.0000	0.0000	0.0012
25	.0013	.0011	.0010	.0020
50	.0016	.0015	.0013	"
75	"	"	"	"
100	"	"	.0015	"
150	"	.0016	"	.0021
200	"	.0020	.0016	.0023
250	"	"	"	.0024
300	"	"	"	"
350	"	"	"	"
400	"	"	"	.0026
500	"	"	"	"
600	"	"	"	.0031
700	"	"	"	"
800	.0018	"	"	"
900	.0019	"	.0019	"
1000	.0020	"	.0021	"
2000	"	"	"	"
10,188	"	"	"	.0032 (10,000 kc)
10,208 [a]	.0037[a]	.0110[a]	.0055[a]	Failed[a] (10,062 kc)
10,255 [a]		Failed[a]		

a. Stress raised to ±110 ksi.

It is suspected that micro-cracks occasionally form at stresses below the fatigue limits of polycrystalline metals but do not propagate because of effective crack-stoppers in their paths.

The crack size that is the upper limit of the hesitation stage has been found to be relatively small. In one aircraft-quality steel (19), the hesitation stage observed in small rotating-beam specimens occurred at crack lengths that were less than 0.005 to 0.010 inches. Above this value, the cracks on the average propagated more-or-less continuously until complete fracture. In two aluminum alloys, a significant hesitation period was observed when the cracks were 0.012 inches long (37).

Although the hesitation stage is usually limited to a small size, it does, however, account for a large part of the number of cycles for complete fracture. It is perhaps for this reason that various authors have disagreed upon the cycle-

ratio corresponding to crack propagation. If an S-N curve is to be defined as that corresponding to the "first visible crack" the crack length should be specified and accurately measured. It is not enough to state that the failure criterion was simply "the first visible crack".

In the study of the growth of fatigue cracks in several hundred rotating beam specimens of 4340 (19), the authors found that the end of the hesitation stage (arbitrarily defined as the number of cycles when the exposed crack was 0.010 in. long) corresponded to a cycle-ratio of from 50 to 70%, and that the crack had started at less than 10% cycle-ratio in all cases. This means that the hesitation stage accounted for about 50% of the cycles for failure and the final crack propagation stage accounted for about 40%. At least 90% of the number of cycles for failure is in the crack growth.

Some observers (19, 40) have found at least in bending specimens that the logarithm of the crack length plots as the best straight line against the corresponding number of cycles (when compared to other common functions). Based on this relation, and the fact that the rate of crack propagation increases rapidly with the stress level, a proposed empirical crack propagation formula for a constant stress may be derived as follows:-

$$\log \frac{\ell}{\ell_0} = k\, s^{\alpha} \quad (N - N_0)$$

where:-

ℓ_0 = specified crack length at end of the hesitation stage

N_0 = number of cycles corresponding to ℓ_0

α, k = constants

s = test stress in ksi

Typical crack growth curves have been plotted in figure 11. These cracks have been measured on rotating-beam specimens tested at one stress level. These curves illustrate the previously discussed points; (1) the greater uniformity of crack growth in the last stage, (2) the hesitation periods and (3) the large variability associated with the nucleation and hesitation stages. Figure 12 illustrates the consistency or low variability in the third stage. Here the origin has been located at the start of this stage.

Recently, Ryder (41) studied the crack propagation in an aluminum wing spar subjected to cycles consisting of a steady load followed by 18 cycles of alternating stress. The length of one crack is plotted as a function of the number of program cycles in figure 13. The propagation of this crack (until it becomes large) also fits the proposed formula.

Ryder noted several important points. First, each striation in the fractured surface corresponded to one cycle of stress. Secondly, the width of these striations increased with increasing stress. In fatigue tests on many parts and types of specimens, the tests are usually conducted at relatively low stress so that many thousands of cycles are imposed thus making it difficult to verify

* Another point is that the three left-hand curves in figure 11 are associated with larger inclusions than the three right-hand curves.

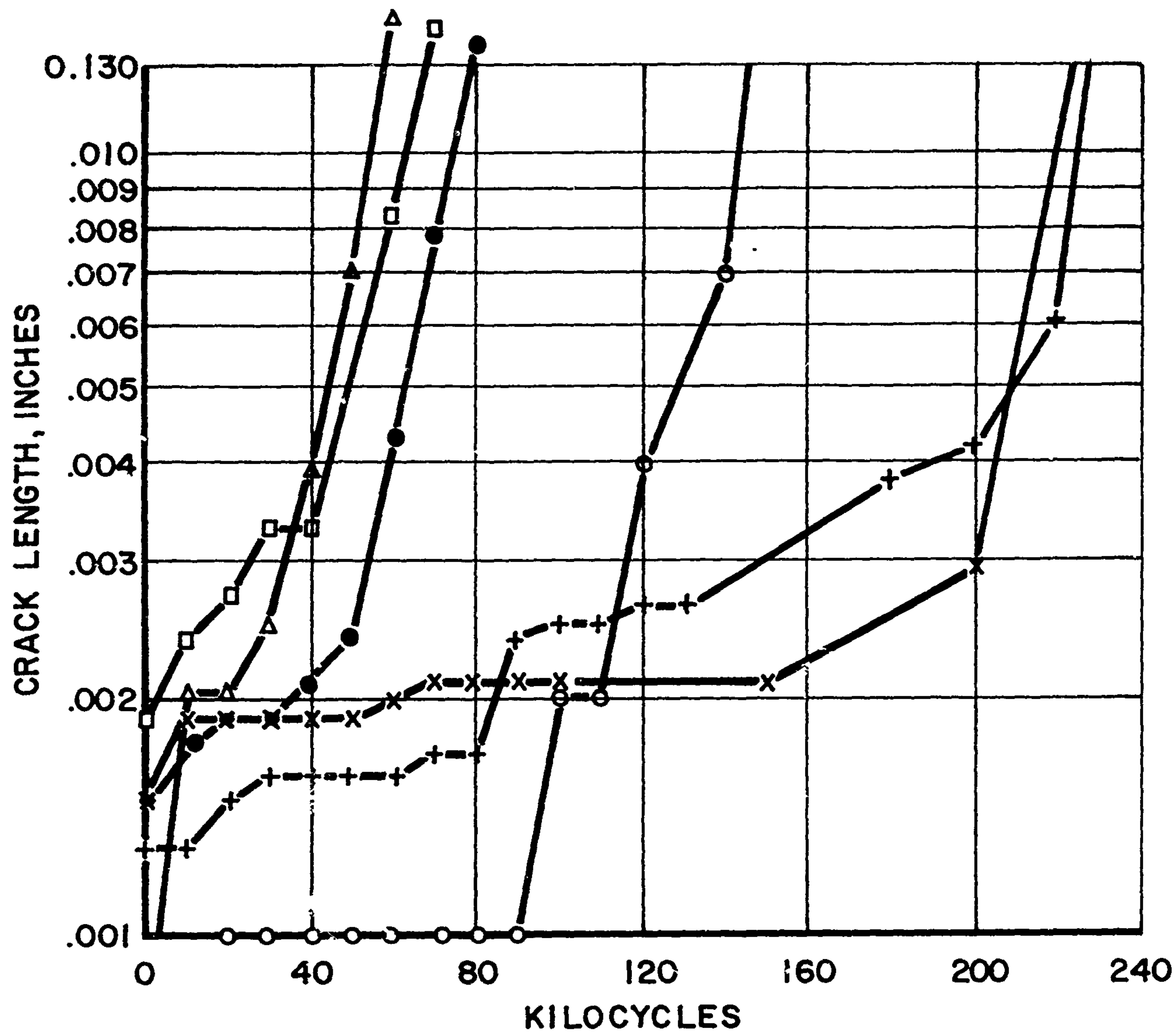


FIG. 11

EARLY CRACK GROWTH OF SIX SPECIMENS OF SAE 4340 STEEL OF 190 ksi UTS STRESSED AT 120 ksi

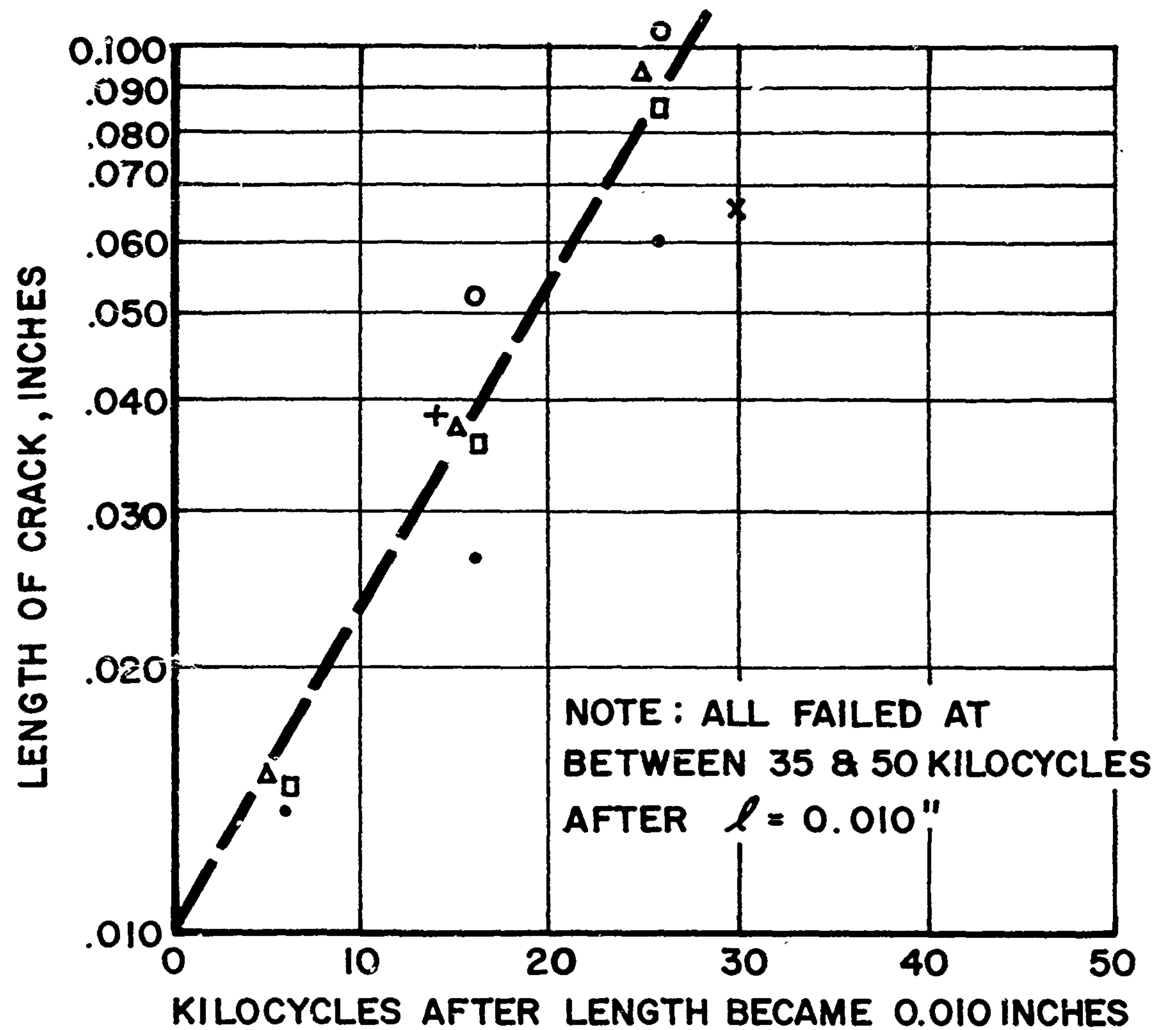


FIG. 12

CRACK GROWTH OF SIX SPECIMENS (SEE FIG. 11) OF SAE 4340 STEEL OF 190 ksi UTS, STRESSED AT 120 Ksi, AFTER CRACK LENGTH BECAME 0.010 INCHES

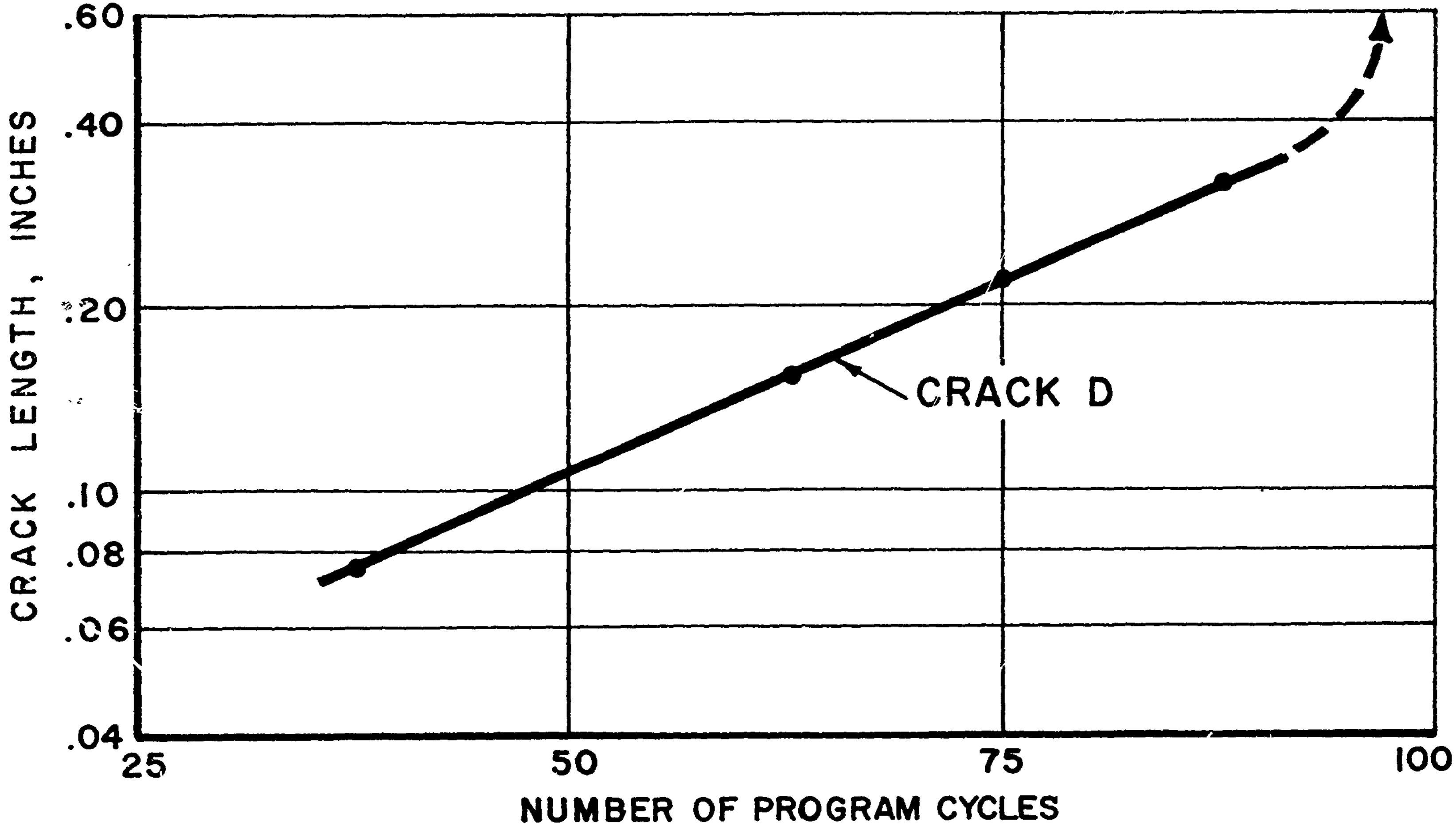


FIG. 13

CRACK PROPAGATION FOUND IN AN ALUMINUM WING SPAR BY RYDER

Ryder's statement that each striation corresponds to one stress cycle. Furthermore in rotating-beam tests, the compressive part of the cycle would probably obliterate these striations.

Lipsitt and co-workers (39) believe "the growth of a fatigue crack to be a discontinuous, non-linear, stepwise process". Their tests were made at a very slow rate of cycling - only eight cycles per minute - which enabled them to observe numerous "hesitation" periods during the life of each specimen.

The foregoing discussion on crack nucleation and propagation was presented so that rational correlations of fatigue characteristics with other material properties may be uncovered. At stresses near or slightly above the long-life fatigue strength, the nucleation stage correlates with the size of the nucleating inclusion. On the other hand, the large number of grains located at the tip front in the last stage of crack growth suggests that the last stage might correlate with some bulk characteristic of the metal, such as ductility or "notch" toughness (at a given tensile strength). At high stresses, the nucleation stage appears to be dependent mainly on the number of inclusions as well as their maximum sizes (15). At high stresses many cracks form at low cycles and eventually join to form the final fracture surface.

One general observation concerning the S-N curve that has been made by several investigators is that if strain is used instead of stress, then many materials having good ductility will fall within one straight-sided "scatter band" on a log strain versus log N plot at sufficiently high values of strain. This is shown in figure 14. Low (42) tested two aluminum alloys and three steels and found that these fell within the band in figure 14, except in the range below the yield points where the curves deviated to the right of the band. Low found that when the strain was less than that corresponding to the yield point, the curve departed rapidly from linearity. If the strain in percent is designated by ϵ, then the equation of the mean line in this scatter band was found to be described by the following relation:-

$$\epsilon N^{0.40} = 14.3$$

It is interesting to note that the two aluminum alloys had the least toughness as roughly measured by the product of their tensile strength and elongation, and that these also had the smallest lives at constant strain. The three steels had the highest toughness and the greatest lives throughout most of the upper range.

Some authors have expressed the point of view that fatigue including crack propagation is related to the depletion of ductility in the metal around the nucleus prior to the formation of a crack, and in the region in front of the crack during crack propagation. Sachs (43) presented some experimental data in support of this viewpoint. Orowan's original theory (44) is somewhat similar in this respect. McClintock's theory (45) of crack propagation in plastic torsion is also based on this assumption.

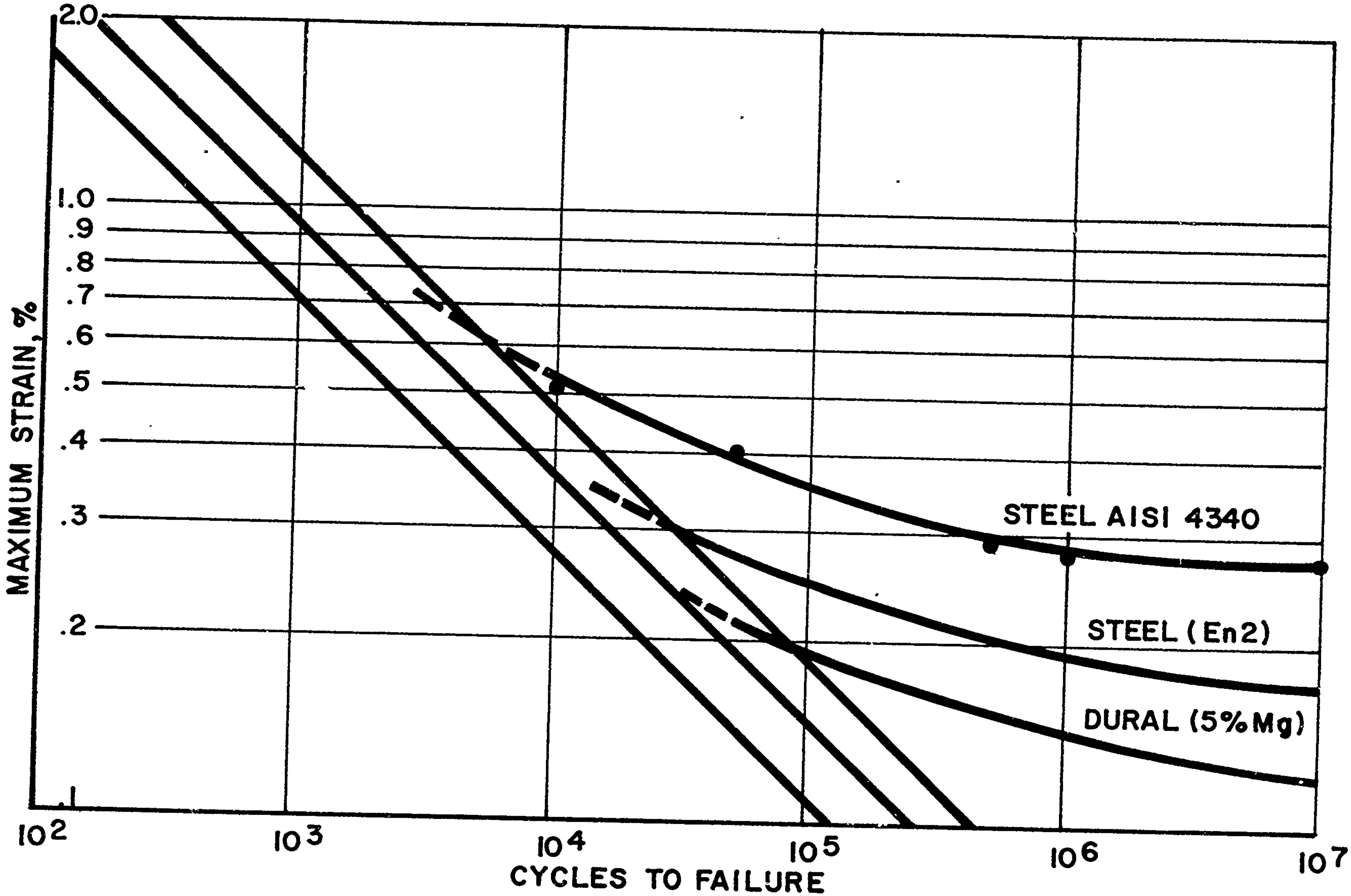


FIG. 14

LOG STRAIN AS A FUNCTION OF LOG N FOR LARGE STRAINS IN VARIOUS METALS

CRITICAL CRACK LENGTH

In some parts that have failed in service by fatigue, the final fracture often happens at a very rapid rate. Prior to this final rupture the fatigue crack progresses at a finite, but increasing rate until it reaches some critical size such that the next cycle of stress causes complete fracture. Partly responsible for this critical condition is the fact that, for a fixed load, the decreasing area increases the nominal stress. However, even had the nominal stress been uniform (by decreasing the cyclic load) a critical crack size might be encountered.

The practical importance of having the critical crack size sufficiently large is obvious. In aircraft structures, if the critical crack size is much greater than that which can be easily detected at overhaul and its attainment requires a number of service overhaul periods, the condition may be acceptable. If, on the other hand, the critical crack size for the maximum stresses imposed is so small as to be readily missed during inspection, a potentially dangerous situation may exist. Since only one load application (or perhaps a few) is necessary after the critical size is reached and since this critical size decreases with increase in stress, the maximum load that occurs when the crack is a given size is a measure of the safety.

Another practical example is the limiting size of defect in high-strength solid-propellant missile cases. Using existing data on steels, critical defects in the steel are estimated to be very small if the design stress is over about 200,000 psi.

Most laboratory fatigue specimens tested at constant load finally fail from excessive stress and display a more-or-less static type of failure. The size of the specimen and the relatively low cyclic stress precludes the attainment of the critical crack size in most laboratory specimens. For example, using the relation discussed by Irwin (46) for the critical crack size and his factor for 4340 steel, the critical crack size is about the same or larger than the periphery of a small rotating beam specimen at fatigue stresses near the fatigue limit. However, some recent fatigue tests on very hard steels conducted at relatively high stresses showed evidence that the critical crack was only 0.05 to 0.10" and was followed by rapid failure. In these tests, the fractured surface displayed two distinct types of areas, one representing the original fatigue crack and the other a rapid fracture. Figure 15 is a photograph of a typical fracture surface of one of these specimens. Figure 16 is a plot of the critical crack size measured on the outside surface as a function of the test stress.

Irwin has revised the Griffith theory for brittle materials to be applicable to engineering materials. His relation (which is similar to Griffith's original relation) is that the product of square of the net-section stress and the critical length is proportional to a certain material constant. This constant is dependent on the degree of constraint of the metal at the tip of the crack, that is, plain strain causes a greater restraint than plain stress, hence a much lower value of the constant. His relation for the critical crack size in a plate or sheet is as follows:-

FIG. 15

FATIGUE FRACTURE, SHOWING FATIGUE FRACTURE AREA (LIGHT) AND FAILURE AREA (DARK)

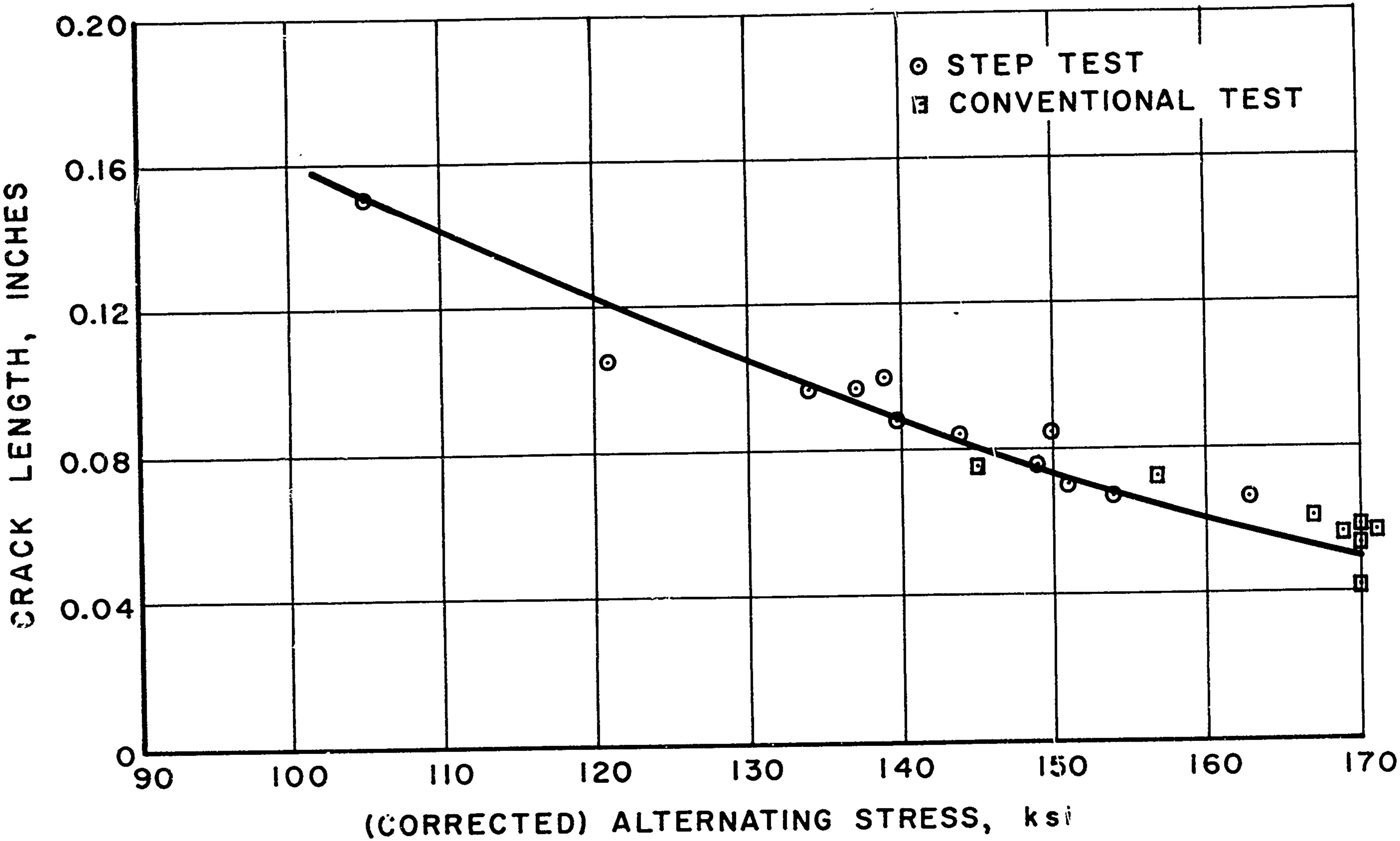


FIG. 16

CRITICAL CRACK LENGTH OF ONE STEEL AS FUNCTION OF STRESS (CORRECTED FOR OFF-CENTER FRACTURE)

$$\sigma^2 \ell_c = \frac{2 E G_{IC}}{\pi (1-v^2)}$$

where:

σ = nominal net-section stress, psi, to propagate the crack

ℓ_c = critical crack length in inches

E = modulus of elasticity, psi

v = Poisson's ratio

G_{IC} = Energy to extend crack through one square inch of metal in in. lbs. per in.^{2} (for plain strain)

McClintock (47, 48) has treated this problem theoretically in some detail. The critical crack size has several important relations in considering the fatigue life of a part or structure. If fatigue cracks do develop in a part in service, the above relation determines the crack size at which the maximum expected load in service will cause complete collapse of the structure. In the estimate of the "cumulative fatigue damage" of a part in service, the analysis should be based on the growth of the fatigue crack and its length compared to the critical value for the maximum expected load.

SUMMARY

This paper is an attempt to disclose some of the relations of material variations on fatigue. It has been found that the fatigue strength of an alloy is a function of its tensile strength, its cleanliness, isotropy, homogeneity, microstructure, microstresses and perhaps some measure of ductility. It is doubtful that these variables are, in themselves, completely independent. For this reason, much work is to be done in the future to uncover those basic factors in the structures of metals that determine their fatigue properties.

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APPLICATION OF REFRACTORY METALS AT HIGH TEMPERATURES

by

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With the ceaseless demand for higher temperature structural materials for flight system components, metallurgical attention has concentrated upon development of the refractory metals (e.g. Cb, Mo and W) whose high melting points give them the potential for useable strength in structural applications above 2000F. Alloys with good strength have been developed, but their engineering application has been generally compromised by the fact that they react rapidly with high temperature air and the protective coatings employed oftentimes fail due to their inability to withstand the service stresses involved in structural applications.

Only by the combined efforts of materials and structural engineers can this problem area be overcome.

Looking to the future there is confidence that the problems of today will be overcome and that the technology of refractory metals will be continuously advanced for use in man's further technical progress.

This paper is not on fatigue. Its inclusion in this symposium might be questionable were it not for the fact that it affords the opportunity - all too rare these hectic days - for a practitioner of one technical discipline to talk to practitioners of another on problems which concern them both. The remarkable developments wrought in the flying machine by aircraft designers have resulted in urgent pressures on my fellow materials

engineers and me to supply materials capable of operating under heretofore unthinkable conditions.

We shall develop, and are developing, such materials. And this we know so far: Designers will experience new and unusual problems with these new materials when designing structures which operate above 2000F. Now as never before must the designer and the materials engineer work in close harmony to surmount the obstacles facing us both.

My remarks, therefore, will be aimed at stimulating you who are interested in structural design thoughtfully to consider both the opportunities and the problems associated with building high temperature structures from the materials which the materials engineer has provided.

WHY REFRACTORY METALS?

The breathtaking advances in flight vehicle technology within the past few years have propelled us into the supersonic flight regime and even to the threshold of space flight. This headlong advance has made one technological factor penetratingly clear: Flight vehicle structures and propulsion systems will be forced to operate at temperatures where conventional materials of construction are grossly inadequate. See Figure 1. Note that operating temperature requirements of some of the various flight system components pointed out are well above 2000F, usually considered the maximum temperature for structural application of the conventional nickel and cobalt base alloys. In fact these materials become melting point limited not far above 2000F. Only the high melting point (i.e. refractory) metals, therefore, can be of use for structures operating much above 2000F. (Ultimately even the refractory metals will be inadequate and we will see structures made from ceramics and graphite.)

Though there are a number of refractory metals, tungsten, molybdenum and columbium have received the most attention from metallurgists in the search for structural metals useful above 2000F.

REFRACTORY ALLOY DEVELOPMENT

Efforts at developing strong refractory metal base alloys have borne fruit. Today the metallurgist can offer the designer alloys with useful strength up to perhaps 2400F-2500F. In Figure 2 the design strengths (based on .2% yield strength and 100-hour stress-rupture strength) of recently developed columbium and molybdenum alloys are compared with those of the best known nickel-base alloys. Let us assume the useful strength range for elevated temperature applications lies between 5,000 psi and 30,000 psi. It is obvious that the nickel base alloys possess little useful strength above 2000F. On the other hand, the columbium and molybdenum alloys offer useful strengths at least to 2400F. (F48 and F50, mentioned in Figure 2, are two in a series of strong columbium alloys recently developed by General Electric metallurgists.)

Additional strength information is given in Figure 3 where individual alloys are shown. In addition to the strengths of Cb and Mo alloys, that of unalloyed W is also shown. Only a modest amount of development work has been done on W alloys and no tungsten alloy has yet been identified which, on a strength-to-weight ratio

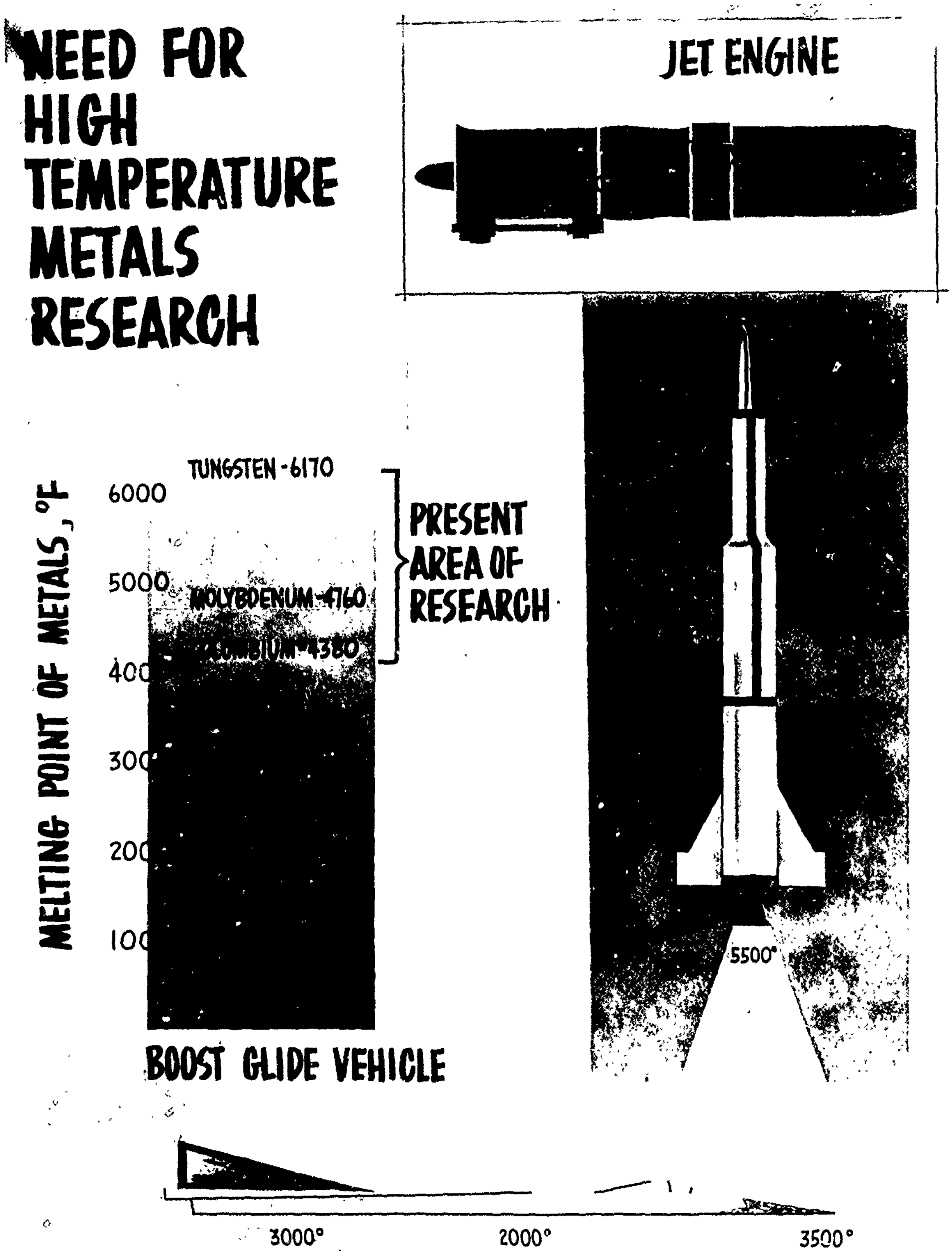

Figure 1

Operating temperature requirements of some flight system components

REFRACTORY METALS
Alloy Development
STRENGTHS

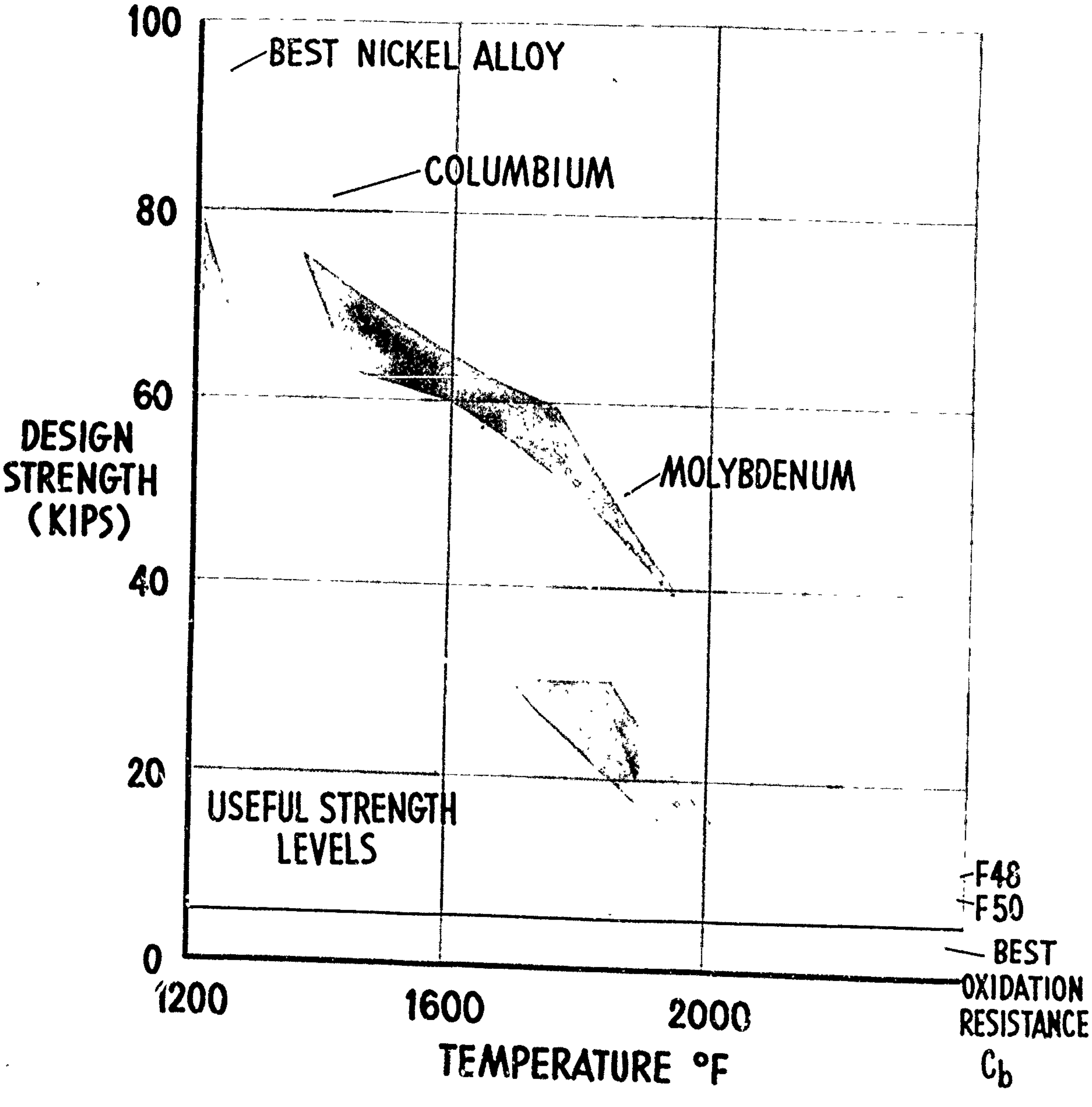

Figure 2

Design strength ranges for columbium and molybdenum alloys as compared with nickel base alloys

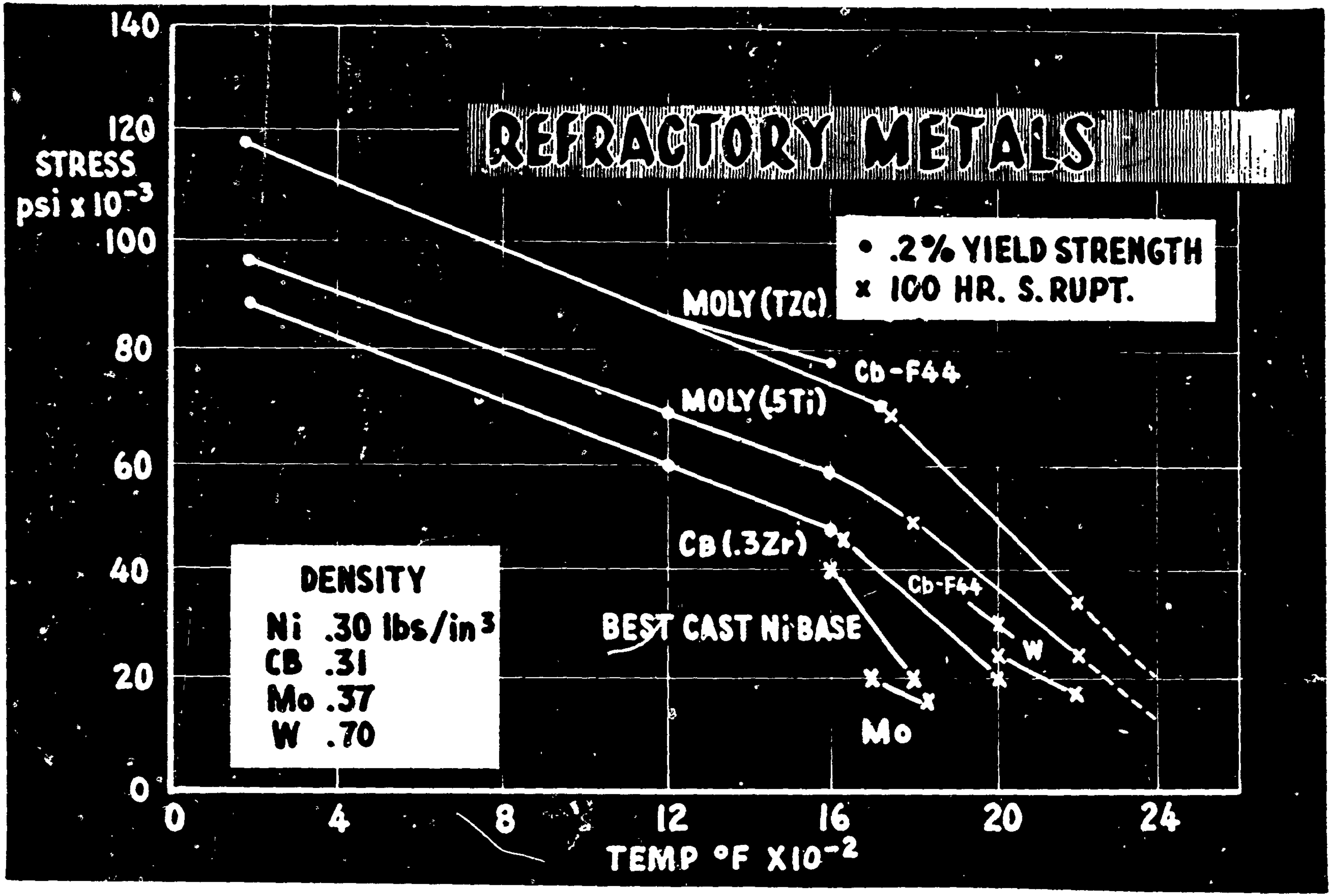

Figure 3

Strengths of Refractory Metals

basis, is as strong as the columbium and molybdenum alloys up to about 2500F. Tungsten will find its true potential at temperatures substantially above 2500F and its technology will be advanced as clear cut needs are defined.

It is evident from the foregoing then, that strong refractory alloys are at the designer's disposal. Their strength stands as a vital and powerful tool to aid in the design of high temperature structures. Furthermore, many potential applications for these materials have been identified (Figure 4) with respect to various components of jet engines, space power plants and re-entry vehicles.

In view of this situation, i.e. strong materials and waiting applications, what else must be considered before materials and applications can be introduced to each other? The answer to this is really the crux of the whole matter.

Actually there are a number of rather formidable problem areas which prevent easy and widespread structural application of refractory metals. Some of these are the concern of the materials engineer, but others must be faced by you who are expert in design.

PROBLEM AREAS

I will not regale you with the materials engineers problems. Suffice it to say that we must learn to produce these refractory alloys in useful form with greater efficiency and at substantially lower cost. We must learn better to form and join them with precision and reliability. We must develop more effective surface protection methods. These and many other problems we are attacking - and will ultimately solve.

On the other hand, I hope that designers are considering, or will consider, those characteristics of refractory alloys which will undoubtedly complicate their job in trying to adapt them to realistic structural design.

Here are some typical problem areas. Refractory alloys are:

1) Relatively heavy,
2) Brittle at low temperatures under high strain rates,
3) Highly susceptible to oxidation.

Certainly you all are familiar with these attributes. Some of you may have already contemplated design innovations aimed at alleviating the weight problem. Beyond this, you know that you must take steps to preclude high strain rate loading of refractory alloy structures at low to moderate temperatures, for such could cause sudden, brittle failure of the structure. (A hammer, or other tool, dropped by a workman is a good means, incidentally, of producing high strain rate loading and may be difficult for a designer to control.)

Finally, you are all aware that the refractory alloys lack oxidation resistance and must have protective coatings. I wonder, though, if all of you have thought about the very marked design ramifications associated with the protection-from-oxidation problem.

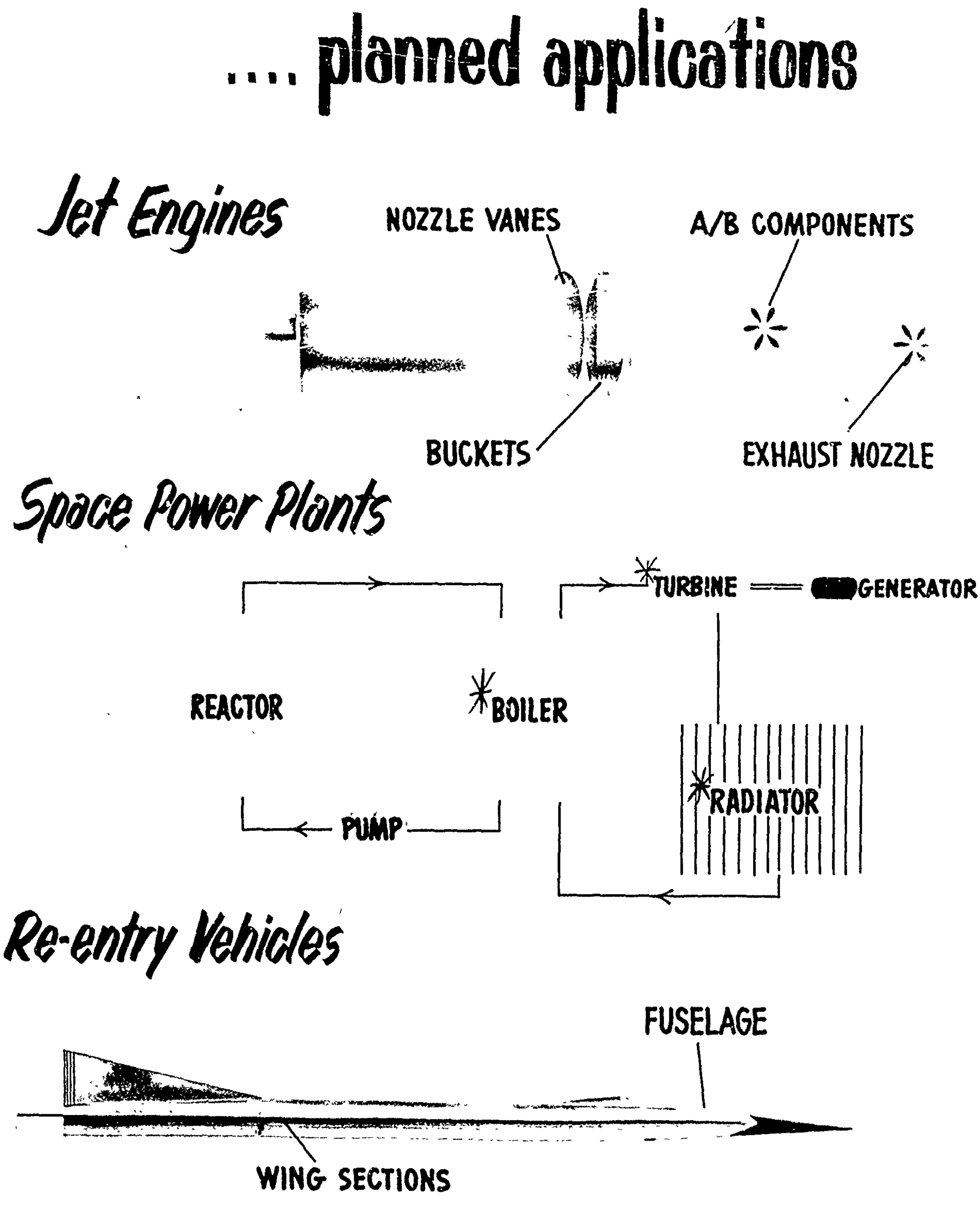

Figure 4

Planned applications for refractory alloys

COMPLICATIONS DUE TO COATINGS

Because of the oxidation problem a vast and complicated technology involving protective coatings has arisen. From this technology have been derived important lessons. Prominent among them are these:

1) Coated refractory metals must be regarded as composite material structures having oxidation resistant, non-stress carrying surfaces and strong, reactive cores.

2) An extremely high percentage of all failures of coated refractory metal parts can be traced to coating failures, which in turn lead to parent metal failures.

This makes it apparent that when designing with refractory alloys it is not realistic to consider only the properties of the parent metal - those of the coating must also be taken into account, for in many cases the properties of the coating will determine the life of the coated part. And remember, the coatings of which I speak will probably be composed largely of ceramic materials whose properties are vastly different from those of the metallic substrate.

To appreciate the significance of the foregoing concept, consider the following factors:

1) Oxidation mechanisms
2) Failure mechanisms for coatings

In Figure 5 are illustrated the oxidation mechanisms which occur in molybdenum and columbium, respectively. In the case of molybdenum, which forms a high vapor pressure (i.e. volatile) oxide, continued exposure to an oxidizing atmosphere would result in continued metal loss and consequent decrease in load carrying ability. Columbium, on the other hand, forms a low vapor pressure (non-volatile) oxide which does not result in rapid and severe metal loss as in the case of molybdenum. Oxygen, however, diffuses into the parent columbium with consequent embrittlement of the oxygen contaminated area. While the deleterious effect of oxidation on columbium is much less than on molybdenum, severe contamination of a columbium structure (as could result from a coating failure) can cause brittle behavior of the structure in service and possible premature failure. (Columbium alloys, however, have such significant advantages over molybdenum alloys with respect to oxidation that they are considered much more reliable for application in the 2000F-2500F range.)

It should be clear, then, that once a coating failure occurs in a refractory alloy component, failure of the component itself certainly is possible, and oftentimes is likely, particularly in the case of molybdenum. It might be said that the structure is "conditioned" for failure. Thus, it follows that the performance in service of a coated refractory metal component is in large measure determined by the performance of its coating.

Now then, what causes coatings to fail? While failures can occur simply due to inadequate oxidation resistance, such failures actually are relatively rare. More likely are failures associated with coating-parent metal relationships and with service operating conditions.

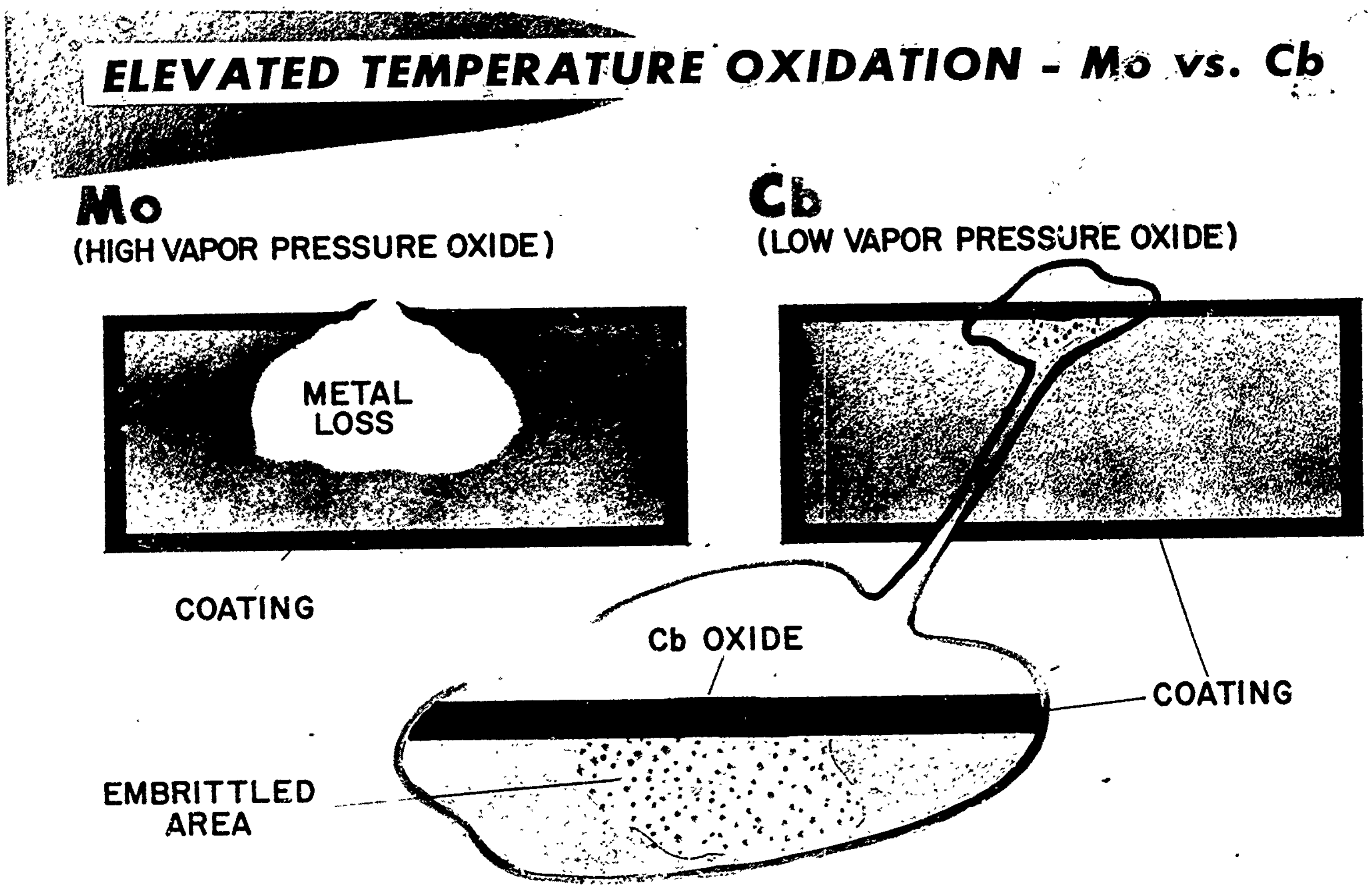

Figure 5

Oxidation mechanisms - molybdenum and columbium

The Flight Propulsion Laboratory Department spent several years studying the feasibility of using coated molybdenum turbine buckets in jet engines. During this time the causes of coating failures were well established. Figure 6 illustrates these coating failure causes, as experienced during engine operation of coated molybdenum turbine buckets. The causes are: 1) Overtemperature, 2) impact (or other mechanical damage), and 3) thermal fatigue. The first two relate to operating conditions; the third to a combination of operating conditions and parent metal-coating relationships. If in addition to the three failure mechanisms just mentioned we add a fourth vibratory fatigue, then all the likely causes of coating failure in a refractory alloy structure are covered.

SIGNIFICANCE TO THE DESIGNER

I have made the proposition that in structural design employing refractory alloys the properties of the coating are, in the final analysis, probably more significant than those of the parent metal. I have specified the causes of coating failure. Now what does this really mean to the designer? In the case of overtemperature, coatings can actually be melted off in but a few seconds. With respect to mechanical damage (such as impact or erosion) foreign particles, meterorites, etc. can penetrate or wear away a coating. While the materials engineer can help here by providing higher melting and more impact and erosion resistant coatings, the designer ultimately must shoulder the responsibility of deciding how to design for these conditions. In other words, he must decide what risks to take.

Consider next the case of thermal fatigue - the repetitive application of thermally induced stresses. (Thermal shock is here considered as a special case of thermal fatigue.) This has been one of the most vexing problems to cloud the coating picture. Severe thermal stresses in a coated refractory alloy component can result from:

1) Differential thermal expansion between elements of the component (e.g. as between coating and parent metal).

2) Sharp thermal gradients in the component (e.g. as might be caused by rapidly heating a part with a low conductivity coating).

3) Mechanical restraint between elements of the component (e.g. as at junctures between light and heavy sections).

Certainly you have faced these thermal stress situations before when designing structures. But how often have you done so when the difference between success and failure of a component rested with a relatively fragile coating (probably a ceramic) which was never meant to be strong but now must survive repeated doses of thermal stresses? You must agree that this is a situation to be approached with all the skill at the designer's command. Adequate design compensation might be made if only one of the three mechanisms were operative. But if two, or all three, were extant the prospect of coping with them by design methods is something less than inviting. Yet I am sure that if proper foresight and skill are applied to such problems the thermal stress situation can, in many cases, be turned to the advantage of the designer.

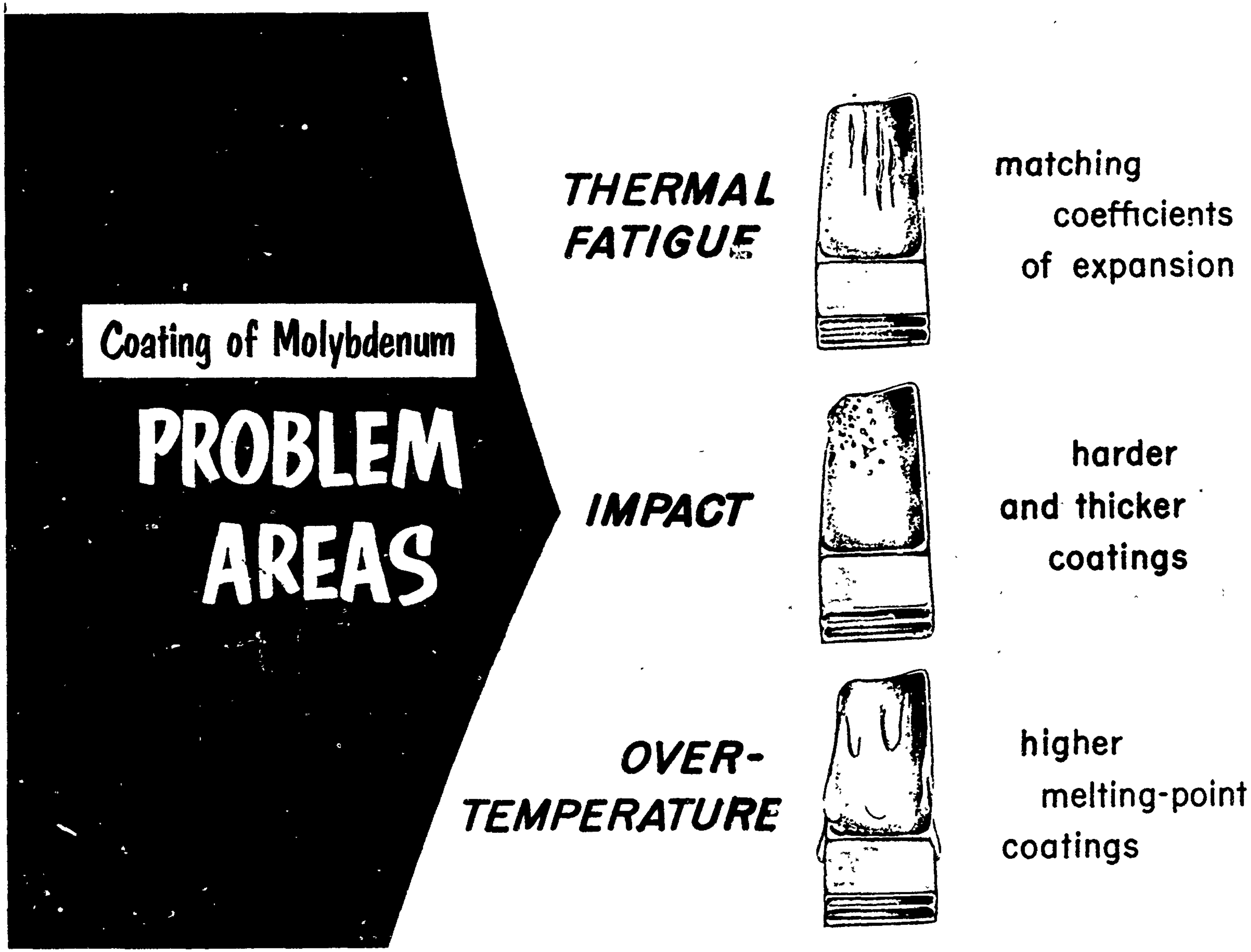

Figure 6 Causes of coating failures in molybdenum turbine buckets

Finally we must think about vibratory fatigue. This phenomenon, of course, adds complexity to the already formidable problem of thermal fatigue. Investigatory work within the Flight Propulsion Laboratory Department has demonstrated the marked reduction in coating life that occurs with all types of coatings when vibratory stresses are imposed simultaneously with thermal stresses on coated refractory metals.

Testing techniques used to demonstrate this factor involve use of a laboratory flame tunnel programmed for automatic thermal cycling and having provision for application (if desired) of vibratory stress to the test specimen.

A single example will be cited. It concerns .5%Ti-Mo alloy coated with G.E. System 300 coating (Cr plate + flame sprayed aluminum oxide). For thermal cycling tests coated turbine buckets were used. For combined thermal cyclic-vibratory stress testing specially designed coated panels were used. The difference in configuration of test specimens does not materially affect, for our purposes, comparability of test results, which are shown below:

Table I

Test Specimen	Test Conditions: Thermal Cycling	Test Conditions: Vibratory Stress	Average Life (Cycles)
1 *Coated Turbine Buckets	1000F-2200F, 2 Mins.	-	2000
2 *Coated Panels	1000F-2200F, 2 Mins.	**10,000 psi	775

*Cr plate + Al_2O_3

**Single amplitude, 20 cps

The definite decrease in coating life attributable to vibratory stress is evident. If the number of thermal cycles withstood prior to failure seems large, remember that these tests were run on specimens of relatively simple configuration, protected with a high quality coating. Moreover, none of the other normally expected service conditions was present (i.e. structural loading, impact, erosion, etc.) and no really complex structural designs were involved.

Again, however, I wonder whether the vibratory stress factor might not be made to work advantageously for coating life and thus for structural life.

From all the foregoing, certain things seem clearly apparent. The development of strong refractory alloys has presented the structural designer with potentially valuable materials of construction. At the same time, however, it has confronted him with the prospect of being forced to design around presently inherent weaknesses largely associated with protective coatings. Certainly here is a challenge to you who are designers to pit your skills and ingenuity against the obstacles. And it would seem there is a particular challenge for you who are interested in fatigue mechanisms. That fatigue is a prime cause of failure in protective coatings seems well established. Yet it is true that little work is being done to define and understand the fatigue characteristics of coatings. Work in

this area could result in a significant contribution to the successful use of refractory alloy structures and I commend it for your consideration.

A final point. I am certain that you specialists in design and fatigue mechanisms simply cannot win the battle alone. You must have the help of us materials engineers who know and understand the metallurgical aspects of coatings. So my plea to you is this. Join forces with us so that our pooled talents can be brought to bear on the problem. Will you accept this challenge?

ABOUT THE FUTURE

As surely as we face problems today, these will be overcome and the technology of refractory alloys will be advanced. It is certain that the useful temperature limits of these materials will push steadily upward. As can be seen from Figure 7 the maximum temperatures for useful strengths of many alloy systems exceeds 60% or even 70% of the melting point of the base metal. If the same can ultimately be expected of refractory alloys then, obviously, they have noticeable room for growth.

As to what the ultimate useful temperature limits will be and how long it will take to attain them Figure 8 provides estimates which are based on current rates of progress and on observed development histories with other alloy systems.

Refractory alloy technology is really still in its youth - though a vigorous youth it is! You and I will see it grow and hopefully will help its growth to maturity as a potent force for us in mankind's future technical progress.

HIGHEST TEMPERATURES at which today's best heat-resistant alloys can be used

	Base metal	Melting point	Temperature for useful strength of best alloys*	Percent of melting point**
LIGHT ALLOYS	Mg	1200°F	650°F	67
	Al	1220	550	60
	Ti	3100	1200	46
SUPER ALLOYS	Fe (mart.)	2800	1350	56
	Fe (aust.)	2800	1600	63
	Ni	2650	1960	78
	Co	2720	1900	74
REFRACTORY ALLOYS	Cb	4470	2200	54
	Mo	4760	2650	59
	W	6170	2550	45

* withstanding 10,000 psi for 100 hours

** per cent of absolute melting point at which alloy is useful

Figure 7 Useful temperature limits for various types of alloys

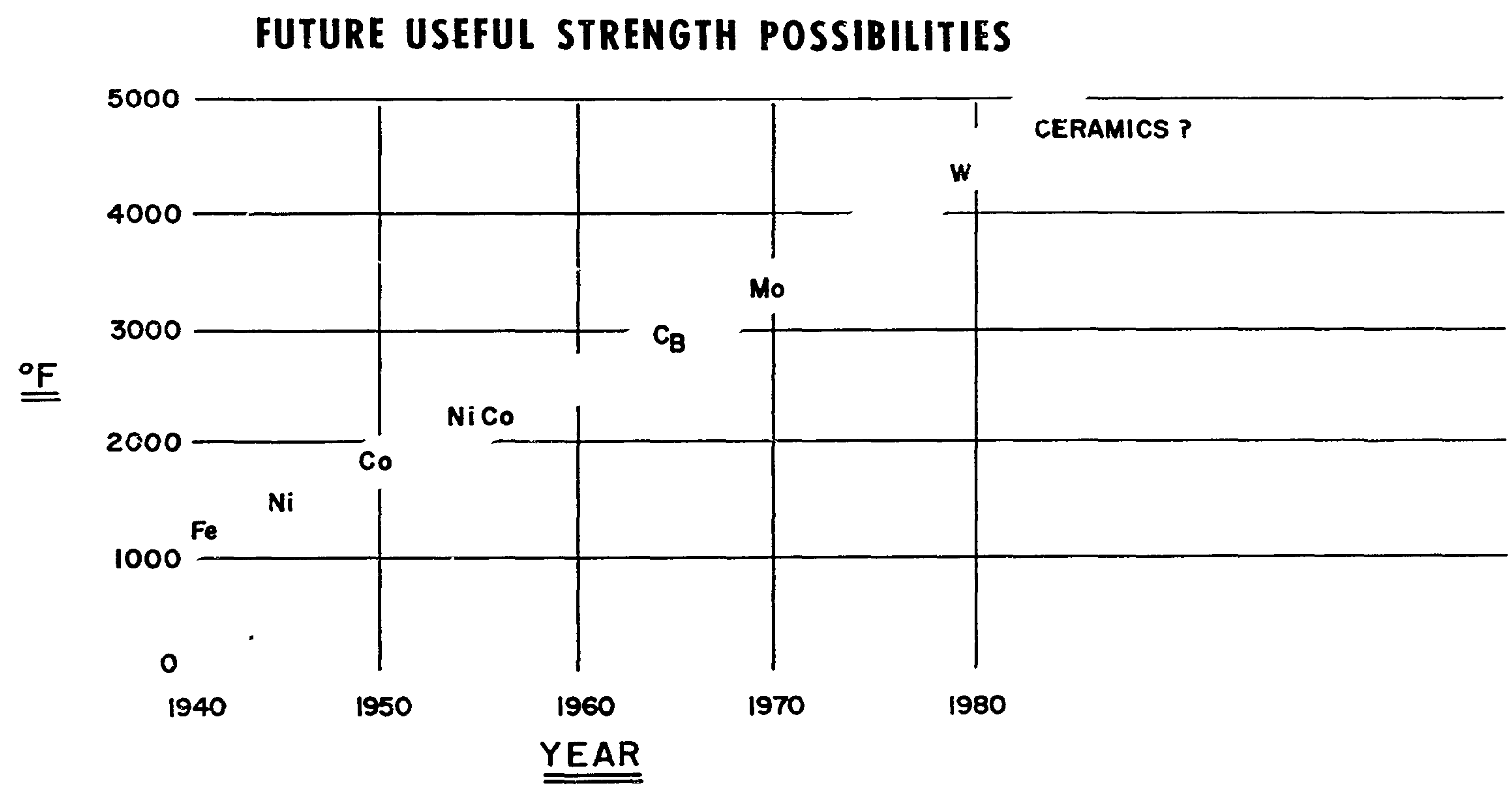

Figure 8

Future useful strength possibilities

AN APPRAISAL OF THE FATIGUE CHARACTERISTICS OF MATERIALS FOR HIGH PERFORMANCE AIR VEHICLES

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ABSTRACT

This paper is concerned with the fatigue characteristics of new structural materials being incorporated into advanced air vehicles. Environments of higher service temperatures and the demand for higher structural efficiency have led to the use of a new class of materials including ultra-high strength steels, and high temperature alloys. Relatively little fatigue information has been available on these materials and considerable effort is being expended to obtain fatigue data in order to support design. In comparing the test results obtained thus far on these new materials with the well established data on conventional low alloy steels and high strength aluminum alloys, certain similarities are noted. Fatigue properties are also being determined at elevated temperatures in view of the higher service temperature environments. An analysis of the data from fatigue test programs shows no cause for alarm in using these new materials for advanced air vehicle structure and indicates satisfactory service life will be obtained.

INTRODUCTION

The advanced air vehicles being designed in this country to meet tomorrow's competition have mission requirements which dictate the use of new structural materials. Longer range means trading structural weight for additional fuel. This dictates the use of materials with higher structural efficiency (viz., higher strength/weight ratio). Higher speed means higher operating temperatures. This dictates the use of materials which retain useful properties at higher temperatures. In addition to the introduction of new materials, new types of air vehicle structure such as welded or sandwich construction are being used in place of the riveted skin and stringer type. In considering these new materials and types of construction, it was apparent that relatively little fatigue information was available. The great wealth of data on high strength aluminum alloys and riveted joints was not considered directly applicable. The service life characteristics of these new materials and structures was a question mark which had to be answered by test. The purpose of this paper is to present some of the preliminary test results obtained on materials typical of those being considered for advanced air vehicles. Included also are some data on welded joints made from these materials. The fatigue data presented are for axial tension ($R = 0$) loading and are at room temperature unless otherwise noted.

MATERIALS SELECTED

Four general types of materials have been selected for discussion in this paper. They are: (1) hot work die steel, (2) semi-austenitic precipitation hardening stainless steel, (3) titanium alloy, and (4) nickel-base alloy. The hot work die steel has been included as being an ultra-high strength steel which, because of its high tempering temperatures, retains useful strength up to about 900 F. The semi-austenitic precipitation hardening stainless steel has been selected because of its combination of corrosion resistance and high strength, with useful properties up to about 850 F. The titanium alloy was chosen as a low density material with good corrosion resistance and useful strength up to about 800 F. The nickel-base alloy has been included as a high temperature alloy with high temperature oxidation resistance and good strength up to about 1600 F.

HOT WORK DIE STEEL

The data presented are on the SAE H-11 alloy steel bar. This alloy is a 5% chromium tool steel which has been used extensively for hot working tools in operations such as forging, heading, and swaging. The steel is capable of being heat treated to a strength level of 280,000 - 300,000 psi and will retain its high strength after extended periods of exposure to temperatures up to approximately 900 F. Figure 1 shows the effect of time and temperature on the static tensile properties of this alloy. The H-11 alloy steel is not corrosion resistant and thereby loses some of its structural efficiency advantage over corrosion resistant material because of the additional weight of the protective finish required. For this reason, the

greatest advantages in using this steel are realized in applications such as heavy structural fittings where the weight of the protective finish is negligible by comparison. Figure 2 presents some SN curves at room temperature for heat treated H-11 steel in the unnotched and notched conditions, under axial tension (R = 0) loading. The data, which are plotted as a percent of the static tensile strength, look quite similar to data we are familiar with for low alloy steel such as the 4100 or 4300 series, heat treated to lower strength levels. Endurance limits equal to 42% of the unotched tensile strength were obtained in the unotched condition and 24% in the notched (K_t = 2.5) condition. Figure 3 shows a comparison of the data for H-11 steel bar, from Figure 2 with SN curves for the 4100 and 4300 low alloy steel bar, heat treated to 180,000 - 220,000 psi. The data are plotted as a percent of the static tensile strength for comparison. Also plotted on the graph are unnotched fatigue data for the H-11 steel at a temperature of 700 F. The elevated temperature fatigue data are plotted as a percent of the short time elevated temperature static tensile strength. The 700 F fatigue properties are actually more meaningful to the aircraft designer because the major use for this material on advanced air vehicles will be at elevated temperatures. The data indicate no significant difference in the unnotched fatigue characteristics for H-11 steel at 700 F as compared to room temperature. Although the SN curves for H-11 lie below those for the low alloy steel, when plotted as a percent of the tensile ultimate strength, a comparison on the basis of actual stress in terms of psi shows that H-11 steel provides an advantage over low alloy steel. This latter comparison is made Figure 4. Further, the advantage of using H-11 steel increases at higher temperatures where the strength of low alloy steels decrease rapidly.

SEMI-AUSTENITIC PRECIPITATION HARDENING STAINLESS STEEL

The data presented are on the PH15-7Mo, Condition RH 950, alloy sheet. This alloy is a comparatively new development in the precipitation hardening grade of stainless steel. Its principle advantages are, (1) high strength at elevated temperatures, (2) weldability, (3) brazeability, (4) corrosion resistance, and (5) availability in the various product forms required for brazed sandwich construction. The elevated temperature static tensile properties of PH15-7Mo alloy sheet are presented on Figure 5. This material is being considered for skins and sheet metal structure, including weldments and brazed sandwich, for advanced air vehicles. Axial tension fatigue data are shown on Figure 6 for PH15-7Mo in the unnotched, notched (K_t = 2.5), and welded conditions. The welded test specimens were fusion welded, using the tungsten, inert gas, shielded arc method, heat treated after welding, and tested with the full weld bead normal to the loading direction. Endurance limits of 38% of the static tensile strength were obtained in the unnotched condition, and 20% in the notched condition for PH15-7Mo sheet. The welded specimens gave similar results to the notched specimens. Analysis showed that PH15-7Mo had somewhat lower fatigue properties than 17-7PH, Condition TH 1050, (a familiar and extensively used predecessor to PH15-7Mo) in the unnotched condition but slightly higher in the notched condition. A comparison of the SN curves for PH15-7Mo and 17-7PH is made in Figure 7.

Also, appropriately included on Figure 7 are fatigue data for unnotched PH15-7Mo at 500 F. As in the case of the H-11 alloy steel, the differences between the elevated and room temperature fatigue properties presented on PH15-7Mo cannot be considered significant. In addition to reviewing the basic fatigue data on PH15-7Mo, it is important to consider its fatigue characteristics when used in typical vehicle structure. In comparing a welded PH15-7Mo joint with a riveted 2024 - T3 aluminum alloy joint, we find the PH15-7Mo joint is superior from a fatigue standpoint. This comparison is shown in Figure 8 for PH15-7Mo welded joints with the weld bead left on and also ground flush, and for 2024 - T3 lap riveted joints with both a single and double row of rivets. It should be noted that the ordinate on Figure 8 is shown as a percent of the parent metal strength. This takes the joint efficiency factor into account and permits a more realistic comparison. On the basis of these comparative data, the use of welded PH15-7Mo structures should present no greater fatigue problems than we are accustomed to in the use of conventional riveted aircraft structure made from commonly used high strength aluminum alloys.

TITANIUM ALLOY

The fatigue data on titanium presented in this paper are on the 6Al-4V alloy bar, heat treated to 160,000 psi minimum tensile strength. This alloy is available in many product forms, including sheet, bar, forging, and extrusion. Its main advantages as a titanium alloy are its combination of high strength and weldability properties, plus good retention of strength at elevated temperatures. The elevated temperature static tensile properties of the 6Al-4V alloy are shown on Figure 9. This alloy is being considered for skins and frame structure on advanced air vehicles, and is primarily in competition with the semi-austenitic precipitation hardening stainless steels such as PH15-7Mo and AM 355. The titanium alloy has the advantage of low density coupled with high strength but is more difficult to weld and has not yet been satisfactorily made in the form of brazed sandwich structure. Figure 10 shows some limited axial tension fatigue test data on heat treated 6Al-4V titanium alloy bar in the unnotched and notched ($K_t = 3.3$) conditions. While these data are not on sheet material as is the case with the PH15-7Mo stainless steel, the fatigue properties are relatively good. Certainly no dangerous trend in fatigue resistance is indicated in this material. Endurance limits at 10^7 cycles were 76% of the unnotched tensile strength in the unnotched condition and 22% in the notched condition. It should be noted that the notched data on the tianium alloy are for a notch factor of 3.3 as compared to a factor of 2.5 for other materials discussed in this paper. Figure 11 presents some comparative room temperature fatigue data on 6Al-4V titanium and the 4100-4300 low alloy steels in the unnotched and notched conditions. Unnotched data on 6Al-4V at 750 F are also included. The comparison shows that the titanium alloy is superior in fatigue to the low alloy steel in both the unnotched and notched conditions.

NICKEL BASE ALLOY

Fatigue data are presented on Inconel X sheet heat treated to 155,000 psi minimum tensile strength. It is recognized that newer nickel-base alloys, such as Inconel 718 and Rene' 41, are now available with improved elevated temperature properties. Figure 12 compares the elevated temperature tensile properties of Inconel X, Inconel 718, and Rene' 41. Fatigue data on the newer alloys was not readily available for this paper and the data on Inconel X are being used to indicate the trends to be expected in the fatigue characteristics of the high temperature nickel-base alloys. Figure 13 shows some room temperature axial tension fatigue properties of Inconel X in the unnotched and welded conditions. The welded specimens were fusion welded, using the tungsten, inert gas, shielded arc method, heat treated after welding, and tested with the full weld bead normal to the direction of loading. The notch effect of the weld has been estimated as being approximately equivalent to $K_t = 2.5$. Analysis of the data showed that the room temperature fatigue properties of Inconel X, in terms of percent of static tensile strength, are somewhat lower than the fatigue properties of the 4100 and 4300 low alloy steels; being more comparable to the 7075 - T6 aluminum alloy. Endurance limits for Inconel X were 34% of the unnotched tensile strength in the unnotched condition and 20% in the welded condition. The comparative fatigue properties of Inconel X, low alloy steel, and 7075 - T6 in the unnotched condition are given in Figure 14. Again, as with the data on H-11 and PH15-7Mo, some fatigue properties of Inconel X are shown at temperatures of 1000-1350 F which temperatures are likely to be commonly applicable to the use of this alloy in service.

SUMMARY

The data presented in this paper can only be considered as being preliminary. However, the trends and comparisons that can be noted indicate that the new materials being considered for advanced air vehicles will provide adequate service life. For purposes of illustration, the comparative fatigue strengths of the materials discussed in this paper are shown on Figure 15. The data have been arbitrarily selected for a fatigue life of 10^6 cycles in the unnotched and notched conditions. A life of 10^6 cycles is considered to approximate the endurance limit for most materials. Figure 16 shows the notched SN curves, plotted as a band, for the selected materials. These materials must be properly used in design in consideration of their known fatigue characteristics. The expected load spectra for the structure under design must be matched against the fatigue properties of the material in order to determine whether the service life requirements will be met. Figure 17 presents an estimated spectrum for an advanced fighter type aircraft, as compared to the envelope of notched fatigue curves (from Figure 16) for the materials discussed in this paper. Analysis of this comparison shows that a service life of 3000 hours for the advanced fighter type airframe can be expected.

The intent of this paper was not to present a complete story on the fatigue properties of the new materials discussed here, but rather to appraise their fatigue characteristics for advanced air

vehicle structure. World War II aircraft were largely fabricated from relatively low strength aluminum alloys such as 17s, 14s, and 24s, and from normalized or low heat treat level low alloy steels. Following World War II, the transition to sonic speed air vehicles, such as the F-86 and F-100, was accomplished with the use of the high strength aluminum alloy, 75S, and with relatively high heat treat level (160,000 - 220,000 psi) steels. The data presented in this paper indicate that the transition from a fatigue standpoint to advanced air vehicles, using new materials and types of construction, is expected to be no more difficult than the transition from World War II aircraft to the sonic speed air vehicle.

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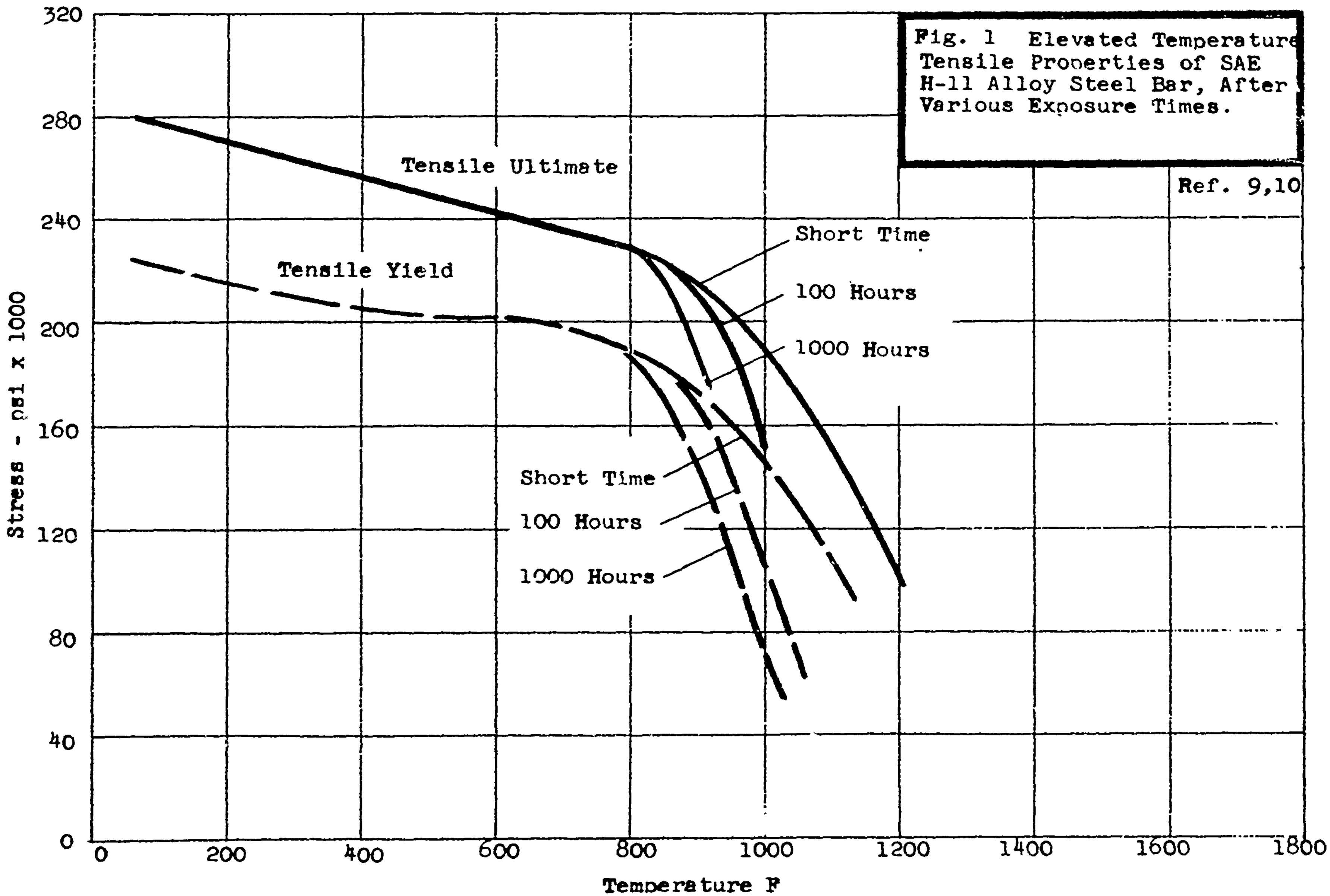

Fig. 1 Elevated Temperature Tensile Properties of SAE H-11 Alloy Steel Bar, After Various Exposure Times.

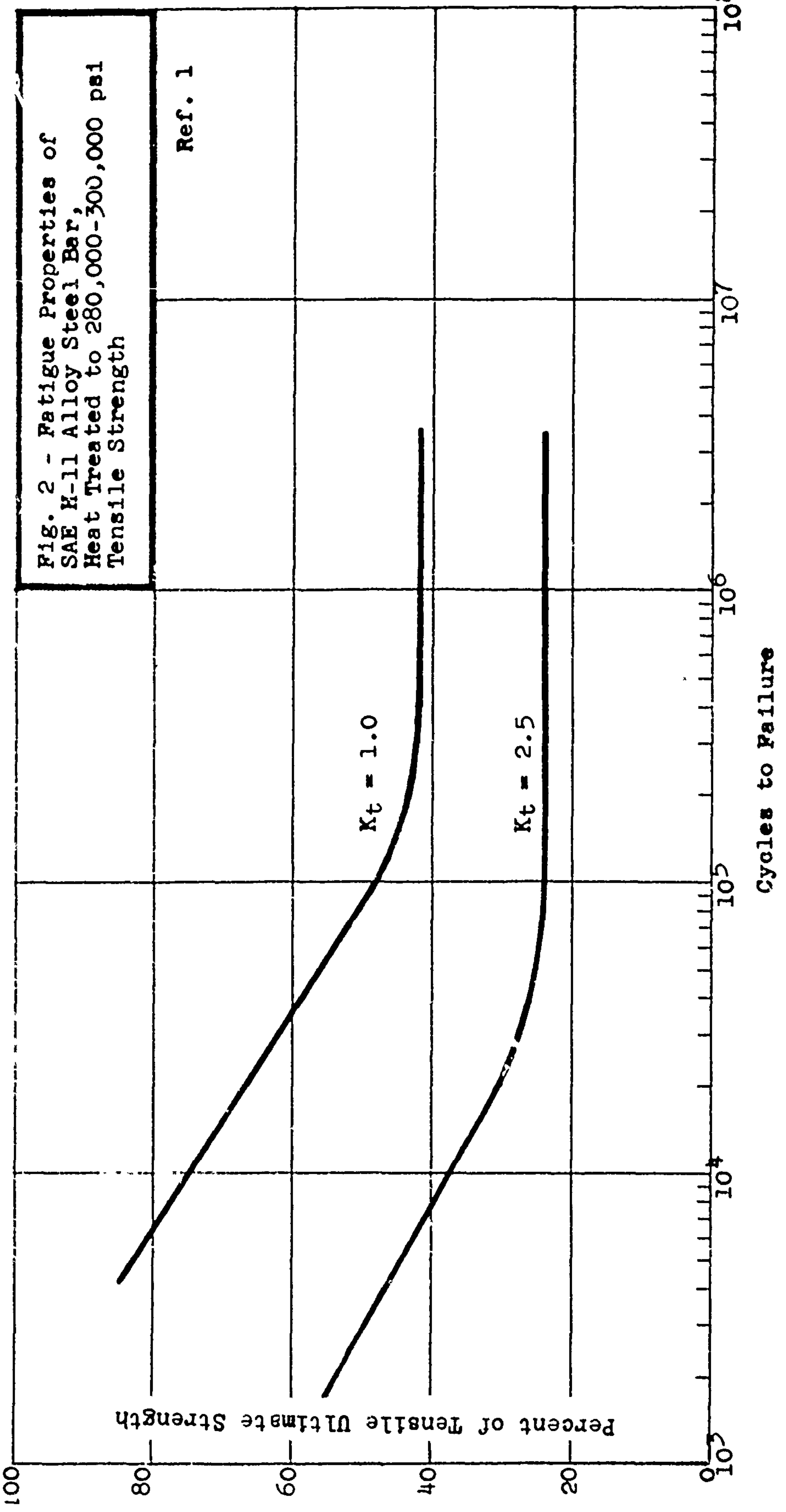

Fig. 2 - Fatigue Properties of SAE H-11 Alloy Steel Bar, Heat Treated to 280,000-300,000 psi Tensile Strength

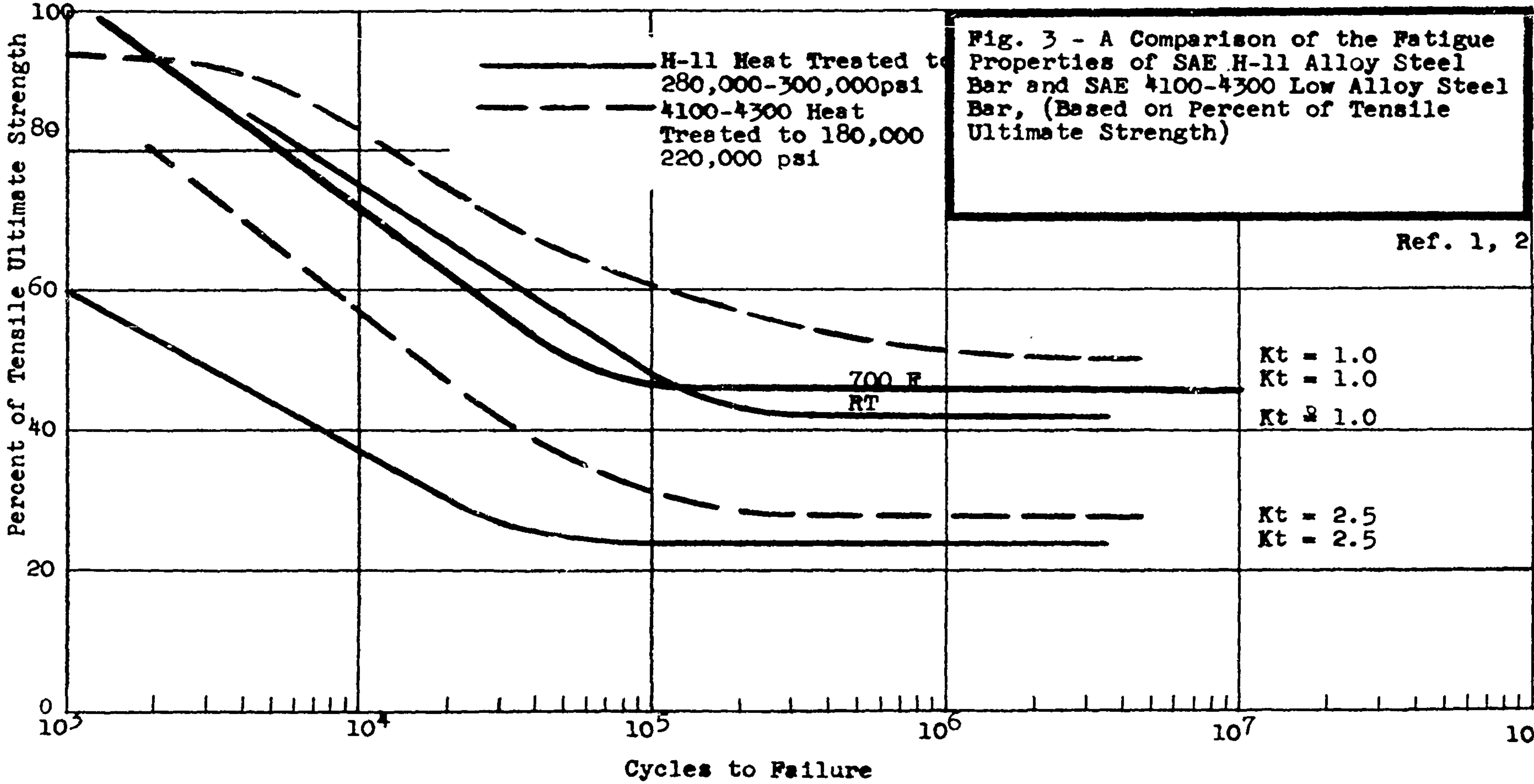

Fig. 3 - A Comparison of the Fatigue Properties of SAE H-11 Alloy Steel Bar and SAE 4100-4300 Low Alloy Steel Bar, (Based on Percent of Tensile Ultimate Strength)

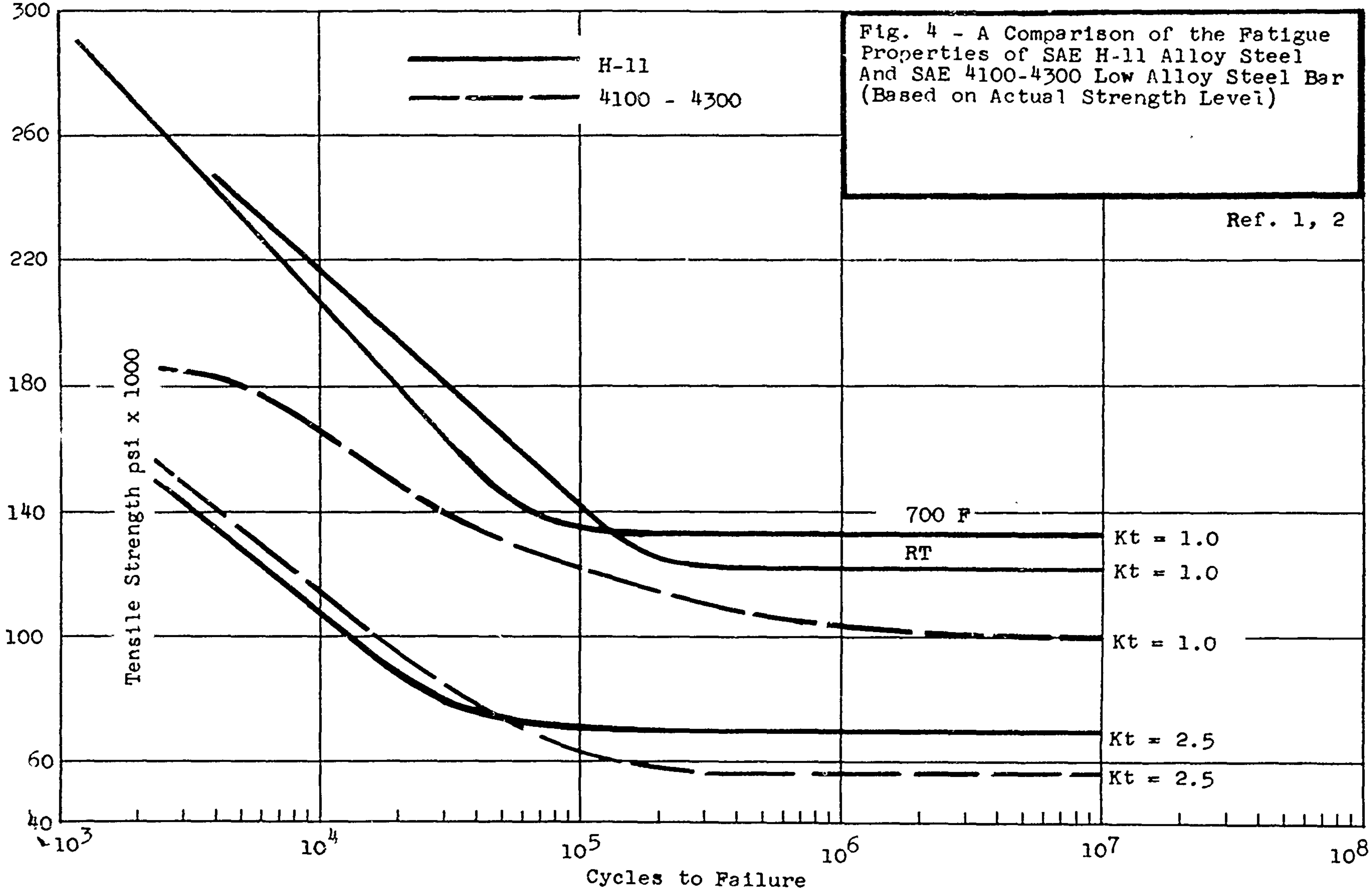

Fig. 4 - A Comparison of the Fatigue Properties of SAE H-11 Alloy Steel And SAE 4100-4300 Low Alloy Steel Bar (Based on Actual Strength Level)

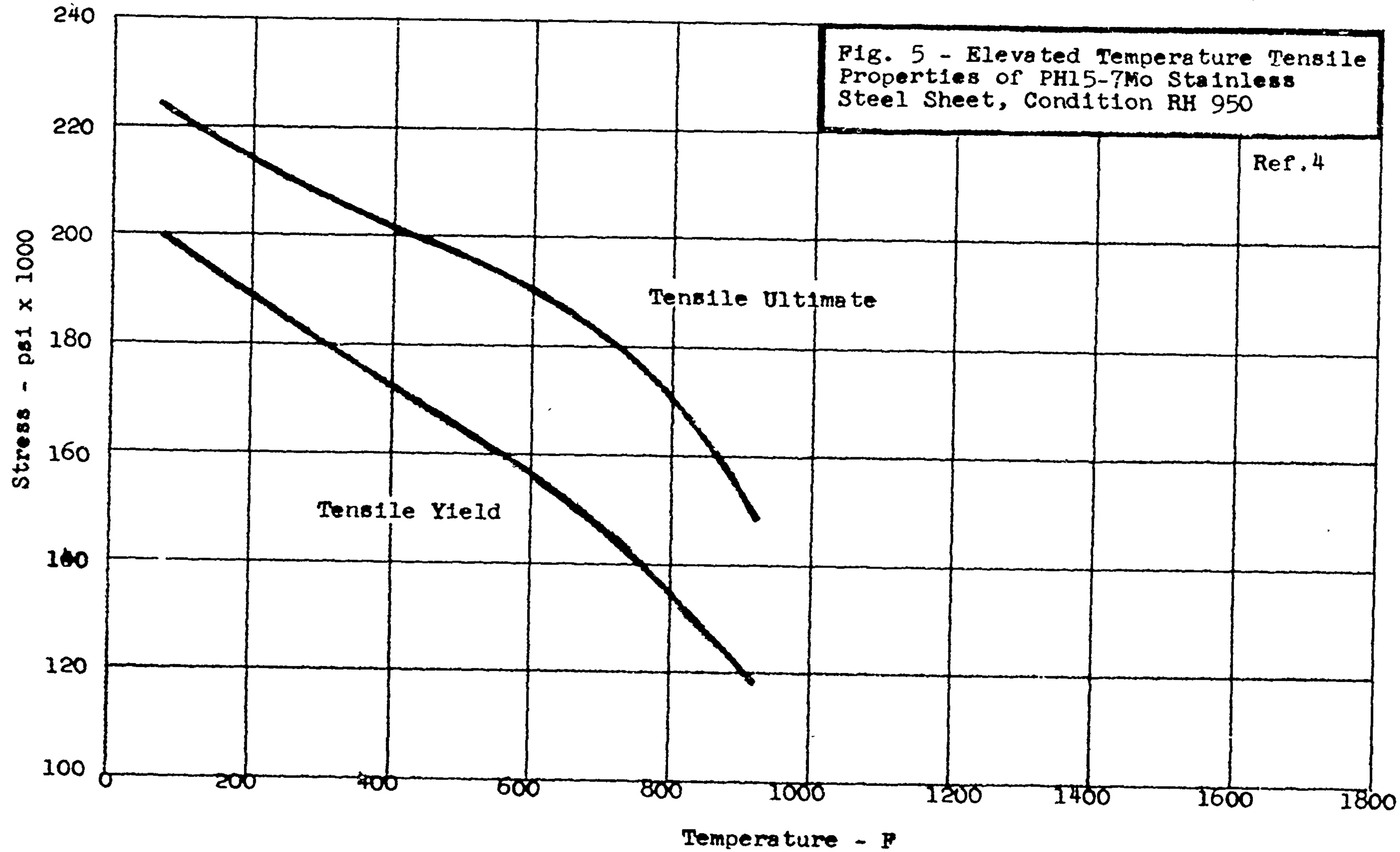

Fig. 5 - Elevated Temperature Tensile Properties of PH15-7Mo Stainless Steel Sheet, Condition RH 950

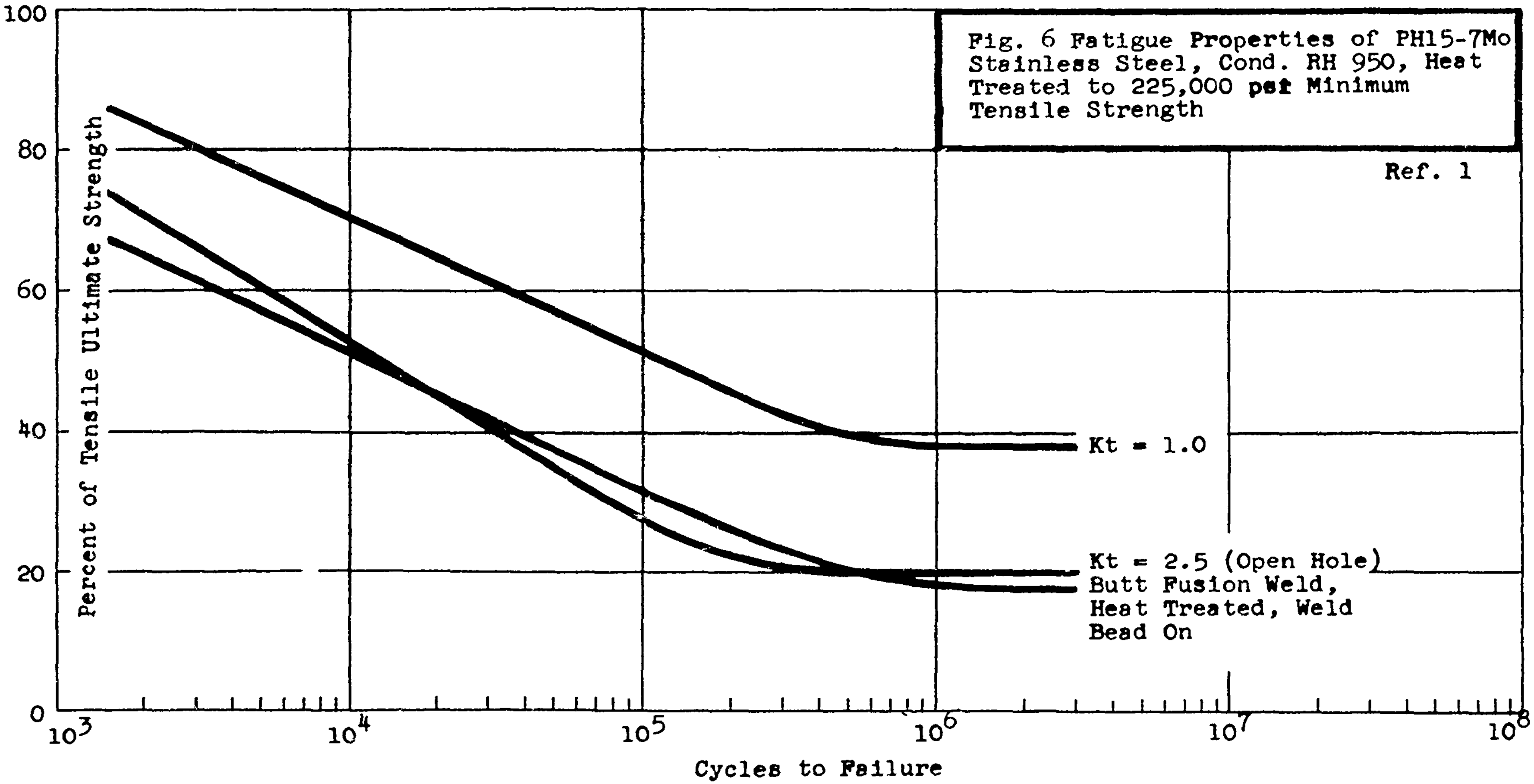

Fig. 6 Fatigue Properties of PH15-7Mo Stainless Steel, Cond. RH 950, Heat Treated to 225,000 psi Minimum Tensile Strength

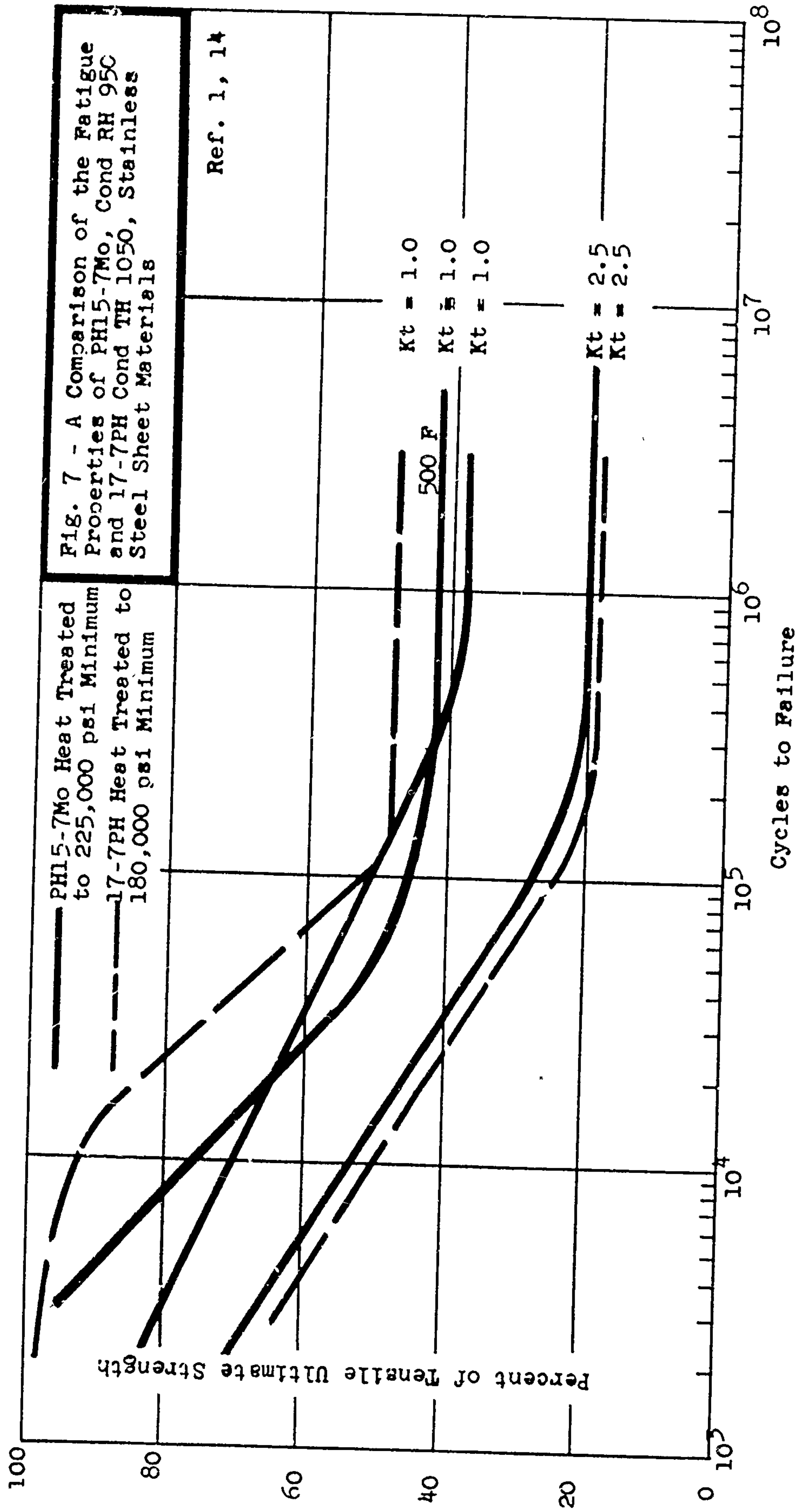

Fig. 7 - A Comparison of the Fatigue Properties of PH15-7Mo, Cond RH 950 and 17-7PH Cond TH 1050, Stainless Steel Sheet Materials

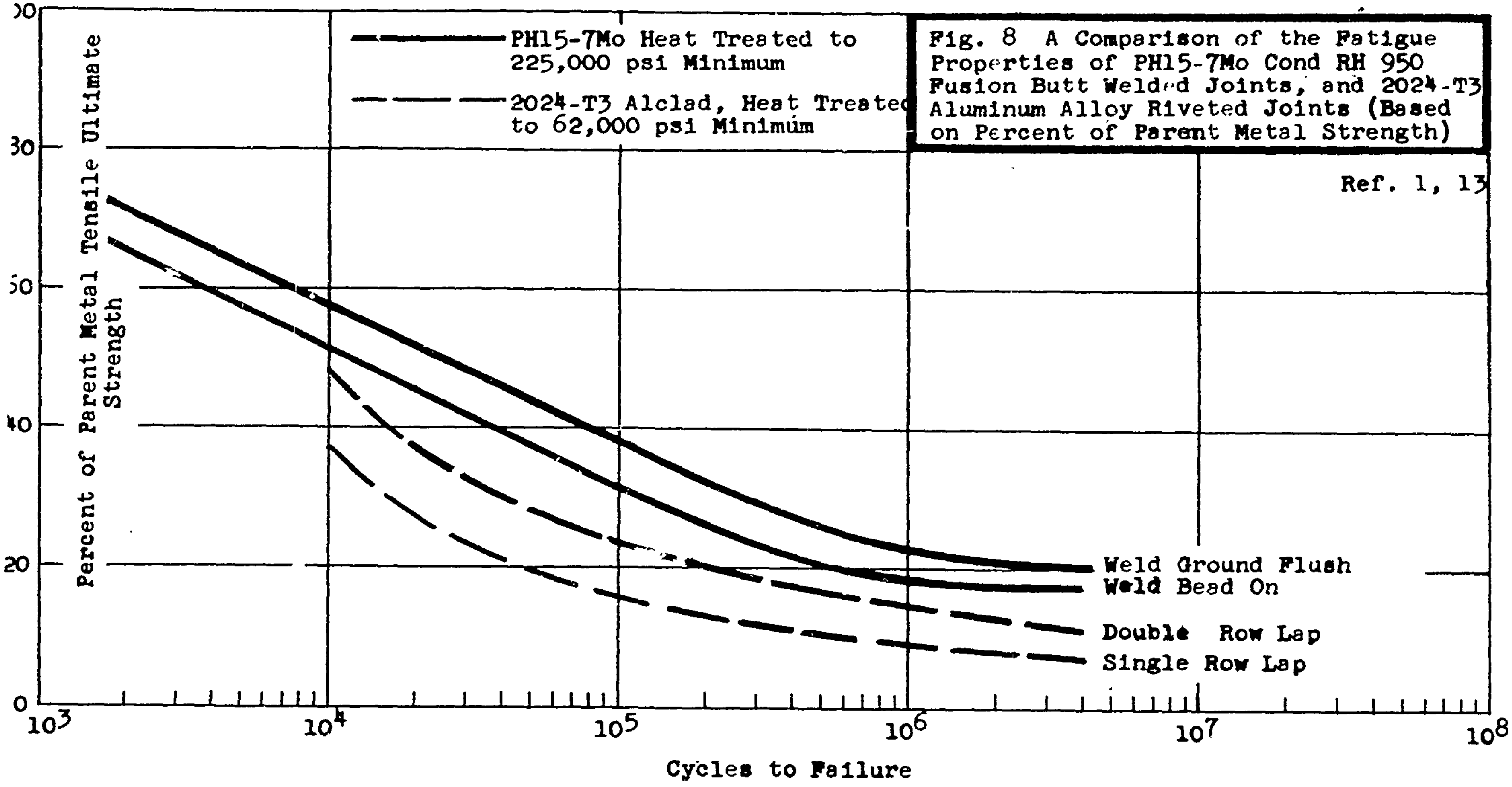

Fig. 8 A Comparison of the Fatigue Properties of PH15-7Mo Cond RH 950 Fusion Butt Welded Joints, and 2024-T3 Aluminum Alloy Riveted Joints (Based on Percent of Parent Metal Strength)

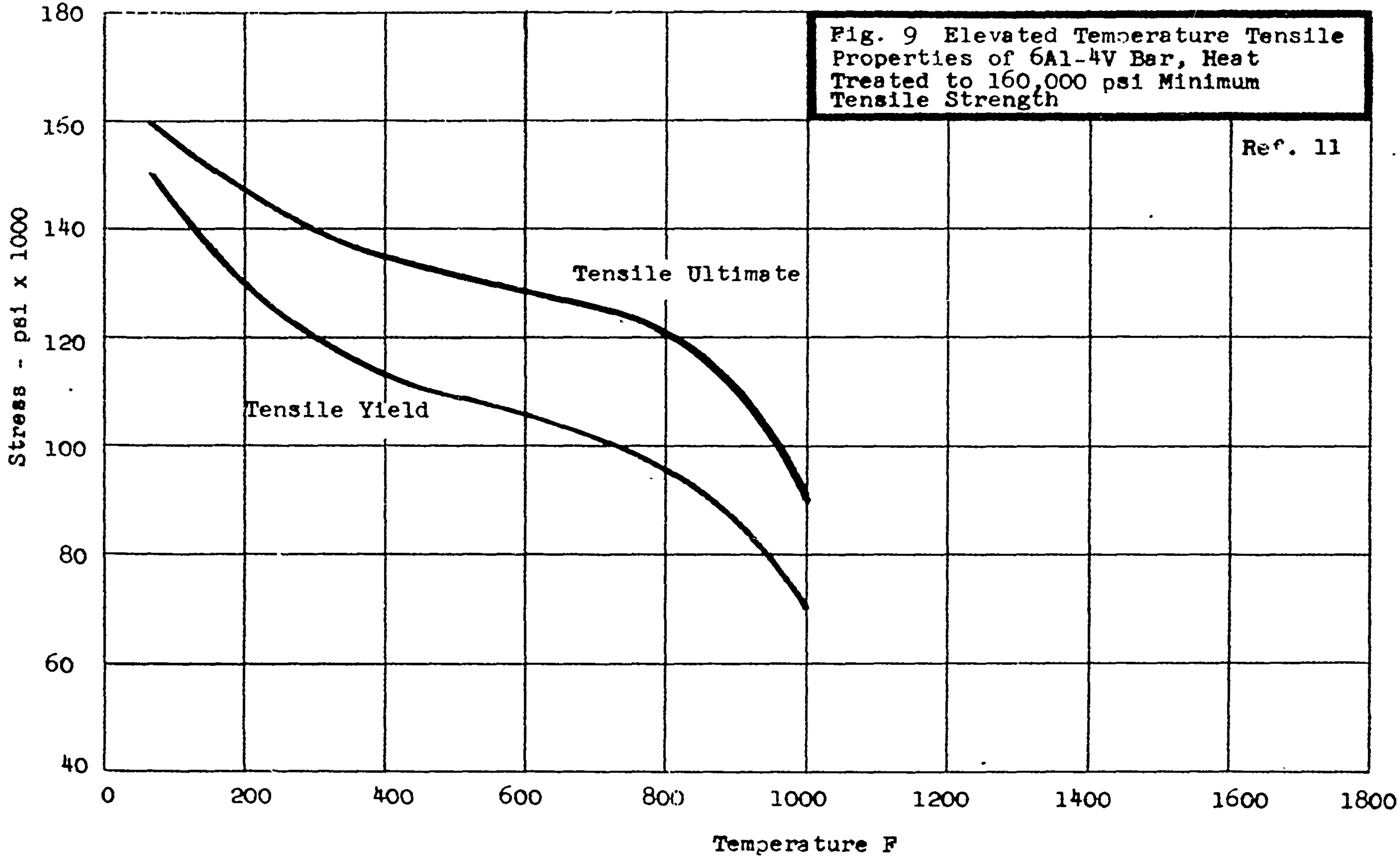

Fig. 9 Elevated Temperature Tensile Properties of 6Al-4V Bar, Heat Treated to 160,000 psi Minimum Tensile Strength

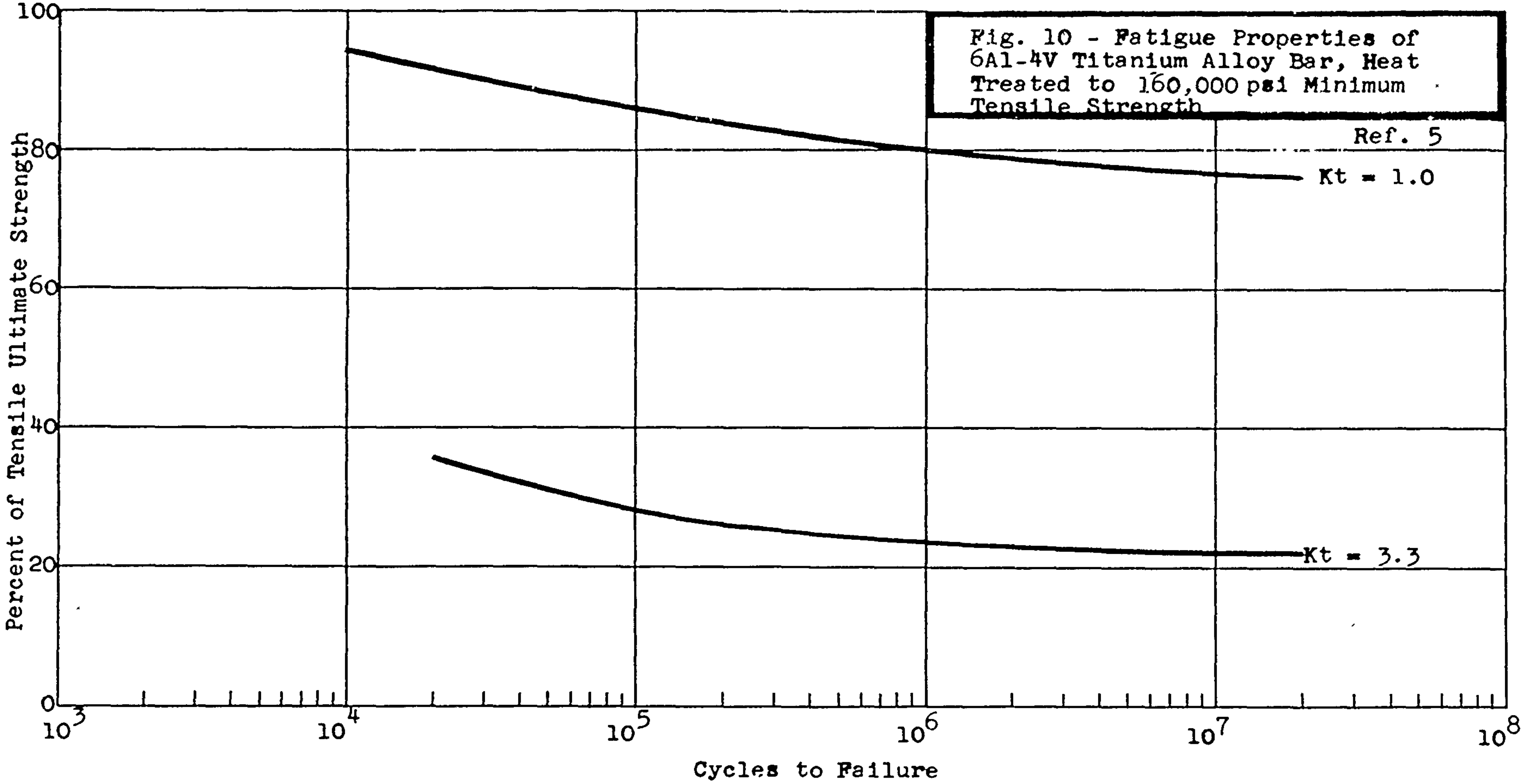

Fig. 10 - Fatigue Properties of 6Al-4V Titanium Alloy Bar, Heat Treated to 160,000 psi Minimum Tensile Strength

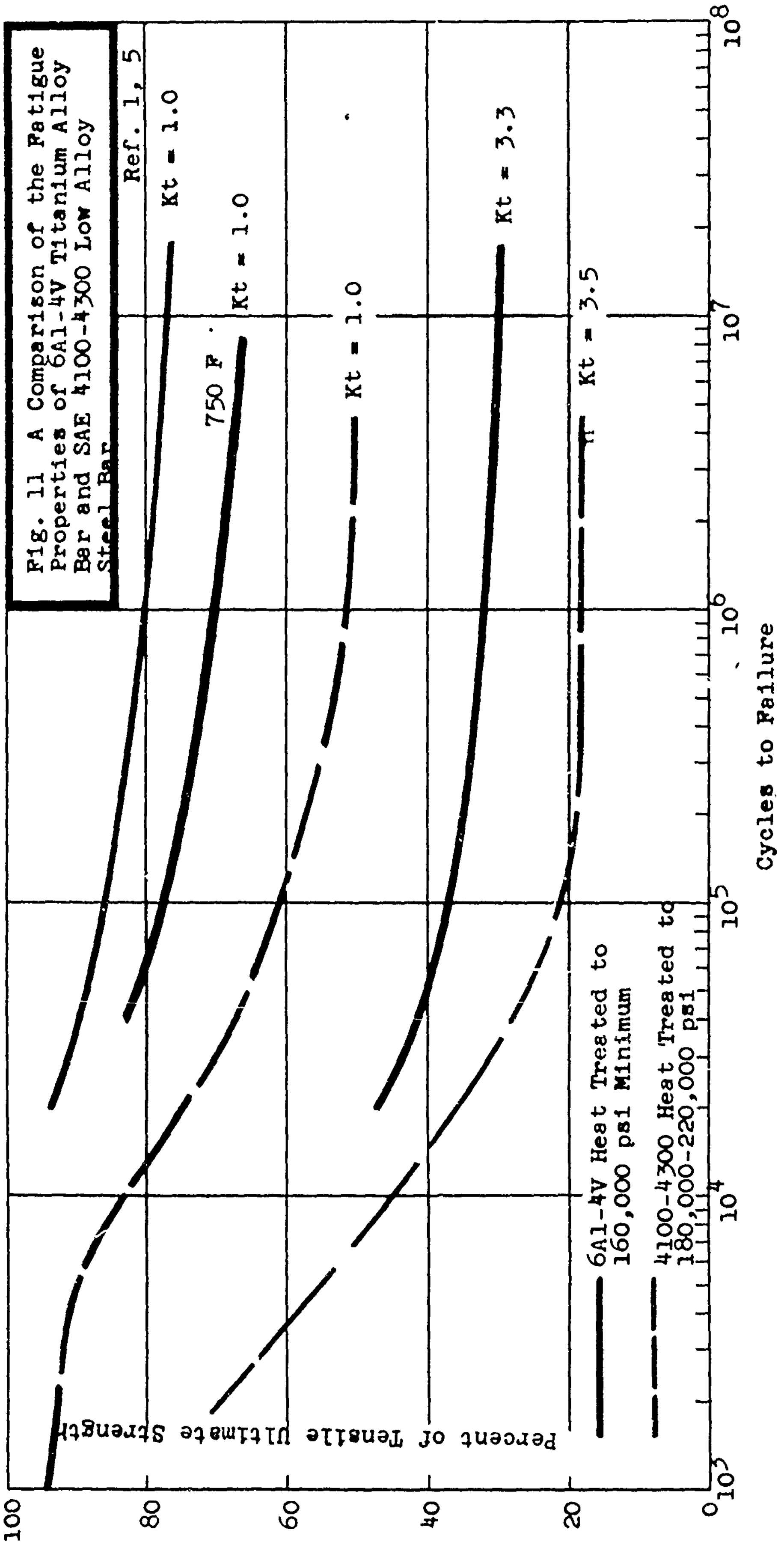

Fig. 11 A Comparison of the Fatigue Properties of 6Al-4V Titanium Alloy Bar and SAE 4100-4300 Low Alloy Steel Bar

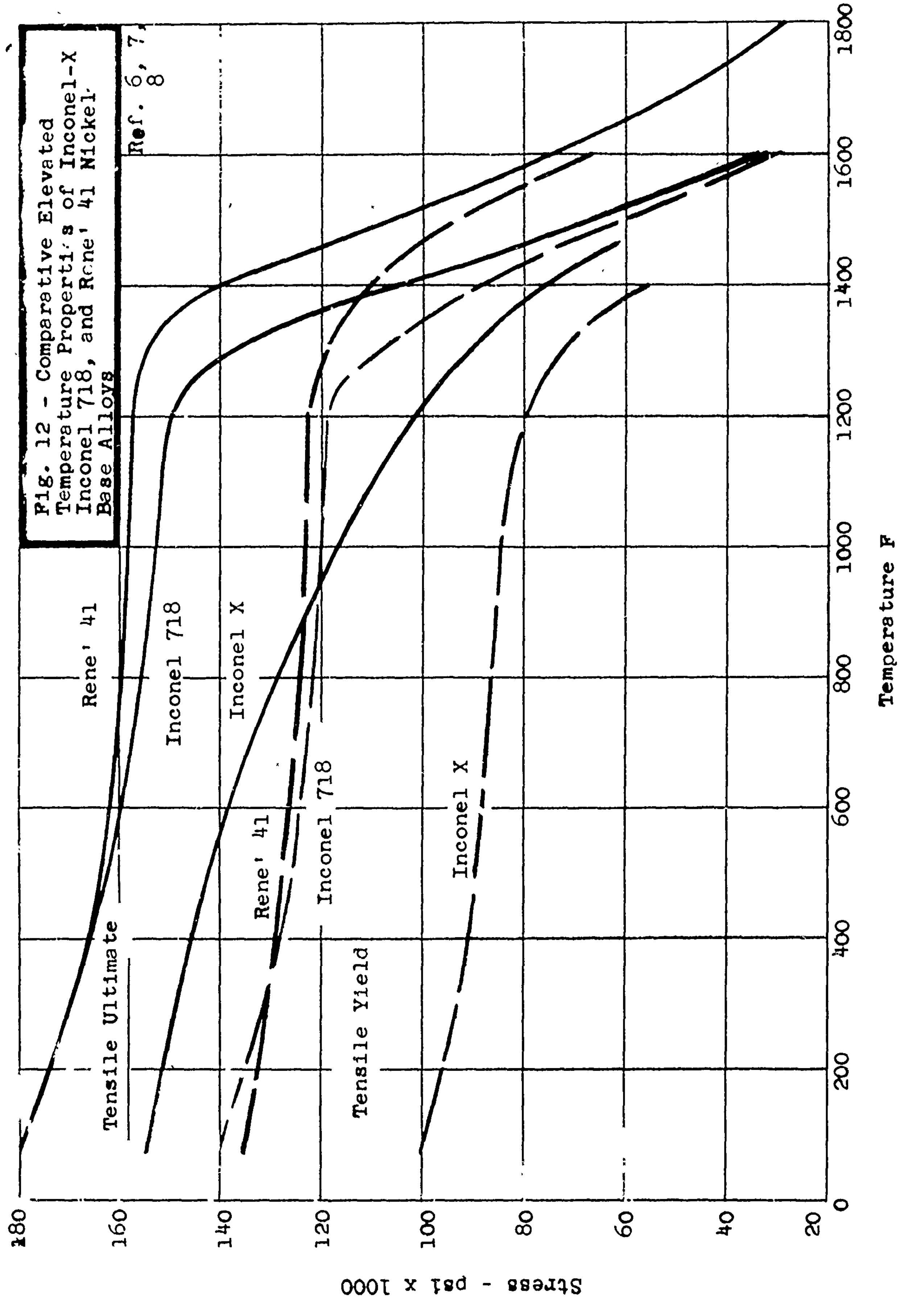

Fig. 12 - Comparative Elevated Temperature Properties of Inconel-X Inconel 718, and Rene' 41 Nickel Base Alloys

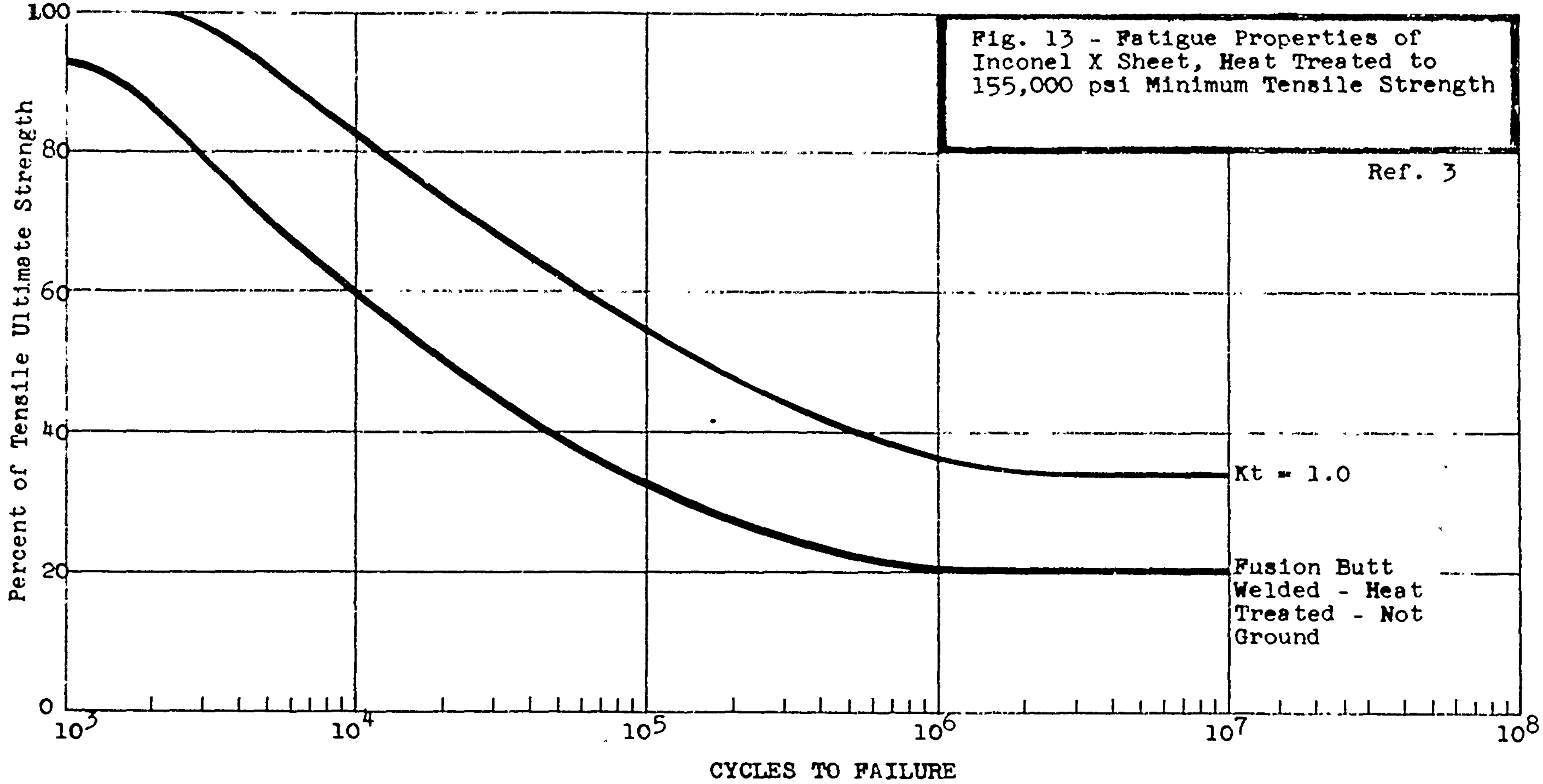

Fig. 13 - Fatigue Properties of Inconel X Sheet, Heat Treated to 155,000 psi Minimum Tensile Strength

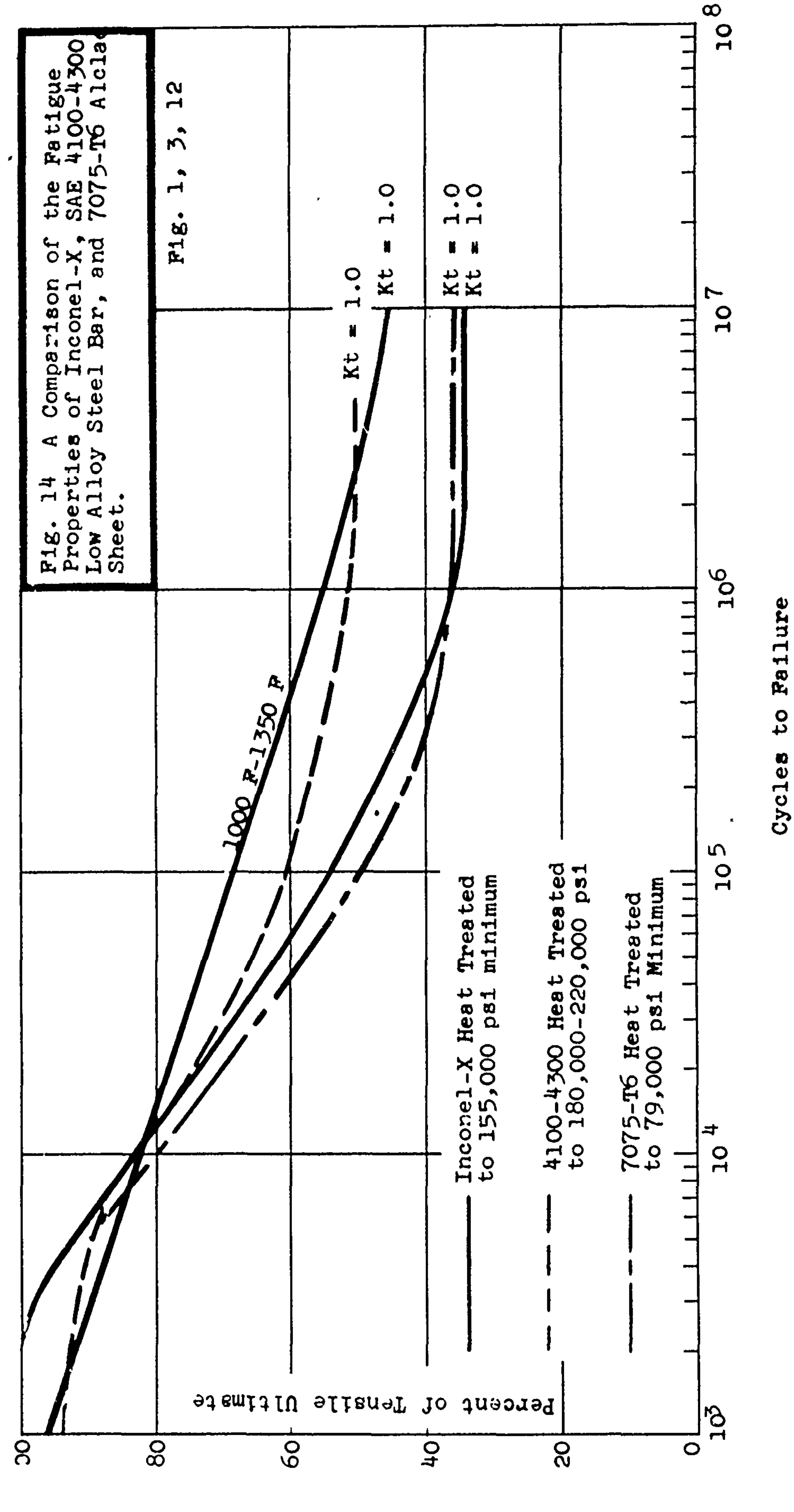

Fig. 14 A Comparison of the Fatigue Properties of Inconel-X, SAE 4100-4300 Low Alloy Steel Bar, and 7075-T6 Alclad Sheet.

FIGURE 15

FATIGUE STRENGTHS OF SELECTED ALLOYS
AT 10^6 CYCLES

Material	Form	Temp.	Fatigue Strength (% Tensile Ultimate)	
			Kt = 1	Kt = 2.5 (a)
7075-T6	Alclad Sheet	RT	36	22
H-11 (280,000-300,000 psi)	Bar	RT 700 F	42 46	24 -
4100-4300 (180,000-220,000 psi)	Bar	RT	51	28
PH15-7Mo (225,000 psi min.)	Sheet	RT 500 F	38 42	20 22
17-7 PH (180,000 psi min.)	Sheet	RT	48	19
6 Al-4V (160,000 psi min.)	Bar	RT	76	24 (Kt = 3.3)
Inconel-X (175,000 psi min.)	Sheet	RT 1000-1350 F	36 55	- -

(a) Except as noted

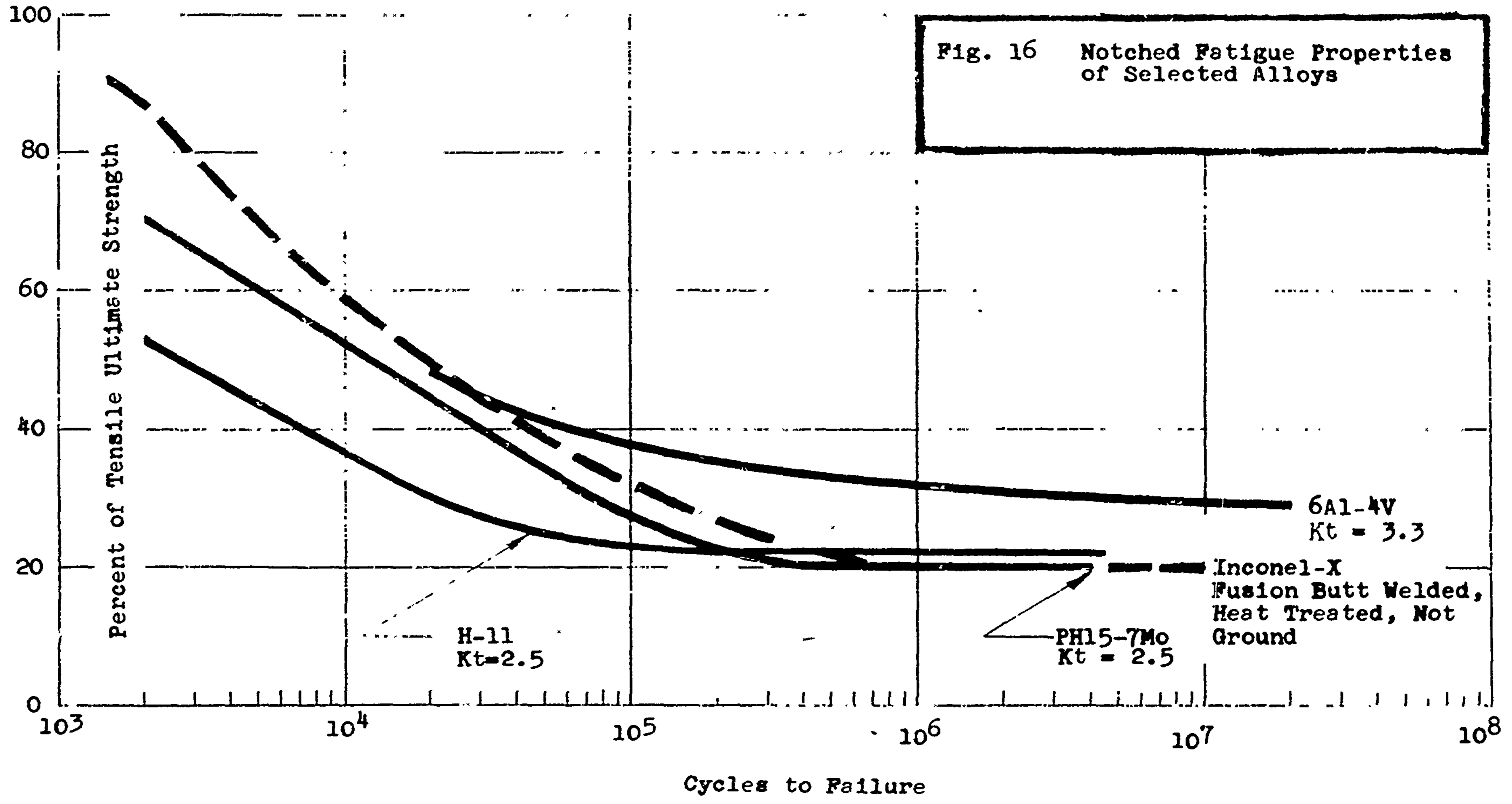
Fig. 16 Notched Fatigue Properties of Selected Alloys
100
80
60
40
20
0
Percent of Tensile Ultimate Strength
6Al-4V
Kt = 3.3
Inconel-X
Fusion Butt Welded,
Heat Treated, Not
Ground
H-11
Kt=2.5
PH15-7Mo
Kt = 2.5
10^3
10^4
10^5
10^6
10^7
10^8
Cycles to Failure

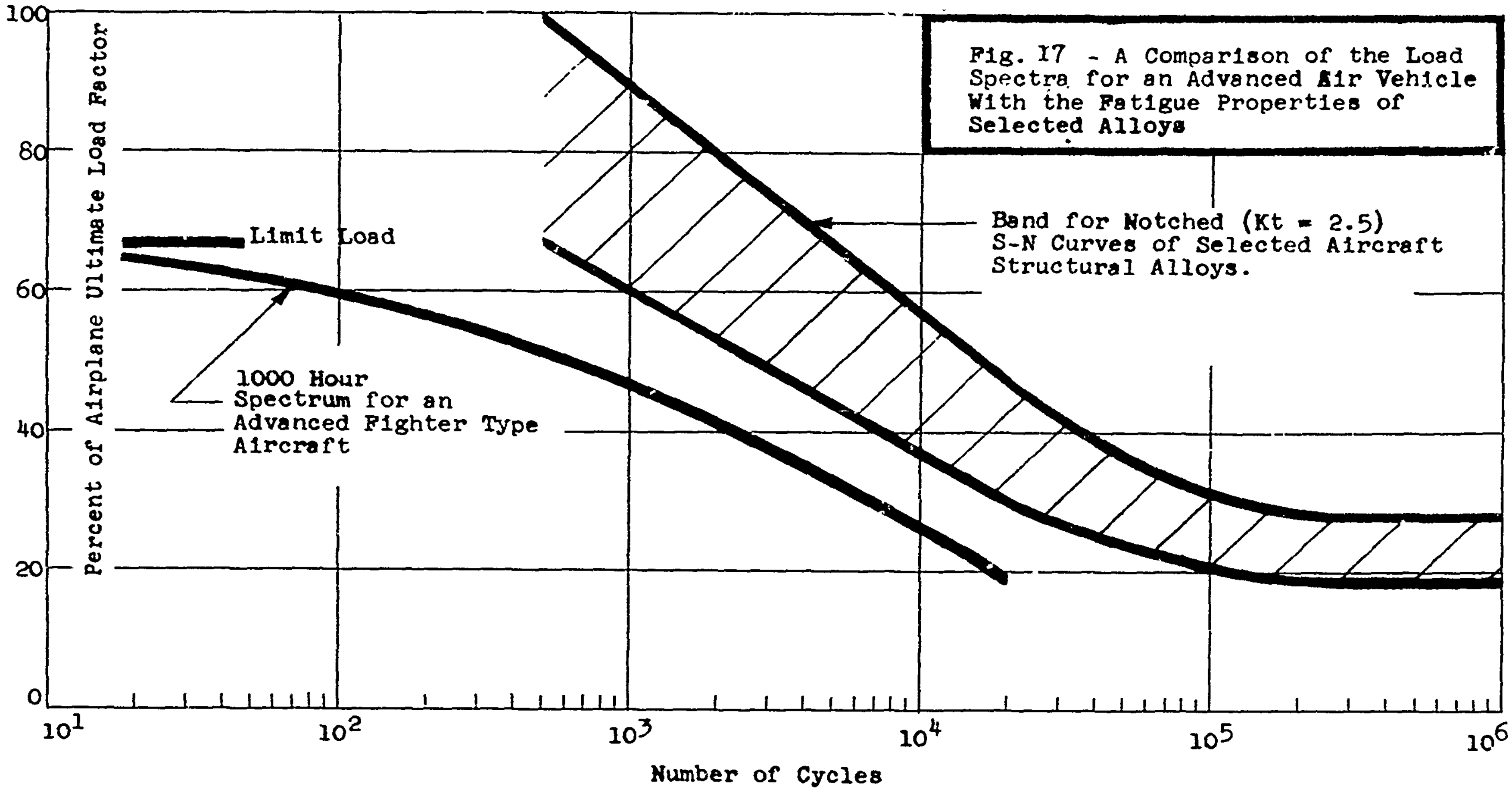

Fig. 17 - A Comparison of the Load Spectra for an Advanced Air Vehicle With the Fatigue Properties of Selected Alloys

IMPROMPTU DISCUSSIONS - SESSION II-B

Editorial Note: Attention is directed to the editorial policies presented in the Preface which were followed in editing the impromptu questions and answers of the session.

QUESTIONS AND ANSWERS FOLLOWING PROFESSOR DOLAN'S SPEECH:

CHAIRMAN, DR. FOUNTAIN:

I think we have time for one or perhaps two quick questions from the floor, if anyone has something.

MR. KRAKOW, OHIO STATE:

I wanted to ask if the Minor hypothesis was applied to this specimen and, if so, what were the results?

PROFESSOR CORTEN:

You are inquiring about the sample right at the end? The Minor hypothesis could be applied simply by inserting an exponent 4.8 instead of 4, and gives a life of 76 days instead of 59 or so estimated.

VOICE, WRIGHT AIR DEVELOPMENT CENTER:

I'd like to ask if the equation which takes into account low stress contributions to fracture, also takes into account the time in which the occurrence of high stresses is in the sequence or will a low stress contribute to fracture prior to the higher stress being applied?

PROFESSOR CORTEN:

Analysis assumes that the high stress occurs early in the history, early enough that the fatigue damage is prior to the peak stress.

QUESTIONS AND ANSWERS FOLLOWING MR. KATTUS' PAPER:

CHAIRMAN:

We have quite a bit of time for discussion, but before we start on it, I would like to re-emphasize Mr. Kattus' conclusions, particularly with respect to the refractory metals. It is becoming more and more important to us today. We all look back on steel as the oldtimer; we know a lot about it, but Mr. Kattus has demonstrated that there are large gaps in our knowledge. I think we should certainly let this be a lesson to us in our work in the refractory metals area, where the problems are many-fold more difficult than those with steel to start with, and not to be prosaic in your approach to these things and concentrate on particular areas at the loss of building up our knowledge in some other aspect of the whole metals technology.

MR. COOK, ALLEGHENY:

I would like to ask Mr. Kattus what was the steel material?

MR. KATTUS:

That was low carbon hot rolled sheet steel.

MR. COOK:

One other thing, was there some reason assigned for the wide variation in the percentages 8% against 30% bracket, for normal topping percents?

MR. KATTUS:

Well, that was what they wanted to do. They wanted some on the low side. I don't know that 30% is particularly high for alloy steel practice. Actually, 30% is their standard practice to obtain sheet.

MR. COOK:

The other thing, I was wondering, have you investigated the effect of temperature variations from the head end to the tail end of the coil, giving that offset of your probability curve shown in Figure 2? That looked to me to be from temperature variations while you're rolling.

MR. KATTUS:

That may be. We are hoping to offer a lot more. We have had an initial look at these things and are hoping to get a lot more answers.

MR. COOK:

The question, I think, has been assigned to other materials, other alloys of steel, and it has been investigated. I think you can find some literature on that.

MR. SIERADZKI, ROHR AIRCRAFT:

With the automation of the process of forging materials, or machining materials and welding, don't you think that we will get more uniform results from automatic welding type programs and machining than we do by the manual methods?

MR. KATTUS:

Yes, I definitely do.

MR. CAPPELL, BELL AIRCRAFT:

Could you tell me the difference in your welding procedure between the samples 2 and 6 of the last slide? The legend seemed to read exactly the same.

MR. KATTUS:

One of them, as I recall, was welded and ground flat with the surface.

MR. FRANKEN, BUREAU OF STANDARDS:

How can you be sure that the variations in the results that you obtained on the various fatigue specimens are a result of the manufacturing process or the various machining procedures that you followed in producing a fatigue specimen? Can you differentiate between the variations?

MR. KATTUS:

The specimens, themselves, were as nearly alike as possible, but the production processes of cropping were deliberately varied over wide limits. I think you would naturally do it that way if you wanted to study the effect of cropping, the different methods of cropping.

MR. COOK:

Have you done anything in connection with your cropping experiments to sonically determine the necessity of cropping?

MR. KATTUS:

No. I am sorry that the particular study, the one I used, hasn't gone along very far. I just had one or two illustrations of what we had found.

GEORGE HILL, ALCOA:

Have you reason to believe that differences in tensile strength are likewise recognized in differences in fatigue strength? Your photograph showed evidence of differences. It reflected the tensile strength but reflected no fatigue strength.

MR. KATTUS:

That is perfectly possible, but it seems to me that the variations in structure and in composition usually have a larger effect on fatigue than they do on tensile, but we don't have any fatigue data on these materials. It was just to illustrate the fact that at least this particular company doesn't have the right ideas as to how to produce uniform properties and structures in their sheet.

CHAIRMAN:

I wasn't quite clear about the tensile samples taken from a sheet as rolled, as to whether the sheet had been coiled and annealed, or just exactly what the heat treating procedure was. This comes back to your idea of the finishing temperature in the top or bottom, if they are coiled. There is considerable difference in a normal box annealed inside or outside of a large coil.

MR. KATTUS:

They were annealed. Actually, the samples of coil as we took them were intermediate stock of production, eventually cold rolled, continuously annealed, and we analyzed and took the samples at the coiling station and annealed them at the lab.

QUESTION AND ANSWER FOLLOWING MR. SCHUETTE'S PAPER:

CHAIRMAN:

We will take a couple of minutes for quick questions.

MR. HORN, WRIGHT AIR DEVELOPMENT CENTER:

I'd like to ask Mr. Schuette if he has run comparative tests on magnesium to magnesium joints in fatigue with magnesium to other materials in fatigue, and the progress of the practice.

MR. SCHUETTE:

I believe this is it, whether we have or have not run tests on magnesium to magnesium joints and magnesium to aluminum joints, and I think the essence is in the part I can't answer. Yes, we have followed the progress of the cracks occurring. We have run such tests of mixed and like joints, primarily for the purpose of evaluating various insulating materials to put in dissimilar joints, but in that case, since our direction was expressly different, we were not concerning ourselves with cracks. As a matter of fact, we have done very little in the way of crack propagation studies and are not too well satisfied with those we did.

QUESTIONS AND ANSWERS FOLLOWING MR. STULEN'S PAPER:

VOICE, AEROJET:

Did you notice any significant differences between the air melted and vacuum melted steel as far as fatigue life is concerned such as the vacuum melted supposedly would reduce the number of inclusions?

MR. STULEN:

We tested vacuum melted steel and found there is a sizeable improvement in the smoothness of the specimens but there is practically no improvement of the notched specimen. We used rather sharp notches and it substantiates what Mr. Schuette said a few minutes ago.

MR. SIERADZKI, ROHR AIRCRAFT:

You said the rate of propagation with increasing cycles was uniform. Wouldn't the vertical scale on your graph plotted on log scale be accelerating proportionately to the propagation?

MR. STULEN:

I think my terminology may have been incorrect. I meant the rate of crack propagation became more uniform rather than becoming constant. I used the wrong terminology.

MR. CHAPON, ALLIED RESEARCH:

In your figure 4, you mentioned the speed effect. I wondered if you could expand on that a little bit.

MR. STULEN:

I think the speed effect may be somewhat simply due to the delay in yield time and the rapid tensile testing. In other words, yield point increases with speed. I think this may explain the speed effect. In other words, if you go down to very low speeds the number of cycles to failure is reduced.

QUESTIONS AND ANSWERS FOLLOWING MR. DOWNEY'S PAPER:

CHAIRMAN:

Being associated in the Union Carbide Metals Company, I am very interested in making columbium. I think Mr. Downey's remarks are well taken. I think we do have many problems. I agree with him and think we should pitch in together. I also would like to add to his remarks, let's get together - not merely materials engineer and design engineer - but the materials producer, the raw materials people back on the front line. We have to go right back to the beginning in many of these problems and work them together. We have time for a couple of questions.

MR. CARTER, MARTIN COMPANY:

In your opinion, is there a possibility five years from now for making coatings less brittle?

MR. DOWNEY:

That is a good question. It is going to be hard to give you a short answer. I think the coatings are going to get better. Most of the coatings we know today are ceramics and because of their very nature do tend to be brittle. We are, in our own plant, working on a program now to see if we can't get some metallic coatings. We tried it before without success but we are going to give it another go and see if we can get metallics to work at these high temperatures. If we could, a lot of the problems mentioned would disappear. And I want to add again that this picture is not completely black. If I could hark back to some experiences with moly turbine buckets, we were able to operate at very high temperatures in actual engines for periods upwards of 25 to 50 hours without failure of the coatings. Initially, we were getting impact, particle erosion, hot gas erosion, fatigue and the whole works. Unfortunately that happens to be a little more simple structure than some we will be getting into before too long. I am not actually being evasive; I just don't know, but I'm probably more optimistic than I sound.

VOICE:

One remark about coatings. I am not quite as optimistic, for the simple fact that we have, in our organization, spent the last year and a half or two years working on coatings for columbium. We feel we have a coating that actually is a duplex coating which we think is outstanding. I am sure that we would be more than happy to talk to any of you people about it and work on some mutual interest basis.

MR. DOWNEY:

I have a counter comment. My counter comment simply is to the effect that we, too, have coatings which we think are really good for molybdenum and reasonably good for columbium. Now, the whole thing depends on the application. I am also willing to talk to anybody. My business is not selling coatings, but I'd really be interested to try.

MR. ANDERSON, DU PONT:

Actually, I don't think we can look forward to any great breakthroughs which are going to solve this problem by improving particularly columbium. By improving the basic material we postpone the onset of the embrittlement to longer and longer times and as we perfect better coatings we arrive eventually just as in this fatigue problem, at certain life expectancy and we have to go ahead and design our structures for it. We have to accept a limited life, that's all.

MR. DOWNEY:

I also say amen to that.

CHAIRMAN:

I just didn't want this discussion to end with a pessimistic note. Frankly, I think most of us, even though overly cautious in describing some of the refractory metals, are optimistic.

REMARKS FOLLOWING MR. FAIRBAIRN'S PAPER:

CHAIRMAN:

Thank you very much, Mr. Fairbairn, for an interesting materials comparison. Because the forum and your chance to really quiz the authors is following immediately, I think we will delay any questions you may have directly concerning this paper. I would like to take this opportunity to thank the authors for their participation in the symposium and for their excellent presentations and for keeping very nicely in their time schedules. I'd like to thank everyone in the audience for your discussions and I am sure the discussions which will follow. Last, but by no means least, I know I can speak for the entire audience in expressing our appreciation at being here and to thank the symposium people and WADC for the very fine job they have done in getting these things together and organizing and arranging things here. It has been a very excellent meeting and we have derived a great deal of benefit from it.

Session Chairman:
Dr. Richard W. Fountain, Union Carbide Metals Company

Panel Members:
Professors T. J. Dolan & H. T. Corten, Univ. of Illinois
Mr. J. R. Kattus, Southern Research Institute
Mr. Evan H. Schuette, Dow Chemical Company
Mr. F. B. Stulen, Curtis Wright Corporation
Mr. R. C. Downey, General Electric Corporation
Mr. G. A. Fairbairn, North American Aviation

Editorial Note: Attention is directed to the editorial policies presented in the Preface which were followed in editing the discussions of the Forum.

CHAIRMAN, DR. FOUNTAIN:

We have several piles of questions here and because it is difficult to arrange them in any order, I think we will just hand them to each of the authors and go down the table, starting at my left, and have each man read the question on one card, answer it, and proceed to the next man. If the author of the question desires more clarification than he receives in the original answer, please hold your hand up in the air and make yourself heard. If that arrangement is satisfactory with you gentlemen at the table, we will try to proceed in that way.

MR. FAIRBAIRN:

Question: "From the standpoint of fatigue, don't some of the high temperature materials lose advantage because of poor machinability, which results in poorer surface finish?" This is from Mr. Sieradzki, Rohr Aircraft. Certainly, the answer to this would have to be true. Although, we are learning many things about machining these new materials and some of them do have poor machinability, we are now able to machine satisfactorily some of the nickle base alloys and cobalt base alloys. They are admittedly very difficult to get good surface finish on. Does that answer the question?

MR. SIERADZKI:

You are about in the same boat as we are. We are making some progress.

MR. FAIRBAIRN:

I am really encouraged by some of the results we have had on machining these new alloys. We are getting much better results than I anticipated.

MR. STULEN:

Question: "Was the improvement in smooth specimen fatigue results observed

on special silicate-inclusion-free steel accompanied by a corresponding improvement in notched fatigue results?" This is from Mr. T. C. Delker, Bendix Aviation Corporation. Actually to my knowledge, I don't believe any notched specimens were run on the inclusion-free steel. Jack Sheehan who ran the test may be in the audience and he could explain this, but in the vacuum melt steel we found no improvement in a very sharp notch. Fatigue strength of a sharp notch of a vacuum melt notch was the same as regular commercial aircraft quality steel. Does that answer the question?

PROFESSOR DOLAN:

Mr. M. Licciardello of Solar Aircraft Company has two questions. The first question: "Why does the order of loading affect results of your theory? Specifically, why must you assume that high stresses occur before low stresses?" Actually, the order of loading affects the results in the theory by the factor of the number of damaged nuclei initiated by the particular stress level involved. We do not necessarily assume that high stresses occurred before low stresses. What we do say, in fairly random complex stress history, that the major damage will be done after you have once hit a peak stress which generates these nuclei and which will be concentrated at increased rates by all subsequent levels of lower stress.

The second question is: "Has there been any experimental evidence to indicate that there is a relationship between number of damaged areas and number of cycles occurring at a given value of stress?" I would say that the experimental evidence which we have is that from the many hundreds of small samples which confirm this type of relationship in that at high stresses we are getting more nuclei which can be observed in these slip bands and microscopic observations, and that the rates of propagation of damage as measured by life confirm this type of observation.

In connection with the same paper, Mr. L. Rastrelli of the Southwest Research Institute asks: "Is your hypothesis limited to that realm wherein the state of stress is elastic? If so, is there any provision to extend the theory to embrace the so-called plastic fatigue?" The theory has been developed principally in terms of experiments run in the long cycle range. We have not checked it by the low cycle random structural test in the so-called plastic fatigue range. There has been some work done, more general work, by Coppen, a few years ago which tended to confirm the general trends that we have found. Since then additional work done by Syracuse indicates it may not be as simple as all that, because it depends on the amount of plastic deformation introduced. If we base it on the peak stress, modified structure for the simple specimen, I think the relationship will be very similar, however.

PROFESSOR CORTEN:

May I make one comment there? Elastic behavior, I remind you, there is no such thing, if you get a fatigue failure the minute you get slip. There is inelastic deformation in plastic form in an elasticized scale.

A question from Mr. C. L. Hall, Wright Air Development Center, says: "It appeared to me that your second slide showed microsopic cracks in a material such as aluminum which has no discernable yield point. Does this damage at low stresses apply

alike to materials which have a yield only by an estimate of offset on stress-strain curves?" We have tested two steels, two aluminum alloys and got extensive data on these. There was no discernible difference between the two materials so far as behavior is concerned to repeated loading. If you look at the microscopic evidence of damage, there is a difference in the slip systems in steels compared to aluminum. On an entirely different basis, and as far as microscopic description of fatigue damage and theory, here are two materials essentially equivalent.

CHAIRMAN:

Mr. Hiram Brown of Solar Aircraft Company would like to make some general comments on certain aspects of all the papers this afternoon and specifcally on Mr. Downey's paper.

MR. BROWN:

When I first heard Mr. Downey make the statement that there should be cooperation between design people and materials people, it seemed a prosaic thing to me, something I took for granted. Suddenly when Mr. Fairbairn talked, something clicked in my mind, and it made me feel it is a very, very important thing, not only for what he said, but for the fact he stopped so soon in what he said. I thought of twilight zone cooperation between the design man going so far and the materials man going so far and stopping. What is going to happen between? What brought it to mind were the terms "application" and "maintenance." I remember those two terms sticking in my mind and the other curves that Mr. Fairbairn gave. If I were designing a structure based strictly on curves, I could make very serious errors, neglecting those other two terms, application and maintenance. For example, to point out titanium, which is the favorite subject, pro and con, the listed factors given for 6 aluminum - 4 vanadium alloy do not include two very important things which could upset the whole apple cart. Titanium, almost every kind which I have worked with (and I have worked with many hundreds of them) becomes difficult under proper conditions of atmosphere or temperature. The material, however small it may be, tends to form a very brittle layer. Even stresses such as welding, remotely removed, will open up cracks which propagate into the parent material. Secondly, with titanium alloys, there is the problem of the presence of chlorides, particularly the presence of very simple trichloroethylene everyone uses for cleaning. Can you imagine the pains you will have to take in the shop to avoid cracking, also can you imagine putting a titanium unit in the field for field maintenance and repair, and simply not being aware of these facts? Now this brings about the twilight zone. It is not enough for the design man to use the curves as drawn and plotted in his design, nor is it enough for the materials man to simply ask what it is going to be used for, and getting the answer "airplane." It has to overlap. It has to be of interest to him and he must find out, for example, the atmosphere, speeds, heats and such things as brought out in Mr. Downey's introduction. That is extremely important.

MR. DOWNEY:

I have a question here from Mr. Frank S. Gadomski of the Marquardt Corporation: "Would you recommend that the evaluation of elevated temperature material properties for those materials using coatings be performed under inert conditions to isolate the coating effect as a parameter? Please amplify on the metallurgical aspects." Actually,

some of the things I may say are a reversal of some of the things I said earlier in my talk. Normally, in evaluating the properties of refractory metals requiring coatings, we use inert atmospheres. As a matter of fact, we use the vacuum and the reason for that is the heating equipment which I like to be resistant type heating. Moly-tungsten elements have to be used in a vacuum so that is one good reason why we do these things in a vacuum. As to whether it is more advisable to do them in air with the coating, I don't really know. As to the metallurgical aspects, I am not really sure, without a more specific question.

CHAIRMAN:

Mr. George Kappelt of Bell Aircraft Company wishes to comment.

MR. KAPPELT:

In answer to the question, Mr. Downey, and as was brought up, this is a very pertinent point. Most of our elevated testing of such materials is performed in actual inert protected atmosphere, and with suitable coatings applied, because very often the coating effect changes the design properties or the recommendations of the materials man can change to his design engineer on the actual application of the material. So, I think that as an addendum to Mr. Downey's comment, very definitely, you must separate these effects. You must know the basic properties of the materials as a material, and you must also know, and if you will remember Mr. Downey's comment, you are now working with a composite material. You must know the properties of that composite material. There is just one word of caution and that is do not prejudice yourself by deriving the properties of the composite material in an atmosphere environment or under conditions which are not representative of the actual design application. I believe you will remember that Mr. Downey indicated that most of the coatings are of a refractory nature. They can give you embrittlement at room temperature. You may affect your design conditions if you look at the same system as the same composite material within the temperature environment, or within the loading spectrum that you wish to use. Just one more thing, a more formal remark. I have adopted the phrase that the pilot of these advanced weapons systems is really no longer on his own to fly the airplane as he wishes to fly it, but is subservient to the design engineer and the thermo dynamicist. He must operate the vehicle within the flight spectrum for which it is designed and for which the material systems, including coatings, have been selected.

MR. DOWNEY:

I certainly agree with what has been said. There is a point you made which I don't think you made as strong as you might have. If you are going to evaluate coated specimens or panels, try to evaluate them under actual service conditions.

MR. SCHUETTE:

A question from Mr. E. H. Sheller of Alcoa: "In magnesium alloys, do the fatigue strengths of transfer specimens fall below those of longitudinal and is this applicable to all wrought forms?" I think I should answer that in terms of each of the forms since a single answer is not applicable. The particular graph I showed was a pellet extrusion and these often tend to have a somewhat fibrous texture in them which gives rise to a strict reduction in the transfer properties. In extrusions from ingots, we get a

difference in longitudinal and transverse. The difference, the transverse is lower in varying degrees depending on the kind of extrusion you are making. In sheet and plate, there is no distinguishable difference. We get the same properties in both directions. In forgings it is a little sticky because if we take a short transverse we don't have any data. I feel sure the plates are lower. The long transverse properties, I think, is somewhat nebulous. We talk quite glibly about a long transverse direction. If one may draw an analogy to a balloon being inflated, as the balloon is inflated the skin is stretched and which is longitudinal and transverse direction of stretch? I don't know. Most of them, close to the general shape of the product, push it in all directions, except insofar as we may have started with extruded stock and this is longitudinal direction and we may have retained some of this in the finished forging.

MR. FAIRBAIRN:

I have a question from Mr. A. Glasser from The Budd Company: "Have you done any testing through a range of heat treatment to determine if there is perhaps an optimum heat treat?" I presume this relates to optimum from the standpoint of fatigue strength. We haven't done much of this. We have been evaluating heat treat level from the other standpoint of the level of crack propagation, testing stress correlations, some of those things, but we have not tried to analyze the heat treat level in fatigue strength.

The second question is from Major Dunning of the Air Research and Development Command: "Is it your belief that titanium sandwich structure does not have the potential, in its temperature range, up to 800 degrees F., that stainless steel sandwich has in this temperature range?" Below 800 degrees F. certainly titanium does have the potential in comparison to stainless steel. The main factors holding back brazed titanium sandwich structures are brazing of alloys, compatibility of heat treat of titanium and the brazing cycle, and the availability of the titanium alloys which are brazable and weldable in the product forms required.

MAJOR DUNNING:

Actually, what I had in mind was the application of the structure in your weapons system, using it as a structure and the difficulties that we might get into in the fabrication of the titanium sandwich versus the difficulties we have had with stainless steel.

MR. FAIRBAIRN:

There is an infinitely greater number of problems involved in making the titanium sandwich than in making stainless steel, and they are related to some of these things I mentioned such as brazing alloys, means of applying the heat treat with the brazing cycle, and so on. But, theoretically, on a comparative basis, titanium could compete very well with stainless steel as brazed sandwich up to 800 degrees F.

MR. STULEN:

Mr. E. L. Horn of the Wright Air Development Center asks: "Have you any feel for inclusion comparisons between axial and rotating beam fatigue properties?" Since

axial fatigue specimens have a greater volume, there is a greater chance of having a larger inclusion in the high stress region and therefore, the fatigue strengths of axial loaded specimens should be lower than that of rotating beam specimens. It is a statistical problem of the exclusion of inclusions as well as the avoidance of inclusions.

Mr. L. Rastrelli of Southwest Research Institute asks: "In your Figure 4, isn't there a limited interpretation on that portion of the probability curves that are horizontal? That is, what do the 10, 20 and 90 per cent curves mean in terms of life?" They mean the same thing as in the finite range. In other words, if at the lowest curve you run some specimens, you will find on it 90 per cent survivals, that ten per cent will fail eventually and 90 per cent will have infinite life, if there is such a thing. Now, the division of this part of the curve requires a different statistical treatment than the finite range of the S-N curve.

PROFESSOR DOLAN:

This question is from Mr. J. Redfern of Curtiss-Wright: "If damage occurs below the endurance limit, why does step testing of some materials show good correlation with tests run at one stress level?" I presume what he is referring to is some instances where tests may be conducted on a component in which it is stressed purposely below the endurance limit and repeated for a large number of cycles, increased to 5,000 psi, the test repeated, and this process repeated until failure occurs or perhaps endurance stress level at which it broke and the previous stress level. We included in our paper that damage below the endurance limit, primarily that which is created really by a prior stress history above the stress limit. Therefore, if one can start at low stress levels and run cycles of stress in various steps, presumably you will not hit a great deal of damage until you get above the endurance limit. If you definitely had a stress history of cycles above the limit, then this kind of step testing, presumably in our theory at its present maturity, failure will occur at a level somewhat lower than you expected the endurance limit.

MR. DOWNEY:

A question from Mr. Carter, Martin Company: "Will you please comment on alloying of noble metal coatings with moly or columbium base metals?" I don't know what the diffusion rates are between the various noble metals and refractory metals. The work we did previously on noble metals had a barrier layer between the refractory layer and the noble layer for reasons other than diffusion. Other people in the audience may have more experience in that than we have had and could give it a try.

CHAIRMAN:

Anyone want to volunteer on this one?

VOICE:

It is much the same as we discussed before and that is if your temperature is

high enough you lose your protective coating and second you will form a new alloy with infinitely different properties. You may have a great degradation of properties that would precipitate a fatigue failure or weaken the fatigue life here before your testing of the bare material would indicate it.

SAM GOLDBERG, BuAer, Washington:

A number of years ago we sponsored some preliminary work at Battelle Institute, and gold acted to reduce the diffusion rate of the moly and columbium. When they prepared some test specimens which were subsequently evaluated at the Naval Air Material Center, the layer contained a significant number of pores so that no real benefit was realized. This was preliminary and may bear some further work.

MR. DOWNEY:

I'll say this to you, Mr. Carter, it has been tried before by other people. We have tried it, and the intermediate barrier layers and results we are looking for are not too good. Maybe you can give it another try, but there is a possibility that we are going to get alloying.

Mr. B. Simon of Aerojet General asks: "Has a coating of Teflon been investigated and if so, with what results?" No, it has not been investigated, but I just wonder if you were operating with a coating of Teflon at temperatures to 2,000 or 2,500 degrees F. if it would simply not decompose. I would think it would decompose. I know somebody is going to say use it in rocket nozzles, but I would think as a protective coating it is not much good.

MR. SIMON:

I am the one who asked that question, and the reason is I understand there is a new general Teflon that can stand temperatures up to 2,000 degrees. I was wondering if you could comment on that.

MR. DOWNEY:

I am not aware of it. Even if it were good to 2,000 degrees, that is not good enough. We must go to 3,000 degrees.

The final question is from Mr. I. R. Kramer, Martin Company: "What coatings are recommended for molybdenum and columbium?" Now, that all depends on who is answering the question, but I will tell you what my recommendations are and I am sure some other people will have some comments. For moly on shorter time applications where the thing does not have to last great lengths of time we have been using a coating which is as good as you can get. It is a chromium and moly disilicate coating. For longer periods of time, and there may be some argument, the best we know of at the lab at GE is a coating which consists of electro deposited chromium, a thin chromium plate topped by a layer of flame sprayed aluminum. For moly, that appears the best for us. For columbium the best coating we know of so far, and our efforts are not as far along, is a coating of flame

sprayed aluminum and an aluminum oxide which is sealed with a high melting point silicate glass. I can't tell you which is best. We are still working on them now, but these things have given us reasonably good results and it is reasonable to hope it will do the job.

The last one is from unknown person: (Editorial Note: Reporter was not able to hear the entire question nor was the question card turned in. This question concerned the effect of manufacturing processes for castings on fatigue life.) I think I'd like to see if I can restate that question. What you are looking for is how are you going to design or otherwise assure yourself that you will get the best possible fatigue properties where you want the best properties. If that is the case, I would have to simply reinforce what Mr. Brown said about working between the designer and the producer on this thing and there have been some papers on this, from Alcoa and other sources. It is absolutely essential, in getting the most out of castings, to work closely with the foundryman to see to it that the job is done in a way which is designed to get the most where it is necessary, because we have to face certain things in castings. In general, we are now limited in magnesium castings in that you simply can't get a perfect quality casting throughout the entire area of the casting in any kind of a normal production part, but you can arrange to chase such things as porosity to places where it doesn't matter. You must work with him to see to it that it gets to such places. Also, it is not possible to select perfect quality. It is unattainable in the cast process, and the minor things you run into probably are not very significant. You may set a top limit on the fatigue strength you can achieve. It may be pretty much fixed, regardless of how you work, and still get good quality where you want it. Grain size and porosity, either one of them, have relatively minor effects in the normal range of good castings and are always obliterated by the effects of design notches and stress raisers that go into the casting. If you machine, this kind of thing is rather dangerous and so if you are going to machine, if you have specifically critical areas, this is the place you must work with your foundryman. The thing can be done with chills and that sort of thing for a specific area with good sound casting material.

CHAIRMAN:

We are over the time when we were supposed to be finished. I know Mr. Schuette has a couple of questions left and Mr. Fairbairn and Mr. Stulen also have several questions left. I am now going to reverse things and instead of asking the audience if they want to continue, I'll ask the authors if they wish to continue.

All right, those of you who have questions that didn't get answered, come quickly forward and make arrangements with the authors to get together and get the questions answered.

Forum Session II-B is adjourned.

INTRODUCTION TO SESSION III

COLONEL HARVEY P. HUGLIN

WRIGHT AIR DEVELOPMENT CENTER

As shown in the program we have a very distinguished gentlemen scheduled for this session, Mr. Bisplinghoff, but unfortunately he could not make it. In seeking a substitute we were fortunate in securing the services of a very capable individual, Mr. Carl E. Reichert. I have known Mr. Reichert for some time and appreciate his capabilities. During his thirty years at Wright Field he has served as Chief, Design Branch, Aircraft Laboratory; Assistant Chief of the Laboratory's Operations Office and currently as Chief, Structures Branch. Because of his technical ability and vigorous approach to the solution of problems within his scope of responsibility, Mr. Reichert has become well known and nationally recognized in industrial and governmental circles. He received the Legion of Merit in 1946 for his outstanding work in aircraft design, and is presently a member of the NASA Research Advisory Committee on Structural Loads. He holds a reserve commission of Colonel in the Air Force Reserve, and is a native Daytonian. It is my pleasure to introduce Mr. Carl Reichert.

BASIC STRUCTURAL CONSIDERATIONS FOR FATIGUE DESIGN

By

Paul Kuhn

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Langley Field, Virginia

Effective fatigue design of structures can be expected only if the three ingredients of fatigue analysis - loads, stress analysis, and fatigue behavior of simple specimens are adequately known and adequately utilized. In view of the large volume of information published in the last five to ten years, no attempt is made here to summarize the knowledge in these three fields. Instead, attention is focused on gaps in our knowledge and apparent weaknesses of procedure in the last two fields. Design procedures are discussed briefly. A realistic appraisal of the problem leads to the conclusion that the mean life of the fleet should be predictable in the foreseeable future with reasonable accuracy. However, the time at which the first failure appears in a fleet is predictable with much less accuracy; consequently, maximum possible use of fail-safe ideas in design and of vigilant inspection in operation is advocated.

INTRODUCTION

Until 1948, most airplane designers considered fatigue simply as a nuisance which resulted in complaints from operators about repair costs. Since then, a number of fatal crashes have demonstrated that fatigue must be treated as a dominant design criterion, and since airplane crashes are always given world-wide publicity, the term "metal-fatigue" has become very familiar even to newspaper reporters and the lay public. Machine designers, automotive designers, naval architects and other engineers also had suffered from fatigue troubles for a long time; it was no great surprise, then, that in the two-year period 1955-1956, there were no less than four international meetings on fatigue, one being held in London and repeated in New York.

Thus, there is no lack of awareness of the fatigue problem in aeronautics. Neither is there lack of information. The London-New York meeting alone resulted in a volume containing about 1,000 large-size pages, and the Index of new fatigue publications published by the ASTM adds about 300 papers each year to a fund of several thousand previous ones. The uninitiated might deduce from this that the problem of aircraft fatigue is well under control. Unfortunately, this deduction appears to be optimistic. Discounting airplanes designed at a time when little attention was paid to fatigue, and considering only new types which are known to have had ample attention lavished on the fatigue aspect of design, one still encounters disconcerting amounts of fatigue difficulties.

In view of this situation, no effort will be made in this paper to produce a comprehensive summary of what is known - a task that would require a book to do justice to it, rather than a paper. On the other hand, an effort has been made to focus the spot-light of attention on areas where adequate knowledge is still lacking and on weaknesses of procedure.

GENERAL SURVEY OF THE PROBLEM

For the purpose of general orientation, let us take a look at the problem of fatigue design as a whole.

Decisions on design are based on considerations of three main factors:

1. Performance
2. Safety
3. Cost (initial plus upkeep)

The requirements dictated by these considerations are often antagonistic toward each other, and the relative weight that should be given to each is a matter of personal opinion, within the latitude left by specifications and regulations.

Whenever a problem is complex, there is a tendency to attempt the solution by enouncing simple principles. Principles often are competitive, and in their initial discussion, much heat is often generated even at strictly sub-sonic speeds. Let us glance at two such pairs of competitors.

Safe-life versus fail-safe. - It is recognized that aircraft cannot be designed for an infinite life, except perhaps a few components; they must be designed for finite life. The simplest approach to the problem - in theory - is to establish by some means a so-called "safe life" and to retire the structure from service when the safe life has been reached.

Unfortunately, fatigue lives are subject to large scatter, and consequently, the determination of a "safe life" is beset with grave uncertainties. To get around this difficulty, the "fail-safe" school of thought attempts to build the structure in such a way that its strength is not reduced catastrophically even by cracks of very sizeable length. Inspection and repair is then counted on to take care of cracks before they produce catastrophe.

In theory, the fail-safe designer need not pay any attention to fatigue life. In practice, he must pay considerable attention to obtaining an adequate fatigue life, because a customer exasperated by excessive repair costs could put him in dire straits. Vice versa, the proponents of safe-life design are forced to admit the difficulty of ascertaining the safe life with adequate accuracy, and consequently they agree that the incorporation of fail-safe ideas is a worth-while precaution. Thus, the gap between the two schools of thought has been narrowed very appreciably.

Analytical versus experimental approach. - Another pair of competitors are analytical and experimental approach. In the last few years, not too much has been heard of this controversy, and indeed, there is scant justification for it to exist. There are mechanic-geniuses who can build complicated machines without ever putting a line on paper, and in a Western state, such a genius is reported to build small suspension bridges without benefit of any calculation. When dealing with airplanes which cost many millions of dollars per copy, such an approach - making no use of theory - is utterly impossible. On the other hand, much theory has been written that ought to be filed in the basement of the library to minimize the risk that a practical engineer might see it - and waste his time reading it. The most successful engineer uses theory wherever it is useful and sound, and experiment where it is more conclusive or faster than theory.

BASIC FATIGUE KNOWLEDGE

In this scientific age, it should be unnecessary to do more than mention the fact that long-range progress in design is possible only by improving our knowledge of all facets of the problem quantitatively - and by applying this knowledge. Let us then take a high-speed flight over the field of basic fatigue knowledge for the purpose of high-lighting some areas where knowledge is still inadequate and some other areas where existing knowledge is not adequately utilized.

Material properties. - The fatigue properties of materials are obtained by tests on plain (unnotched) specimens, the most important types being axial-load, rotating-beam, and sheet-bending specimens.

For the design of airframes, axial-load tests are the most important ones because they are directly applicable. The results on rotating beams and sheet-bending specimens are subject to a stress-gradient effect closely related to the size effect to be discussed later. The problem of deriving axial-load fatigue allowables from either type of bending test has not been studied sufficiently to arrive at generally accepted conversion factors.

Surface roughness and residual stresses near the surface can be important, particularly residual stresses due to heat treatment. The polished finish used on plain fatigue specimens is seldom representative of service articles, but is usually justified by claiming that it produces a "par value." However, since notches on notched specimens are often not polished (especially on light alloys), it is

questionable whether this arbitrary par value is the most useful one.

Notched specimens under simple loading. Plasticity and size effect. – All practical structural parts contain stress raisers required by their functional design. As a result, fatigue research has always been to a large extent research on the effect of various stress raisers. Let us confine our attention first to specimens with simple notches, subjected to fully reversed loading of constant amplitude.

The simplest case of a notch is a circular hole in a thin sheet under axial load. Figure 1 shows on the left the stress distribution over the net section as given by the theory of elasticity for a steady or static load. The theoretical factor of stress concentration K_T has a value of 3, when the sheet is wide compared with the diameter of the hole.

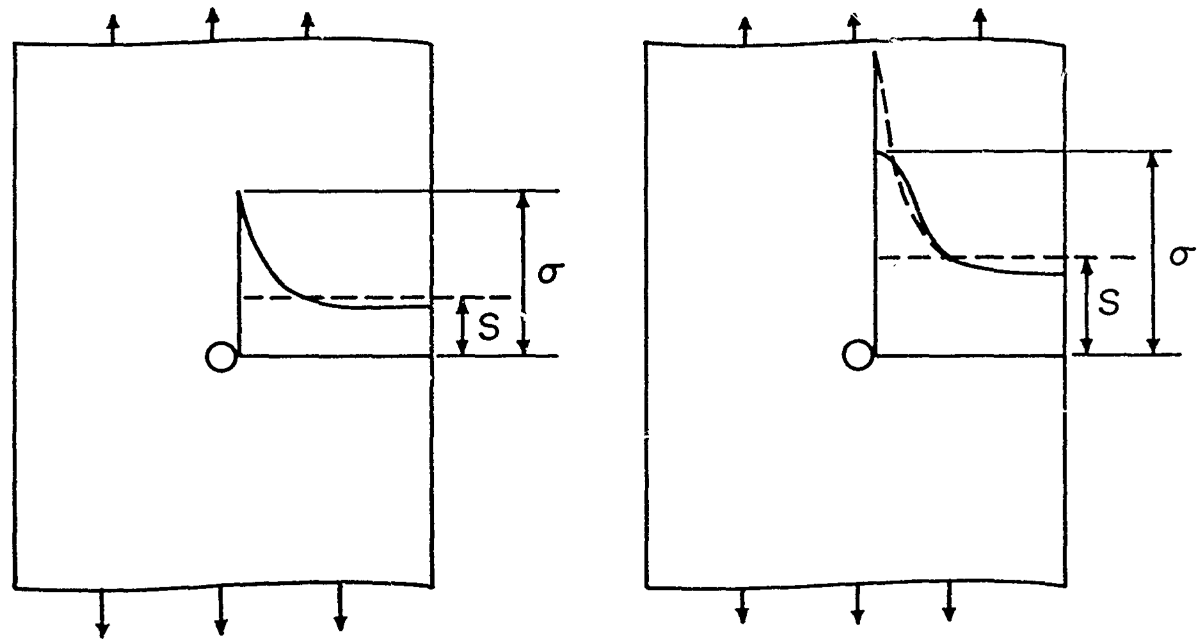
On the right is shown the stress distribution for the case where the peak stress σ is in the plastic range. The stress-concentration factor for this case, denoted by K_P, can be estimated by the formula shown, generally with reasonable accuracy as long as σ is not close to the failing stress. The value E_1 is the secant modulus corresponding to the stress σ, while E_2 is the secant modulus corresponding to the stress at a large distance from the hole.

Let us now consider the same type of specimen under fatigue loading – to begin with, fully reversed axial loading. Figure 2(a) shows the S–N curve for such a specimen and the S–N curve for a similar specimen without hole. The ratio of the ordinates S_A/S_B at any given value of N such as N_x is called the fatigue factor K_F. In Figure 2(b), K_F is plotted against N. The value of K_T is also shown as a dashed line, but only in the high-cycle region because it is applicable only in this region. As N becomes large, K_F approaches K_T (if the hole is large); as N becomes small, K_F generally approaches a value near unity.

It should be noted that the stress for the notched specimen plotted in Figure 2(a) may be based either on the net section or on the gross section. Correspondingly, there should be two curves for K_F in Figure 2(b). Failure to note this possibility of ambiguity has caused – and is causing – great confusion; care should therefore be taken to specify on which section the stress is based. Evidently, a specification of K_F should also be accompanied by the specification of the cycle number N at which it applies. The value of K_F at large values of N (at the fatigue limit) is the most useful one and will be designated hereafter as $K_{F\infty}$.

The procedure of plotting K_F against N as in Figure 2(b) is a natural one if one begins with a set of S–N curves as in Figure 2(a). However, it is physically more meaningful to consider K_F as a function of the stress S rather than the cycle number N. This is possible because S and N are related by the S–N curve of the material. The Stowell formula given in Figure 1(b) has been used with fair

STRESS CONCENTRATIONS AT HOLES



$\sigma = K_T S$

$\sigma = K_P S$

$K_P = 1 + (K_T - 1)\frac{E_1}{E_2}$

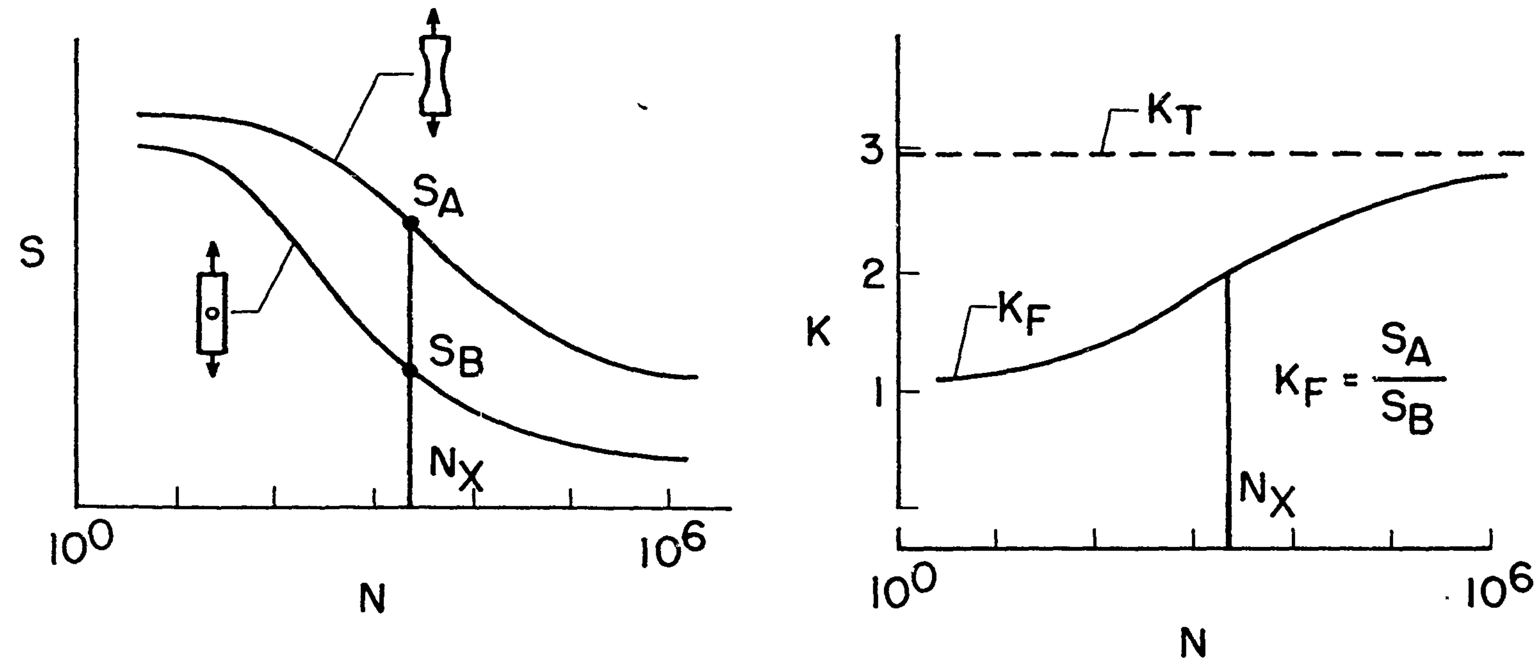
(STOWELL FORMULA)

ELASTIC CASE
1a

PLASTIC CASE
1b

NASA FIGURE 1 KUHN

FATIGUE FACTOR



2a 2b

NASA FIGURE 2 KUHN

success to estimate the variation of K_F with stress. The reliability of this method of estimating, however, should be investigated more fully.

Figure 3 shows the variation of $K_{F\infty}$ for a sequence of geometrically similar specimens. Since the specimens are similar, the value of K_T is the same for all of them, as shown by the dashed line. The values of $K_{F\infty}$, however, vary. For large specimens, $K_{F\infty}$ is nearly equal to K_T, as indicated in the previous figure. As the specimen becomes smaller, however, $K_{F\infty}$ becomes steadily less and approaches unity for a very small specimen; in other words, a very small hole has no effect as stress raiser. This variation has been termed geometric size effect, or stress-gradient effect, in order to distinguish it from metallurgical size effects.

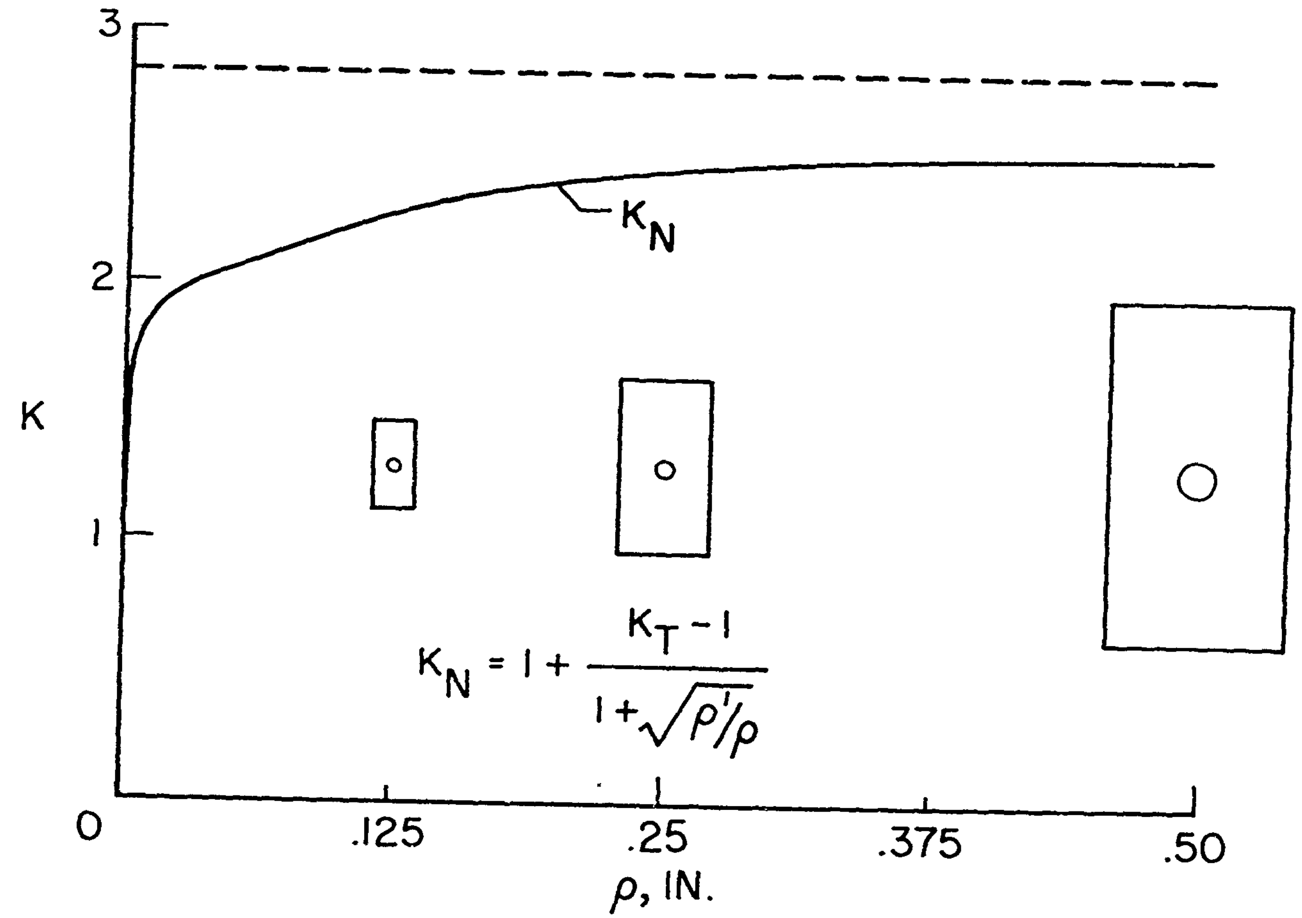
The size effect can be estimated quite well by the formula shown at the bottom of Figure 3, which gives the K-factor for any given size of notch as a function of three quantities: the theoretical factor K_T, the notch-radius ρ, and a material's constant ρ' which we call "Neuber constant" because the formula was proposed in a book by H. Neuber (Ref. 4). Values of ρ' for steels are given in ref. 5. For high-strength aluminum alloys, $\rho' = 0.02$ inches gives an acceptable approximation, although there appears to be more scatter than for steels.

The importance of this size effect is illustrated by Figure 4. This figure shows S-N curves obtained from axial-load tests on aluminum-alloy sheet specimens. The upper curve is for plain specimens, of either 2024-T3 or 7075-T6 alloy, which give identical results for the range shown. The middle curve is for specimens with Vee-notches, in a configuration which is closely similar to the longitudinal cross section through a standard rotating beam as used by the Alcoa Research Laboratories. The bottom curve finally is for geometrically similar notched specimens 24 times larger. The difference in stress carried is obviously large. The small specimens are typical of those used to obtain notch-fatigue data on material. It is obvious that data of this type are highly optimistic and must be corrected for size effect before being applied to full-scale parts.

The size effect can be described by means other than the Neuber constant ρ'. The effect is basically due to the granular structure of the material and the existence of a stress-gradient, and this gradient may be used as parameter. Another possibility is to use an "effective notch radius." The notch sensitivity index introduced by R. E. Peterson, plotted against the notch radius, serves the same purpose. Whatever method is used, the important item is that the fatigue engineer should be always fully aware of the existence of the size effect and make allowance for it.

One long-range goal of fatigue research is to be able to predict the fatigue behavior of a notched specimen from the S-N curve of the unnotched material, some tests to define the size-effect or the notch-sensitivity characteristics of the material, and the theoretical stress-concentration factor. For specimens under completely reversed loading, it may be said – with a bit of optimism – that this goal is in sight, for cases in which the complications by residual stresses are not too severe.

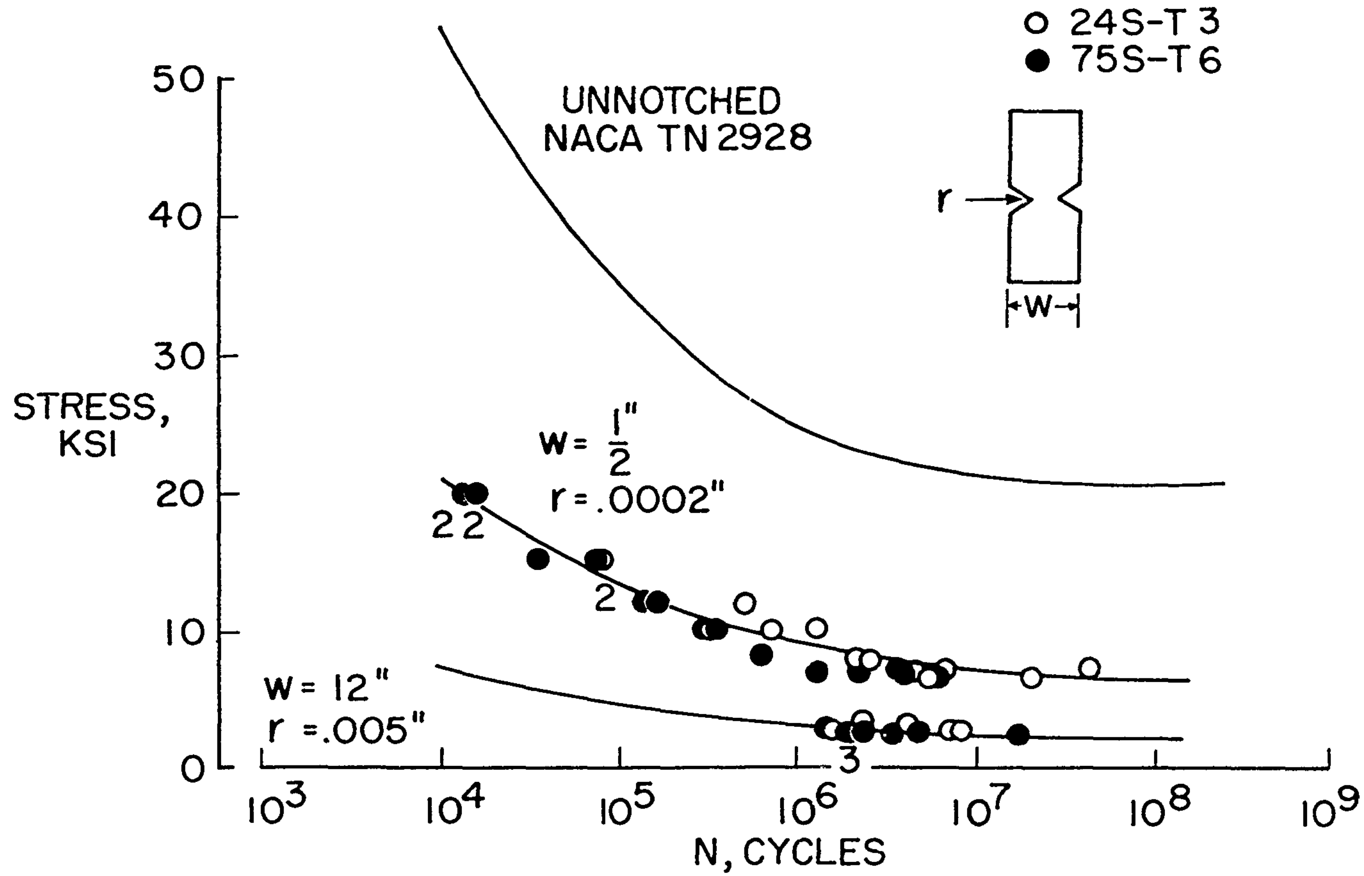
GEOMETRIC SIZE EFFECT



NASA FIGURE 3 KUHN

AXIAL LOAD FATIGUE TESTS OF SHEET SPECIMENS WITH SHARP NOTCHES

R = -1



NASA

FIGURE 4 KUHN

The fatigue behavior of notched as well as unnotched specimens may be affected by residual stresses. For stresses which are introduced intentionally, as in shot-peening, there is a certain amount of empirical knowledge. For stresses which are not introduced intentionally, however (quenching, machining, and forming stresses), our knowledge is very spotty. The problem becomes very severe for ultra-high strength steels, often causing disappointingly low fatigue strength as well as large scatter.

Complex loadings and complex specimens. - In practice, fully reversed loading is encountered only in special cases. A fairly wide range of practical loadings can be described as a steady mean stress (which varies only infrequently) on which is superposed an alternating stress of frequently varied amplitude.

It has been customary to provide materials fatigue data for a simpler case: a steady mean stress on which an alternating stress of constant amplitude is superposed. Even for this rather simple case, a severe difficulty is encountered in attempting to deal systematically with notched specimens: it is no longer obvious how the fatigue factor should be defined. A number of definitions have been proposed, and research has so far led only to the conclusion that none of these definitions is satisfactory over the entire range, even with very nominal demands on accuracy.

When a notched specimen is loaded the first time, the peak stress at the bottom of the notch will become plastic above a certain load. On unloading, there will be a residual stress system, and on additional loadings, the material in the critical region at the bottom of the notch will operate under different conditions (chiefly, different mean stress) than those deduced naively from the external loading. A procedure which takes the residual stresses into account appears to be capable of dealing with the problem over quite a range. However, more comprehensive research is desirable to establish the limitations of this procedure.

The problem of variable-amplitude loading has usually been dealt with, for purposes of an analytical approach, by using a cumulative-damage criterion. A number of investigators have been working very actively on this problem for several years, and it is hoped that some useful conclusions will emerge from all this work in the not-too-distant future.

Besides the problem of complex loadings, the designer must face the problem of complex specimens such as fittings and joints. Failure may be influenced by fretting; since this involves chemical actions (in the opinion of most investigators), it is clearly beyond the scope of mathematical stress analysis. In other cases (for example, at edges of countersunk holes), areas of stress concentration may be identifiable qualitatively, but the geometry is too complex to permit an analytical attack. For items of this type, then, fatigue tests are the only possible method of design approach for the indefinite future. It is becoming more and more customary to run fitting and joints tests as spectrum tests, thus solving the problems of complex geometry and of complex loading at the same time.

For convenience, the fatigue behavior of a joint is characterized by an effective or equivalent K-factor, which is the K-factor of a simple notched specimen displaying similar fatigue behavior as the joint in question. (This definition is necessarily very loose because different procedures are used to arrive at a factor.) This, incidentally, is also a field in which confusion can arise due to neglect of

size effect. It is common practice to quote the theoretical factor K_T of the notched specimens used as basis of comparison, but since most of these specimens have rather small notch radii, their fatigue factors are substantially less than their theoretical factors.

Environmental effects. – Environmental effects on fatigue fall into two main classes: corrosion and elevated temperature.

Corrosion effects obviously depend on the nature and the concentration of the corrosive medium. Rather nominal changes in temperature may have large effects, as may the presence of catalysts. A large amount of ad-hoc testing has been done for certain applications such as pump design, and the experts in this field appear to agree on one fundamental advice: the only way to obtain reasonably reliable information is by tests which simulate service conditions as closely as possible. This condition is often extremely difficult to meet.

Atmospheric corrosion is a potential source of trouble. An investigation of this subject was initiated by the NASA about two years ago, and the results obtained so far (Ref. 6) indicate a larger effect than anticipated, although the near-coastal atmosphere at the test station is not considered to be unduly severe. There has been some talk about an international effort to study this subject more extensively. On the problem of less frequently encountered, but more potent, corrosive agents, the designer can do no better than to listen to a maintenance man for some free advice on design (after putting asbestos shingles over his ears).

Elevated temperature combined with fatigue has always been a problem in engines; consequently, there is some information on this subject. Unfortunately, most of the tests made have been restricted to narrow bands of temperature and frequency of application of load. The latter restriction dodged a key-issue in the more general problem: room-temperature fatigue depends on the number of cycles, but is essentially independent of time (or frequency of load application). If the temperature is raised sufficiently, however, the useable strength of materials becomes essentially a function of time. At intermediate temperatures, the fatigue strength is thus a function of cycle numbers as well as time. Knowledge in this field is very spotty, but there appears to be one relieving feature: the ratio of fatigue strength to static strength tends to increase as the temperature increases in the range of interest here.

Scatter in fatigue. – It is well known now that fatigue tests on nominally identical specimens result in considerable scatter. If an attempt is made to estimate the life of a component by tests of a few specimens, the mean life of the test group is divided by a "scatter factor" to arrive at a first estimate of service life.

From the statistical point of view, such a procedure is very crude; it may be termed a "first-order approximation" and can be justified as reasonably adequate only if the test conditions used are on the severe side.

In a more refined approach, a specification is given of the percentage of components which may fail to reach the specified life. If the component in question is a single vital component, then the desirable percentage would presumably be of the order of 0.1%. In order to be able to apply such a specification, the designer must have at his disposal the probability-of-failure curve with a reasonable

confidence level at this probability level. Such a curve can be obtained only by testing at least 1,000 identical specimens under identical conditions.

For simple material specimens, test series of this magnitude would be feasible, but would constitute quite an effort. The material for the specimens should be obtained by random sampling at the place of manufacture, with the sampling period long enough to cover possible variations in the manufacturing process. It is rather doubtful that such test series can be carried out for more than a very few materials. It can only be hoped that the statistical characteristics of interest do not differ too much between materials.

A failure rate of 1 in 1,000 may be appropriate for a single vital component, as suggested. Under less stringent conditions, that is, when fail-safe features exist, the tolerable failure rate may be between one and two orders of magnitude higher; the task of providing fatigue probability data then becomes much more practicable.

CRACK PROPAGATION AND CRACK SENSITIVITY

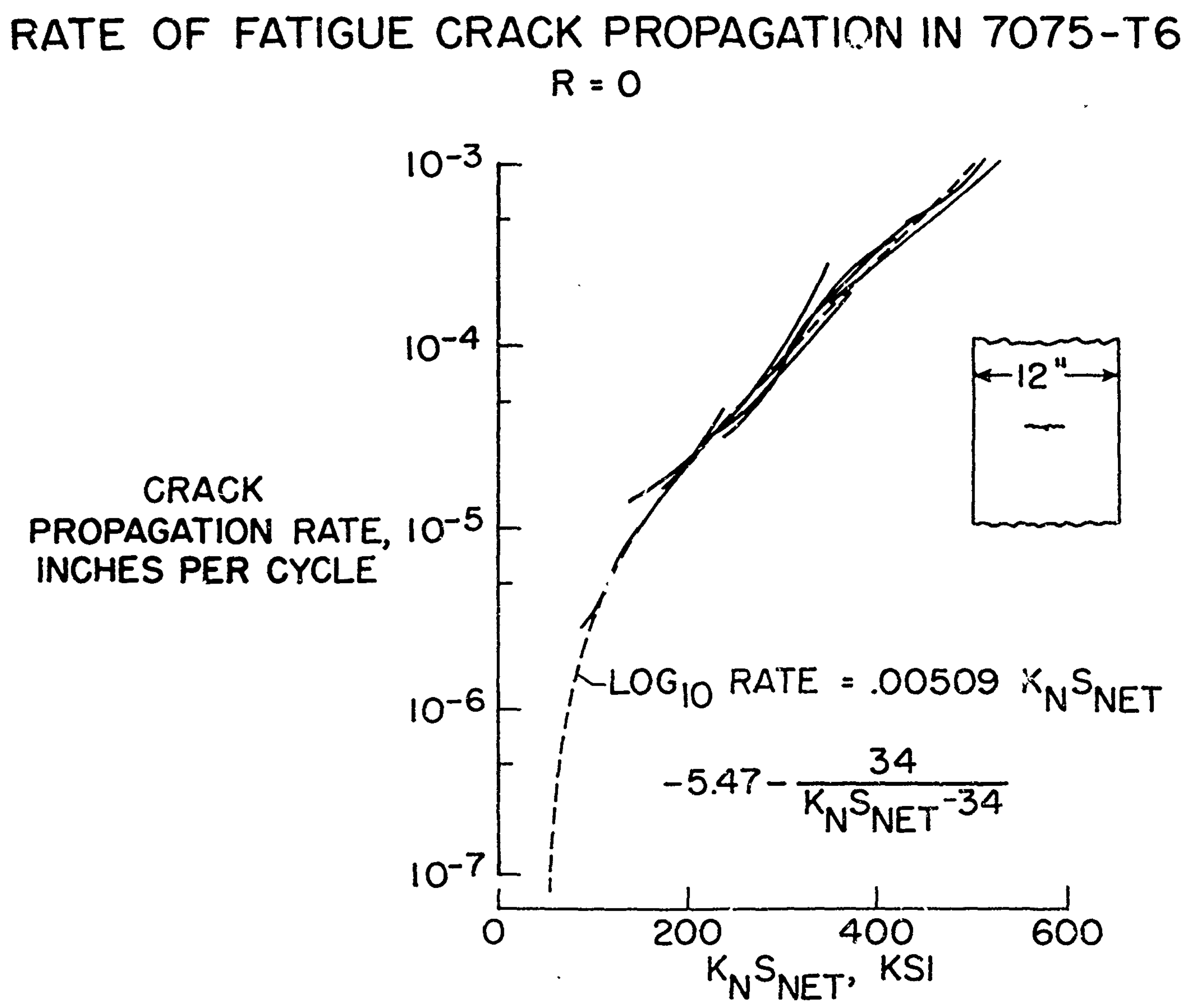
The process of fatigue failure is usually divided into three stages: the incubation period, the crack-propagation period, and final failure. The dividing line between the first two periods is the appearance of a detectable crack. Since the ability of detection depends on the magnification of the microscope used and on the perseverance of the observer, the dividing line is obviously not very well fixed.

Crack propagation is usually discussed under the heading of fatigue. The subject of crack sensitivity, or strength of parts containing cracks, is in a no-mans land between fatigue and strength of materials. However, both subjects are key items in the engineering discipline of fail-safe design; it is therefore appropriate to discuss them jointly.

For a good fail-safe design, the material should have a low rate of crack propagation and a low crack sensitivity. Not so obvious is the question whether a long incubation period is desirable. Insofar as known, the static strength of a structural part does not change at all, or only to an insignificant amount, during the incubation period; thus, a long incubation period may be considered desirable. On the other hand, if this period is long, there is little time left for the development of a crack large enough to be readily detectable and serve as warning.

In one instance, a very important but readily replaceable part was being replaced when cracks were found. A higher-strength alloy was tried for this part, but was not introduced into service because the warning period was too short - the part failed before cracks were detected.

The problem of crack-propagation as well as that of crack sensitivity has been made amenable to prediction for aluminum alloys by utilizing the size-effect formula given in Figure 3 (Refs. 7 and 8). Figure 5 shows a sample comparison between calculated and experimental rates of fatigue crack propagation. Figure 6 shows the strengths of flat-sheet specimens containing cracks for two different widths of specimen; attention is called to the fact that the theory predicts the effect of width of specimen very well. Figure 7 shows - also for flat-sheet specimens - the superiority of 2024-T3 over 7075-T6 alloy. Figure 8 finally shows two typical curves for the bursting strength of cylinders with longitudinal cracks. The two



NASA FIGURE 5 KUHN

STRENGTH OF CRACKED SPECIMENS

7075-T6 ALLOY

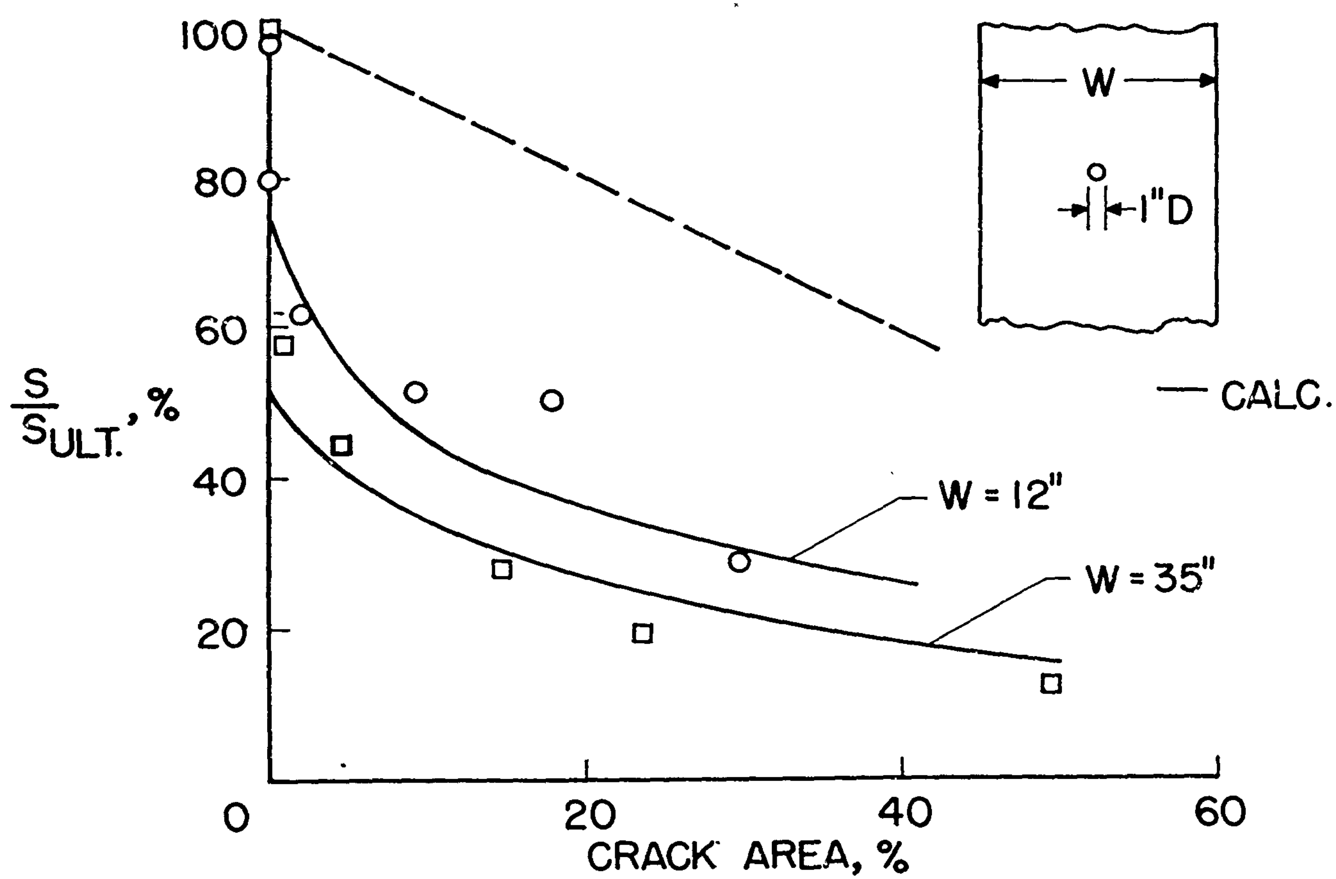


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FIGURE 6

KUHN

STRENGTH OF CRACKED SPECIMENS
2024-T3 AND 7075-T6 ALLOY

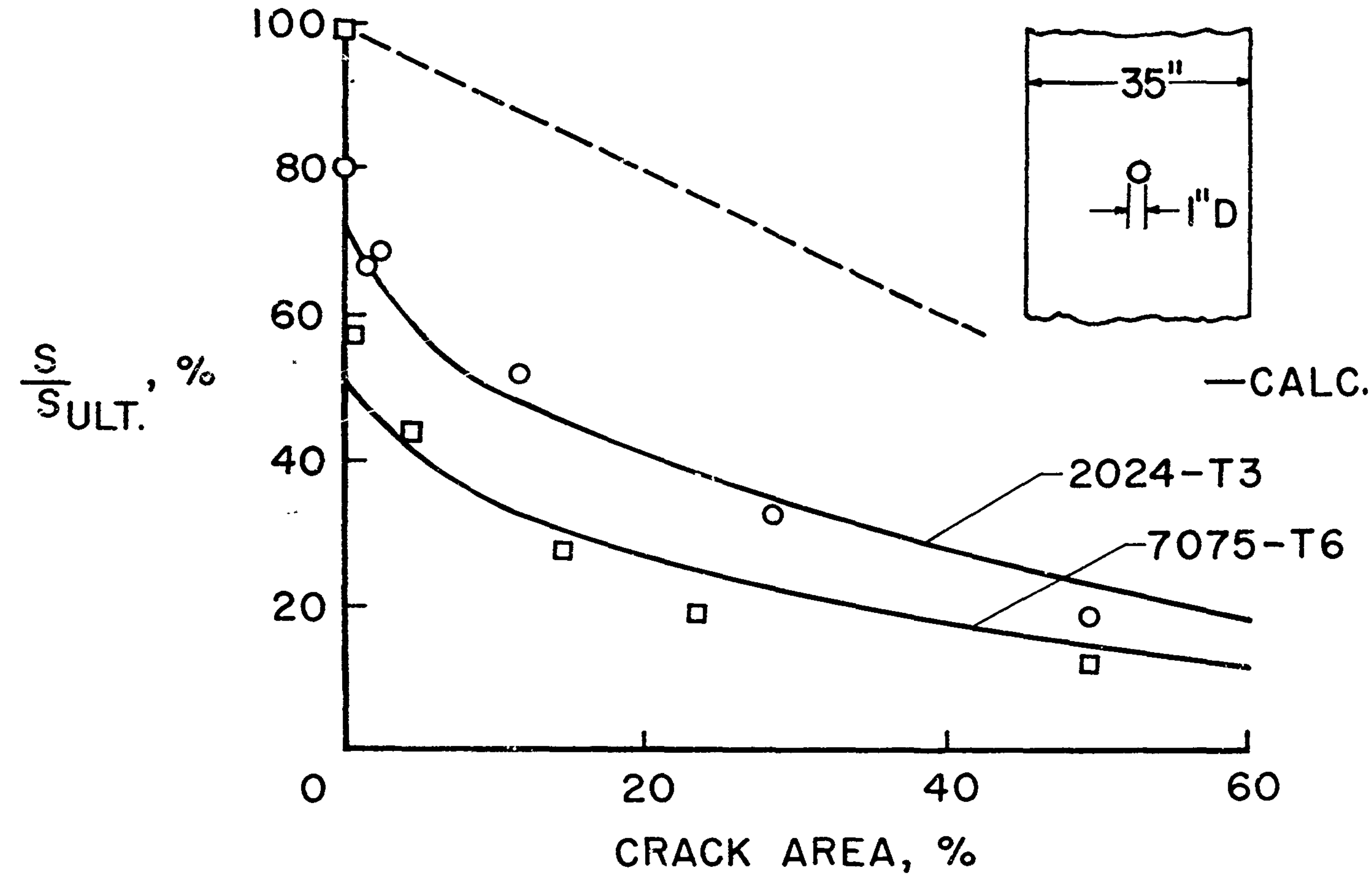

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FIGURE 7 KUHN

BURSTING STRENGTH OF CRACKED CYLINDERS

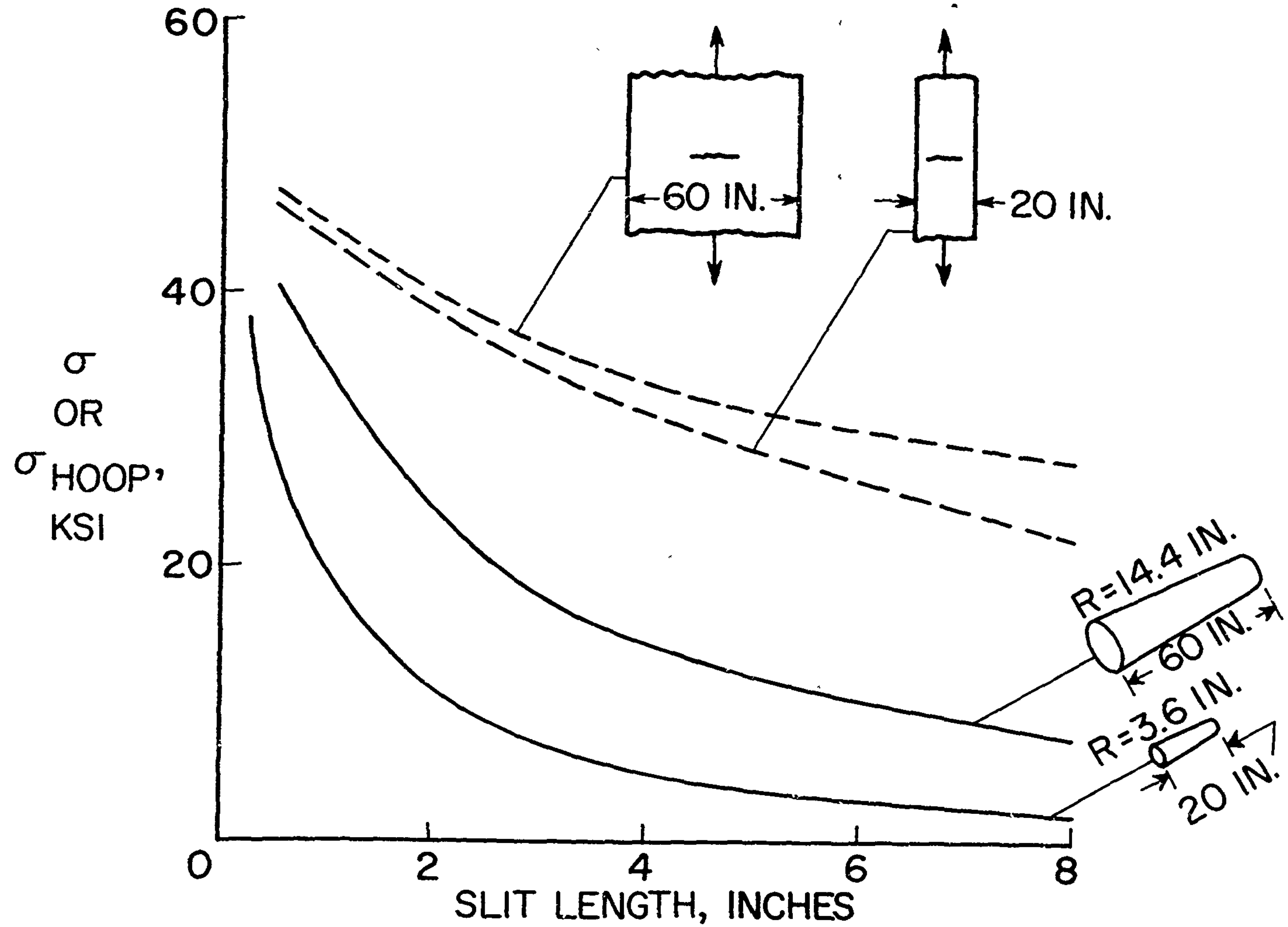

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FIGURE 8 KUHN

upper curves are for flat sheet under uniaxial tension. It is obvious that curvature has a very important bearing on the results. (Ref. 9)

At present, there is considerable activity in various places devoted to the study of crack sensitivity in high-strength and ultra-high-strength materials. Low strength encountered in structures such as pressure vessels is attributed chiefly to crack sensitivity and presents a serious obstacle to the use of these materials. The problem is complicated for many materials by such phenomena as transition temperature effects and high sensitivity to trace elements, with hydrogen being the best-known offender. Since the failures are associated in some materials with welded joints, the multitude of parameters which affect weldments further complicate an already complex picture. Much research is evidently necessary in this field.

HOW CAN THE STATE OF THE ART BE IMPROVED?

In the introduction, it was intimated that the present state of the art is somewhat less than satisfactory, inspite of a seemingly stupendous volume of information. How can this state of affairs be improved?

In the preceding discussion, it has been pointed out that research knowledge is still inadequate in a number of areas. Consequently, there should be no let-up in the conduct of research.

The research and test information already available constitutes a flood - enough to inundate and suffocate the hapless designer. No greater contribution can be made by any one man than to correlate and digest all the information available in a sizeable area and to condense it into a small package readily useable by the designer.

Concerted and continued efforts should be made to educate detail designers and analysts in fatigue. It has often been said that detail makes or breaks a design - and this saying is much more applicable to fatigue than to static design.

More efforts should be made to improve the transmission of knowledge and know-how from one generation to the next. (A generation of detail designers, according to some sources, may mean as little as three years.)

Less reliance should be placed on isolated tests, often made in frantic haste in the search for a fix.

Finally, very concerted efforts should be made to analyze available tests as a help towards better quantitative understanding. Many tests are made initially simply to verify the design of a specific item for a specific airplane. Viewed in this light, the test does not require any analysis. However, viewed as a proof-stone for checking the adequacy of our methods of calculation, it can be very valuable - if properly analyzed.

Proper analysis of the type referred to cannot be expected from test engineers. Hounded by urgent test schedules and often quite unfamiliar with analysis, test engineers tend to see a test as an end in itself, rather than a means to an end. A small group should be divorced from current routine, but should have ample contacts with research workers in all sorts of institutions and with their counter-

parts in other organizations. This group, in turn, should give regular lectures to the designers and stress analysts. It would, in the course of time, materially improve the quantitative knowledge of fatigue, and it could provide the transmission of know-how to newer generations.

STRESS ANALYSIS AND FATIGUE

The procedure of design may be viewed as a three-link chain: loads-stresses-allowables. The subject of load statistics will not be discussed here, because a number of papers in this Symposium deal in detail with this problem. We have taken a look at the flaws in the last link - fatigue allowables. Let us now look at the central link: stress analysis, the procedure of computing the stresses in the structure produced by the specified or chosen loads, again focusing attention mainly on weaknesses of current practice.

Levels of stress analysis. - Not too many years ago, the attitude towards stress analysis was often the one summarized by the saying: "A good stress man knows P/A, Mc/I, $T/2At$ and nothing else." This attitude that very elementary theory is adequate for practical purposes has some foundation in fact: for static-strength design with reasonably ductile materials, the elementary theories often do give very good results. However, as the ductility of materials has decreased through the years, the elementary theories have become less and less adequate even for static-strength design, and for fatigue design, they are more often than not completely inadequate.

In some quarters, there appears to be a belief that elementary theories can be up-dated for application to fatigue design by adding stress concentration factors developed for simple stress raisers such as a hole. These factors must, indeed, be applied. However, they must be applied to the local stress in the vicinity of the hole - a region, say, a few inches square. The elementary theories do not give adequate accuracy over such small regions. If the stress distribution on the cross section of a wing is considered, for instance, then the stresses given by elementary theories can only be regarded as averages over regions 5-10 feet square, which is grossly inadequate.

More advanced theories which give adequate accuracy over regions about one foot square have been available for a long time. They were not practical without automatic computing machines; since such machines have become available, these theories have been - or at least are being - introduced into practice. The introduction of these methods constitutes a very long step towards more adequate stress analysis; however, severe limitations still exist.

A complete structure such as a wing or a fuselage with cutouts cannot be handled in one step without simplifying assumptions. For conventional - not to say old-fashioned-configurations, it is reasonably well known what assumptions are permissible. For others, however, - for instance, delta-wings - very little is known about what simplifying assumptions are permissible.

Theories of the type just discussed - with adequate accuracy over one-foot regions - should be adequate for fatigue design (in conjunction with stress-concentration factors) for a large portion of the structure. Unfortunately, they are still inadequate in the vicinity of severe discontinuities, such as corners of cutouts, where many severe fatigue troubles occur. The difficulty is chiefly -

although not entirely – the difficulty in numerical mathematics of having to use a much finer grid. Since the best computing machines now existing are marginal even when a coarse grid is used, a brute-force approach is impossible. It will be necessary to employ either specialists in computer-mathematics or specialists in structural theory, or some combination. In other words, the introduction of automatic computing will not decrease the need for specialists in structural analysis.

An attempt to summarize this discussion is shown in Figure 9, which shows the cover stresses in a box beam as indicated by strain gages of different gage length, representative of the different levels of theory. Figure 9(a) shows the picture seen by a gage one fathom long (a rather unusual measure, for this purpose, but the only length measure available in the 5-10 foot range); this corresponds to the Mc/I theory. Figure 9(b), the foot-gage, represents the theories now being incorporated into automatic computing procedures. Figure 9(c), the inch-gage, represents the picture we hope to get some day from machine computations. Figure 9(d), finally, represents the effects of stress raisers (rivet holes) added to the previous picture.

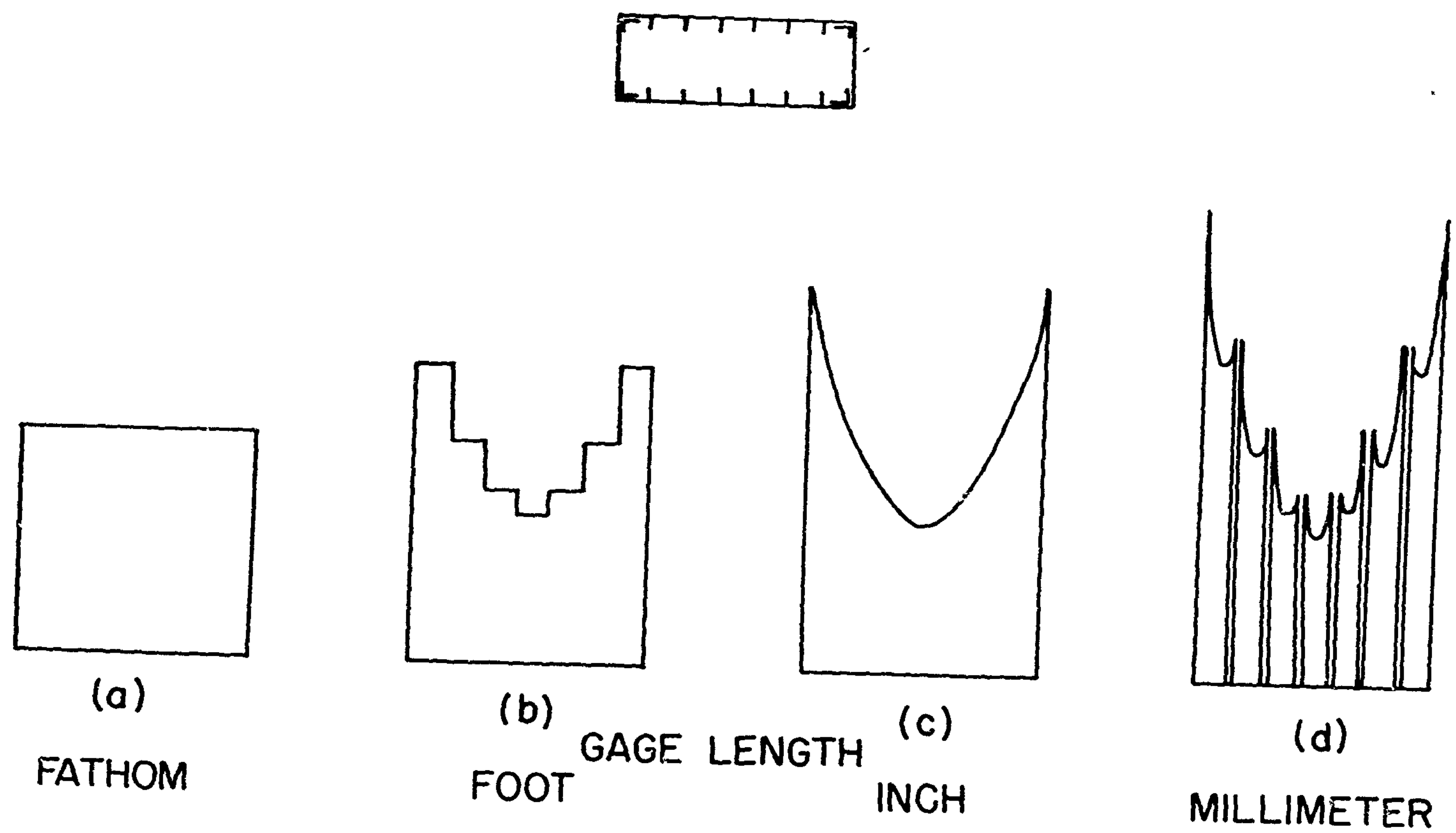
The question of stress over large areas versus stress over small areas came in for considerable discussion in the court inquiry into the Comet accidents in England. Other discussions in this inquiry dealt with the validity of tests on parts of the complete structure, with non-representative boundary conditions. The final report on this inquiry is remarkably clear and concise, and should be required reading for everybody even remotely concerned with the structural design of aircraft (Ref. 10).

<u>Relations between stress analysis, fatigue analysis, and tests.</u> – In many cases, it may be a moot question whether test failures should be attributed to inadequate fatigue knowledge or to inadequate stress analysis. However, stress analysis is most likely to be blamed in the following very typical cases:

a) A large number of failures at cut outs.
b) Heavy concentration of cracks near root of rear spar of a swept wing.
c) Major failure in totally unexpected place.
d) Large number of early failures in a structure from conventional material and with conventional over-all configuration.

The fatigue test on a complete structure is conducted in the expectation of obtaining at least semi-quantitative information on the life of the structure. If stress analysis has missed the critical area, the complete test article may be lost before this information is obtained. A heavy cost is then incurred in procuring a new test article, and probably worse, considerable time is lost for development. At the present state of the art of stress analysis, gross mis-predictions are possible due either to over-sights or to bad simplifying assumptions. A test structure makes no assumptions and commits no over-sights in reacting to the applied loads. These features will make tests on the complete structure highly desirable for the foreseeable future. Tests on complex structures, on the other hand, are quite useless as guides for more efficient design, particularly if novel configurations are used. Progress in the efficiency of structures can be achieved only by analysis.

COVER STRESSES IN BOX BEAMS



NASA FIGURE 9 KUHN

Since test specimens of complete structures are expensive, it is desirable to make maximum use of them. Strain gages should be used as much as possible. As pointed out before, these tests should be analyzed with a view toward providing knowledge for better future design; they should not be filed and forgotten, to be disinterred only in case of service trouble.

DESIGN PROCEDURES

The procedure of designing the tension side of a shell structure for static strength consists, in principle, simply in adjusting the cross-sections until the stress produced by the loads is slightly less than the allowable value (tensile ultimate). Until about fifteen years ago, this procedure resulted in a design which had also a satisfactory fatigue life. Now, this is no longer true.

The design developments which led to this change were rather gradual changes. Moreover, to produce a certain percentage change in life, only a much smaller percentage change in stress is required. Consequently, a first approximation to a satisfactory fatigue design can be obtained by using the static design procedure, but reducing the allowable stresses.

The allowable fatigue stress for a given case can be determined by comparative calculations, by comparative tests, by special tests, or by a combination of these. Since all fatigue failures are caused by design details which cause stress concentrations, a design standard may be set, expressed by an equivalent K-factor. An important part of the design procedure then becomes the process of scrutinizing all design details and testing all questionable ones to insure no detail is worse than the standard set. Obviously, the procedure can- and should-be refined by using different allowables for areas in which all details are either better or worse than the standard. Great refinement of this type, however, is not justified at the present state of the art.

PREDICTION OF BEHAVIOR OF STRUCTURES

The purpose of fatigue design is to achieve a structure which has a satisfactory length of life, an adequate degree of safety, and a tolerable level of repair and replacement needs.

For a fail-safe airplane, the life of interest is essentially the mean life of the fleet, since this determines the time at which the fleet should be phased out of service. An appraisal of the factors influencing the determination of the mean life suggests that the largest uncertainty in the early design stage is probably the translation of loads into stresses. The fatigue test on the complete structure is performed to reduce this uncertainty. Since only one test is likely to be made, not much confidence can be placed in the numerical results. However, the qualitative results - the location of the trouble spots - can be followed up with strain gages, fatigue analysis and special tests to obtain better numerical results. Thus, it does not appear to be unduly optimistic to feel that in the near future, the prediction of the mean life of the fleet should become reasonably reliable on structures which do not depart too radically from the realm of past experience.

The picture looks much less promising when one considers the question of predicting when the first airplane of a fleet will start having fatigue trouble.

Predictions of this nature require much more detailed statistical knowledge than predictions for the mean of the fleet. Again, the question of load statistics will not be discussed here because it is dealt with by other speakers.

On the structural side, one needs first of all adequate statistical data on the fatigue behavior of material - which might be obtained, at a large effort. In addition, however, one would need statistical data which, in effect, define the scatter in knowledge and skill of the design staff and of the shop staff. Partial statistics of this type might be obtained by special research projects. In the main, however, the statistics would have to be developed by continuous analyses of the performance of design and shop staffs, and this would require a cooperative effort of unprecedented difficulty.

The prediction of the first appearance of trouble in a fleet is of critical importance only when no fail-safe features have been incorporated. Recognition of the facts regarding the lack of appropriate statistics therefore leads inevitably to the conclusion that fail-safe features should be incorporated as much as possible in order to minimize the need for these statistics.

The fail-safe design of complex shell structures is a rather new line of engineering. Fairly extensive investigations have been made for one specific problem: the pressure cabin. This problem was investigated, of course, under the spur of specific accidents, but a reasonably systematic investigation was made possible by two fortunate facts: the basic structure is simple, and a simple loading case dominates the design. In wing structures, a much larger variety of structural arrangements is possible, as well as a larger variety of loading cases; systematic investigation is therefore much more difficult, and even the guiding principles are much less understood.

A main principle of fail-safe design is the provision of alternate load paths. To be unequivocally effective, the alternate paths must be so completely independent of each other that a crack in one path is not propagated into the alternate path. Such complete independence exists beyond question only if the members in question are separated by air gaps, and the end fittings are such that failure of one member does not substantially alter the stress distribution in the remaining member. But completely separate members are impractical, if not impossible, in a shell structure. In general, then, alternate paths are only partially independent, and tests are necessary to demonstrate that the required degree of independence exists. The number of such tests on a complete structure is severely limited by the fact that a patched-up structure is not representative of the design; consequently, components of adequate size to represent the key features must be tested under an adequate variety of conditions.

Since component tests incorporate interactions only to the extent that fallible engineering judgment foresees them, the true degree to which a complete structure is "fail-safe" may still be doubtful sometimes. In such cases, the inspection schedule in service should start earlier and be more comprehensive than for structures which are beyond question highly fail-safe.

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THE STATISTICAL BASIS OF LOADING SPECTRA FOR FATIGUE DESIGN CRITERIA

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ABSTRACT

This paper reviews and discusses the requirements for and problems of formulating a rational but practical fatigue design criteria. Background information for fatigue design criteria is presented. The four elements needed to establish criteria are presented. They are: (1) Parameters required for loading spectra; (2) Sources for spectra parameters; (3) Synthesis of spectra for new systems; and (4) Formulation of fatigue design criteria. It is postulated that literally there is no such thing as a fatigue failure. Generalized aircraft spectra are not sufficient to determine fatigue failure rate, so structural component and element spectra must be derived. This spectra must define two conditions: the past loading history of the element and the present probability of exceeding a given load. These spectra of MIL-A-8866 are inadequate for determination of the required element loading spectra.

INTRODUCTION

The basic objective of structural design criteria is to formulate requirements for a structure which will reliably attain a specified operational life. Quantitative definition of how reliably what life is to be attained is a very difficult procedure. Implementation of the requirements in a structure, with confidence that the requirements will, in truth, be met, is an extremely challenging task.

EVOLUTION OF PRESENT STRUCTURAL DESIGN CRITERIA

In the past, structural design criteria did not explicitly state the quantitative requirements for reliability and life; instead, the typical criteria defined loading conditions, which, if met, resulted in extremely high structural reliability over the lifetime of all aircraft designed to that criteria. The criteria was the end result of past experience. Any failures, other than occasional random failure, resulted in criteria modification. New developments such as abrupt checked maneuvers and inertial coupling conditions, were added as necessity dictated and outmoded conditions were gradually removed. This empirical approach to structural design criteria has, in general, proved sufficient until recently for the needs of designers and the service administrators of structural design policy. However, certain anomalies have always been present which resulted in some airplanes being overdesigned and others underdesigned.

NEED FOR PROBABALISTIC CRITERIA

Recent rapid advances in weapon system design have resulted in radical changes in structural design requirements, so much so that a change in criteria philosophy appears to be an obvious necessity. To this author, a probabalistic approach to design criteria is the only rational concept. The big difficulty is to find a practicable solution to the problems that arise while reducing theory to practice.

At first blush, use of a probabalistic philosophy is a radical departure from the past with the revolutionary changes in design concepts currently in progress justifying corresponding revolutionary changes in criteria concepts. However, the change in criteria philosophy is more apparent than real. Structural reliability or probability of failure has always been implicit in previous regulations. The qualitative probability input to criteria formation has been either success in performing the desired mission or what someone's judgement indicated to be "too many" failures. Furthermore, fatigue has always been a consideration in past criteria - occasionally explicitly, usually implicitly. Examples are fitting factors; knowledge of what constitutes good detail design in matters of sharp corners, scratches, avoidance of brittle material, etc; and designing to normal operating stresses below the endurance limit.

ADVENTITIOUS AND FATIGUE FAILURES

Furthermore, there is a little realized but close relationship between the adventitious, or discrete, failure and the fatigue failure. Adventitious failures are the end point of the range of fatigue failures possible from the millions of cycles of low amplitude loading to the few cycles of extremely high amplitude loading. Also, "fatigue" failure in any structure always occurs from a single application of some load. This load may be a relatively low load - the same magnitude as has been continuously applied for millions of cycles - but it nevertheless is a discrete load that finally administers the coup de grace to the structure. In

this case, most of the original strength of the material has been lost due to fatigue deterioration. Similarly, the final failing load may be a relatively large load that occurs after only a small amount of fatigue damage has been inflicted on the structure. Finally, and obviously the end point of this sequence of possibilities, failure may come from a single application of very high load before any fatigue damage has been done. Thus, it is apparent that there is no fundamental difference between adventitious and fatigue failures. The one blends into the other with no line of demarcation being evident. It is a rather startling concept; but, in fact, it can be said that there is no such thing as a fatigue failure. There are only adventitious failures at different magnitudes of loading, the magnitude of the failing load being a function of the loading history (both magnitude and sequence) and the consequent amount of fatigue damage incurred.

NEED FOR CHANGE IN CRITERIA EMPHASIS

From the foregoing, it is evident that past structural design criteria established adequate qualitative coverage for a high degree of structural reliability for both adventitious and "fatigue" modes of failure. Adventitious failures were held to a minimum by specifying limit and ultimate conditions that rarely occurred. Fatigue failures were limited by indirect methods as described previously. Both modes of failure were guarded against by implicit methods which resulted qualitatively in the desired structural reliability without ever quantitizing either the reliability level or the operational life.

Past criteria emphasized the prevention of adventitious failures while only obliquely treating the problem of fatigue failure. The situation with which designers are now faced requires no new concepts. The old tried and true concepts are completely adequate. All that is needed are methods to rationalize the criteria objectives by quantitizing structural reliability for a given life and by changing emphasis from the adventitious failure mode to the fatigue mode.

INITIAL CONCLUSIONS

Rationalizing and quantitizing the criteria may require initially a mixture of rational and empirical requirements until all the data and analytical techniques necessary for the completely rational approach are available.

ESTABLISHING STATISTICAL CRITERIA

With the foregoing providing a suitable background for studying the problem of providing a statistical basis for fatigue design criteria, it is now appropriate to examine the rationale for such criteria.

Establishing a fatigue design criteria is a fantastically complex undertaking if a thorough job is to be done. The undertaking will be approached as follows:

(1) Determine the parameters required to establish the required loading spectra.

(2) Investigate the sources for establishing such parameters quantitatively.

(3) Develop analytical techniques for synthesizing the loading spectra for any new aircraft system from the parameters established for past systems or from a rational analysis of the new mission requirements.

(4) Formulate structural design criteria embodying the principles established in the preceding steps.

PARAMETERS REQUIRED FOR LOADING SPECTRA

At present, there is no complete study defining the parameters needed to establish the required loading spectra. Much work has been accomplished in this field but much remains. A few pertinent observations can be made.

Fatigue is a condition which occurs at localized points on the structure. It is, therefore, a function of the loading history and the material at that point. Determination of the load at a point requires knowledge of the complex distribution of loads on an aircraft for a given set of aircraft parameters. Performing these calculations for a few discrete conditions as required by present criteria is a major undertaking. To do it for multiple combinations of all the parameters defining the flight conditions of the aircraft is a colossal endeavor.

At this point in the discussion, it seems proper to review the parameters that define the magnitude and distribution of loads on an aircraft. In 1954, a Special Statistical Maneuver Loads Panel of the Loads Subcommittee (ref. 1), recommended that statistical data on eight basic aircraft parameters be obtained from operational usage. These eight parameters were the three linear accelerations, the three angular accelerations, airspeed and altitude. With these parameters known, the motion of the airplane and the aerodynamic conditions determining the distribution of load are completely defined. It is in the statistical definition of these eight parameters that difficulty is encountered. Many of the statistical functions of these parameters are themselves functions of some or all of the other seven parameters. This means the probability functions are multi-dimensional and, as a result, extremely difficult to define and manipulate in probability calculations.

To define the load on a specific structural component on the aft fuselage, it is generally necessary to know (1) the horizontal and (2) the vertical tail load plus (3) the inertial loads for the particular flight condition being investigated. Each of these three loads is a function of the eight basic parameters defined previously plus other derived parameters such as the angular velocities, Mach number, and airplane lift coefficient. Thus, each of the three loads is a complicated function of many parameters and the load on the fuselage component is the sum of the three complicated functions. The complication inherent in the calculation of the single load is compounded when the spectrum of this load is desired and it must be derived from knowledge of the statistics of each of the parameters entering into the calculation of the desired load. Some idea of the scope of the problem can be obtained from the work by Bouton and Scrooc (ref. 2) for a two element structure and by Bouton and Shirley (ref. 3) for a 16-element structure.

The immensity of the task is compounded again when the spectrum of the loading at a single point is expanded to include the thousands of points all over the aircraft. Spectra for these thousands of points are needed because each of these points must be individually investigated to determine the fatigue failure rate at each point and from this the airplane failure rate and structural reliability. This, then, completes the picture of what parameters are needed to define the loading spectra for fatigue analysis for an aircraft.

SOURCES FOR SPECTRA PARAMETERS

The data sources for establishing the above-mentioned parameters are many and varied. Past work on the subject is well known to aeronautical engineers and will not be considered at length here except to mention a few. Work such as Doolittle's (ref. 4), Rhode's (ref. 5), Donely's (ref. 6), Reynold's (ref. 7), Gray's (ref. 8), Press and Mazelsky's (ref. 9), Mayer and Hamer's (ref. 10), Hamer, Huss, and Mayer's (ref. 11), Sissenwine's (ref. 12), Barrett's (ref. 13), Clementson's (ref. 14), and others have contributed much to the store of statistical knowledge.

Despite all these efforts in the past, we simply do not have enough information to design an aircraft on a statistical basis. From the best information available there is no indication that any operational airplane has used statistical data as the sole basis for structural design. Another indication of this is the action taken by the NACA Loads Subcommittee in 1954. A Special Panel was established to recommend to the Subcommittee what action should be taken regarding statistical maneuver loads. Included in the recommendations of the Panel, which were endorsed by the full Subcommittee, was one that a joint Military-NACA program be established to measure statistically eight parameters chosen to establish the motion of the aircraft and the aerodynamic conditions during the motion. These eight parameters were the three angular accelerations, the three linear accelerations, airspeed and altitude. Also, the panel recommended that NACA do more research on fundamental criteria problems.

This eight-channel program gathered momentum slowly in the years following until the recent impetus, given by the fatigue failures described in previous papers, necessitated quick action on the subject. As a result money became available to implement the program. An eight-channel recorder was brought to the prototype stage and the remainder of the program, originally suggested by the Special Panel, is currently being implemented. The reduction of the eight-channel data to desired form is being implemented as a joint Air Force-Navy-NASA program. A facility with a large digital computer and necessary analog equipment will be put under contract in the near future. This facility will be operated under cognizance of the Bureau of Aeronautics. It is expected to require full time operation simply to keep up with the data received from the 240 Air Force eight-channel recorders and the 50 Navy recorders. This information will all go into a common pool and will be accumulated as appropriate to class of aircraft, mission, etc. Some idea of the magnitude of the job is that the present data reduction program established over 13,000 separate combinations of flight conditions that will be recorded and accumulated for statistical analysis.

Norair was awarded a contract by WADC to review the eight-channel program already established, recommend improvements and determine the ultimate objective of a structural criteria. This study is principally directed towards maneuver loads. However, the end product should be one in which both maneuver and gust loadings can be fitted together in determining the loading spectrum for any given type of aircraft. There are many other studies in the total WADC program. One of major interest, in connection with the subject of this paper, is the program by Douglas to review all of the past data and determine applicable methods for converting this data into loading spectra.

In addition to the eight-channel program, the Air Force is instrumenting 2400 airplanes for a VGH program set up explicitly to obtain fatigue information on gusts and landing conditions. Additional data has been obtained from airplanes

such as the B-47's and FJ-4's. Furthermore, NASA is currently in the final stage of a report on the F-84, giving a large amount of statistical data. In addition, F-100 data is being worked up on the basis of the information that will be available from the eight-channel program. This analysis is to show what kind of information can be derived from the eight-channel recorder program.

In recent years it has become apparent that information on low altitude gusts is necessary for military operations and various procedures have been followed to obtain this data. Norair had a contract to measure low altitude gusts with the F-89 and published this data in reference 13. Subsequent to this, WADC contracted with Douglas to obtain low-altitude gust data on the RB-66. This operation is currently in progress. Gust analysis has become, almost universally, one of establishing the power-spectral density of the atmosphere. This stems from work done by Press and Mazelsky (ref. 9), Clementson (ref. 14), Bendat (ref. 15), and others.

There are many other sources of statistical loading data for aircraft, but time prohibits noting them all. Unfortunately, much of the early statistical data is not in a form amenable for use in establishing fatigue spectra. The reduction of all of the statistical data that has been and is being generated into maneuver, gust, and ground loading spectra is a big enough problem to warrant a paper on this subject alone.

SYNTHESIS OF SPECTRA FOR NEW SYSTEMS

In the Introduction it was stated that "The basic objective of structural design criteria is to formulate requirements for a structure that will reliably attain a specified operational life." In order to formulate these requirements, how the spectrum should be defined in order to accomplish what purpose must be known. It follows that a rational loading spectrum is one which, if designed for in the structure proper, will result in a structure which has the required lifetime. The determination of this spectrum is, on the surface, relatively easy, but in practice it will be extremely difficult.

It seems to be a general belief that a simple spectrum consisting of frequency of attaining a load factor as a function of the load factor is all that is needed to define the fatigue environment for an aircraft structure. This is simply not the case. Fatigue is a condition which occurs at a point in the structure and, therefore, must be the result of the lifetime loading history at that point. It is a truism that no structural component of any aircraft has ever failed in fatigue because the frequency of occurrence of various center of gravity load factors has been excessive. There is no way to determine the fatigue life of a composite structure except by determining the fatigue life of the basic structural elements of that composite structure. This requires knowledge of the loading spectrum of each of those basic elements.

Loading spectrum is used here with a much less restrictive meaning than is common. "Loading Spectrum" is assumed to mean not only the frequency of occurrence of a given load; but, also, the type of load-tension, compression, shear, etc., the order of occurrence of these loads, and the probability of encountering a different frequency of occurrence. This definition of loading spectrum and an understanding of the fatigue process in aircraft leads to another interesting observation: In order to determine the fatigue life of a structural element, it is necessary (1) to define the loading history of that element at any time during its life, and (2) to

determine the probability of encountering loads of various magnitudes at that particular time in the element's history. The statement above contains the inherent assumption that fatigue life is defined - not as a number of hours that a structural element will survive, period, but - as the probability that the element will survive any given number of hours.

With these ground rules understood to be established, the analytical techniques necessary to define the loading spectrum for an aircraft structural element may be explored. First, it should be said that designers, analysts, and those responsible for establishing criteria do not fully appreciate the immensity of the task of specifying and being able to develop the loading spectrum for each of the myriad structural elements. This spectrum is the end result of a long chain of calculations starting with the determination of the spectra of various aircraft parameters. These airplane parameter spectra, in turn, must be derived from definition of the mission of the aircraft.

The starting point of a fatigue design criteria is obviously an abundance of statistics. It is assumed the sources for spectra parameters, discussed in a previous section, plus other new sources of data, will continue to develop statistical data of the type described in the section on parameters required for loading spectra. Someday there will be sufficient data available to meet the requirements of the analysis described in this paper. When that day arrives, it is hoped that the capability of the analyst to utilize such data has increased commensurately. To date, there have been many researchers in this field of fatigue design criteria, but each one has tackled only a small portion of the total problem.

Mayer and Hamer (ref. 10) have considered the problem of rationalizing the load spectrum from knowledge of the mission of the aircraft and the utilization of the aircraft in various phases of that mission. Bouton and Scrooc (ref. 2), and Mayer, Stone and Hamer (ref. 16) have examined the problem of deriving, statistically, major component loads. Bouton and Shirley (ref. 3), and Hilton and Feigen (ref. 17) have looked into some aspects of the loading spectrum for structural elements. References 2, 3, 17, English in reference 18, Freudenthal in reference 19, Pugsley in reference 20, and van der Neut in reference 21 all analyze the failure rate of components and from this derive the total failure rate of the aircraft. Some of these references included the effects of scatter in strength properties while some did not. Reference 16 takes up the question of the effect of confidence limits on the load spectra. Reference 19 studies the probability of failure under fatigue conditions.

Each of the reports mentioned above only begins the investigation necessary before the techniques described can become routine tools of the analyst. No one has performed the task of integrating all these partial analyses into a complete analysis of structural reliability under fatigue failure conditions. Until such a study is undertaken and all the requirements for loading spectra defined, there will be some uncertainty as to whether the present efforts are progressing in the proper direction. Thermal effects on structural reliability are considered to be beyond the scope of the present paper.

At this time, it is of interest to consider the spectra presented in the proposed fatigue design criteria, MIL-A-8866. The -8866 spectra represent a considerable advance in the problem of designing rationally to avoid fatigue failures. However, these spectra and the other requirements of -8866 are not sufficient to obtain the freedom from fatigue failures that is desired. Consider the case of a

population of single elements, each of which has a 50 percent probability of survival for a given time. Under these circumstances, approximately 63 percent of the elements will have failed at the end of the given time. By analogy this can be extended to the failure rate of aircraft designed for a specified number of flight hours and the load factor spectrum of -8866. Since the -8866 spectra represent averages, it might be expected that 63 percent of the aircraft designed to MIL-A-8866 will fail sometime before the specified number of hours have elapsed. This assumes the fatigue analysis is precise and that critical components are not replaced. The average time to fail may not be much less than the specified time, so the resulting effects on fleet cost, combat readiness, spares requirements, and model life span may be small. Nevertheless, the average life will be less than the specified life, even with perfect analysis techniques.

To summarize, the requirements for synthesizing loading spectra for new systems and the problems involved in calculating such spectra appear to be overwhelming.

FORMULATION OF FATIGUE DESIGN CRITERIA

The objective of a fatigue design criteria is to provide an aircraft structure which will not fail, except in minor, inconsequential, easily repairable ways, during the lifetime of the aircraft which performs its specified mission. Regardless of whether or not the state of the art can provide such capability, the objective above should be the stated intent of the criteria. If, as, and when the capability is developed to synthesize loading spectra and to analyze accurately the fatigue strength and reliability of the structure, then fatigue design criteria can be rationally specified. Until that time, certain portions of the problem can be defined well enough so that the designer and analyst can eliminate the large majority of fatigue failures.

Because of the present state-of-the-art, the actual specification of fatigue design criteria should be stated in as general terms as possible with specific methods for accomplishing the stated objectives left to the designer. It is suggested that the specification contain only the basic requirements such as desired life, quantitative definition of structural reliability, policy decisions on matters of the relation between structural reliability and total system reliability and performance, need for establishing secondary missions, etc. A fundamental decision must be made on whether there shall be one requirement for average life and another requirement for safe life. Further, it is suggested that a Criteria Manual be published which discusses and, as appropriate, presents the basic numerical data required to synthesize loading spectra and acceptable methods for performing the fatigue analysis. Criteria which results from considerations of optimizing the structure have no place in the criteria specification, but optimization procedures could logically be included in the Manual.

It should be understood that, no matter how carefully the fatigue design criteria is phrased, the state-of-the-art is not such that all fatigue failures can be eliminated. Elimination of the majority of fatigue failures is a more realistic objective. If the criteria is complicated too much in the vain attempt for perfection, losses rather than gains may result in the fight against fatigue failures.

Above all, the fatigue design criteria should not take form such that tomorrow's aircraft are being designed to overcome yesterday's problems.

SUMMARY

The following summarizes the major subjects discussed in this paper:

1. Previous specifications for structural design criteria presented fatigue requirements qualitatively and implicitly. Quantitative and explicit requirements are needed now.

2. Elimination of the majority of fatigue failures is a reasonable objective for fatigue design criteria, but complete elimination is not.

3. The relationship between adventitious and fatigue failures is discussed. It is noted that, literally, there is no such thing as a fatigue failure.

4. Synthesis of spectra is a colossal task, but it can be done.

5. There is still a need for a large-scale program to gather statistical data for loading spectra. Much more research is necessary on methods for utilizing the statistical data in the design criteria.

6. Statistical data on the eight parameters established for the present eight-channel data recording program form the minimum basis for a rational fatigue criteria.

7. The concept is established that fatigue is a localized condition so loading spectra for a myriad of structural elements must be the end product of the basic loading spectra for the complete aircraft.

8. The loading spectra really must establish two functions. First, the past history of an aircraft at any time in its life must be determinable statistically in order to determine the probable strength level (fatigue damage) of the structural elements at all times during the lifetime of the aircraft. Second, the probability of encountering a given magnitude load at any given time in the history of the component must be known in order to compute the probability of failure at any given time. The aircraft probability of failure and its inverse, structural reliability, are the sum total of all the probabilities of failure during the previous history of the aircraft.

9. Many decisions are needed regarding fundamental questions of design objectives.

10. It is suggested that a Fatigue Criteria Manual be established to present numerical data on the various phases of aircraft utilization and acceptable procedures for synthesizing the loading spectra.

11. Fatigue analysis with the loading spectra of MIL-A-8866 will necessarily be imperfect. Component and elemental load spectra cannot be determined with only load factor spectra defined. The average life resulting from use of the present spectra will result in fatigue failures at some time less than that specified.

12. Some new concepts are introduced and some old ones restated. Some of these concepts may be controversial. In any event, it is hoped that presentation of these concepts will stimulate rational thinking on the subject.

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SOME FATIGUE DESIGN REQUIREMENTS
FOR
FUTURE AIR AND SPACE VEHICLES

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ABSTRACT

Past and present fatigue design requirements for airframe structures are briefly reviewed. The present evaluation approaches for satisfying these design requirements, to assure safe and reliable structures, are outlined and discussed. Future fatigue design requirements are proposed with particular emphasis on the considerations of airframes operating at high temperature. Speculations are made only as to the influence some space environments will have upon fatigue characteristics of structural elements. The closing section presents a few numerical examples in evaluating the useful life of present and near-future vehicles.

INTRODUCTION

The fatigue design philosophy of past and present aircraft structures is presented. Initially, the fatigue life of elements of the basic structure of a vehicle are evaluated by means of comparative and/or spectrum testing. Joints, subassemblies, and major components are tested and compared with these basic elements. In some cases, the complete airframe is tested for final design confirmation.

This kind of evaluation has resulted in the achievement of the present day fatigue design rules and requirements. By taking full advantage of available theoretical knowledge and practical experience, today's designer can now create efficient aircraft structures which are both safe and reliable. These requirements are appropriate whether in regard to commercial operation or in regard to effective military use.

It is believed that a successful transition can now be made for the fatigue design of the air and space vehicles of the future. However, in preparing the criteria for the structural fatigue requirements of future air and space vehicles, it will be essential to employ the information gained from the design, construction, and operation of vehicles that are already in service. Undoubtedly, there will be certain transitional and environmental changes from the present to the future designs that will expand the fatigue problem. Nevertheless, the basic design rules for structurally efficient members will be similar whether the vehicle mission be intercontinental, to the fringe-of-space, or that of a re-entry satellite.

Although it is not possible to present in this paper the fatigue test experience and evaluation of a full-scale future space vehicle, it is believed that it may be possible to offer an outline and approach to the problem that may not be generally familiar. Of particular importance, for future research and testing at elevated temperatures, is the manner in which the tests are to be conducted. This point is stressed in the paper. It will be shown by way of numerical examples how the fatigue evaluation of a few hypothetical, near future, vehicles might be made. Speculations about far-future vehicles are made only as to their environmental conditions for the purpose of further stimulating such considerations.

GENERAL FATIGUE DESIGN REQUIREMENTS FOR AIRCRAFT

In both past and current design philosophies the most important factors found to affect structural safety and service life are material behavior, fabricating conditions, stress concentrations, operating environment, and working stress levels. Some of these items have not been reliably accounted for by accepted procedures of stress analysis and test. Improved methods of design evaluation which properly accounts for these factors will always be needed. The need for a reliable and comprehensive measure of design quality can be further emphasized by some service experiences. For instance, it is known that airframe structures may be designed at the same experience level, to the same standards, and with the same objectives, but the actual performance of many individual designs will vary widely, even for the same kind of vehicle.

The primary and basic objective to be attained is that all parts have a life expectancy that exceeds the useful life of the airplane. Most parts, even though they are highly stressed, highly efficient designs, do this very adequately indeed; while some fail after short periods of service. Basic continuous structure, for example, has performed satisfactorily on many airplanes with service times ranging from

20,000 to 60,000 hours. In other cases, failures in detail parts have occurred within service periods as short as 1,000 to 10,000 hours. The design of structures requires a combination of theoretical work and practical knowledge gained through service experience and laboratory testing. Extensive and important research efforts to provide basic data necessary for the design of structures subjected to dynamic loading are being carried out in two major fields; the fatigue properties of materials, and the load environment to which airframes are exposed.

Probably because of the inadequate performance of some structures in service, research activities have been extended beyond the work on basic data to include the evaluation of simple and complex structural designs and the estimation of the life of aircraft structures in service. Such estimates of structure life are based on assumptions of design quality expressed in terms of theoretical stress concentration factors (K_T), or experimental factors (K_f), and are limited, therefore, to the characteristics of particular designs. In addition, fatigue life calculations are subject to large inaccuracies because of unknowns, inexact simplifying assumptions, (which may be quite adequate for static analysis) and variations in behavior resulting from scatter, test technique, and other causes. As a result, estimates of fatigue life can easily be in error by a factor of ten or more. This order of scatter is not necessarily experienced, however, for carefully conducted tests. Various types of fatigue tests, such as single load level, programmed load and frequency, and the quasi-random applied load and frequency type have shown test scatter no greater than 1.0 to 2.5. In the majority of simple element tests, the test scatter has been in the order of 1.0 to 1.8.

Methods of treating scatter, as far as test evaluations are concerned, can be simply and reliably handled in a variety of ways. One of the more common methods is the Least-of-Four principle, (reference 1). An alternate method, based on a preselected probability level, is as that shown in Figure 1. Figure 2 indicates the service fatigue cracking of a fail-safe design component after high time exposure. It should be noted that for the rather limited load range of vehicles of the commercial category the scatter in service experience may be rather low; 1.0 to 1.4, for example. This very brief chronology of the fatigue design developments that have transpired in the past few years is given as a prologue to the design philosophy that is constantly developing and improving the rather simple yet adequate rules and requirements. Prior to a discussion of the future fatigue design requirements, a presentation will be made of the more familiar and presently used evaluation methods. These methods are discussed in the following section.

In summary, the consideration of reliability and safety of the airplane for its crew and passengers is so important that design thinking should extend beyond any test or stress level criteria. The arrangement of the structure must be such that it is safe by its very nature, rather than merely strong enough.

The general fatigue design requirements for reliability of past and present aircraft structures are, (reference 2):

1. Provide optimum structural arrangements which ensure structural integrity through multiple load path, fail-safe construction.
2. Give the fullest attention to detail design, employing the latest knowledge and techniques available.
3. Provide for adequate inspection so that any cracks may be discovered by routine inspection before strength is reduced appreciably.
4. Provide for periodic replacement of critical parts where necessary.
5. Provide dual safety for pressurized windows, windshields, and canopies.

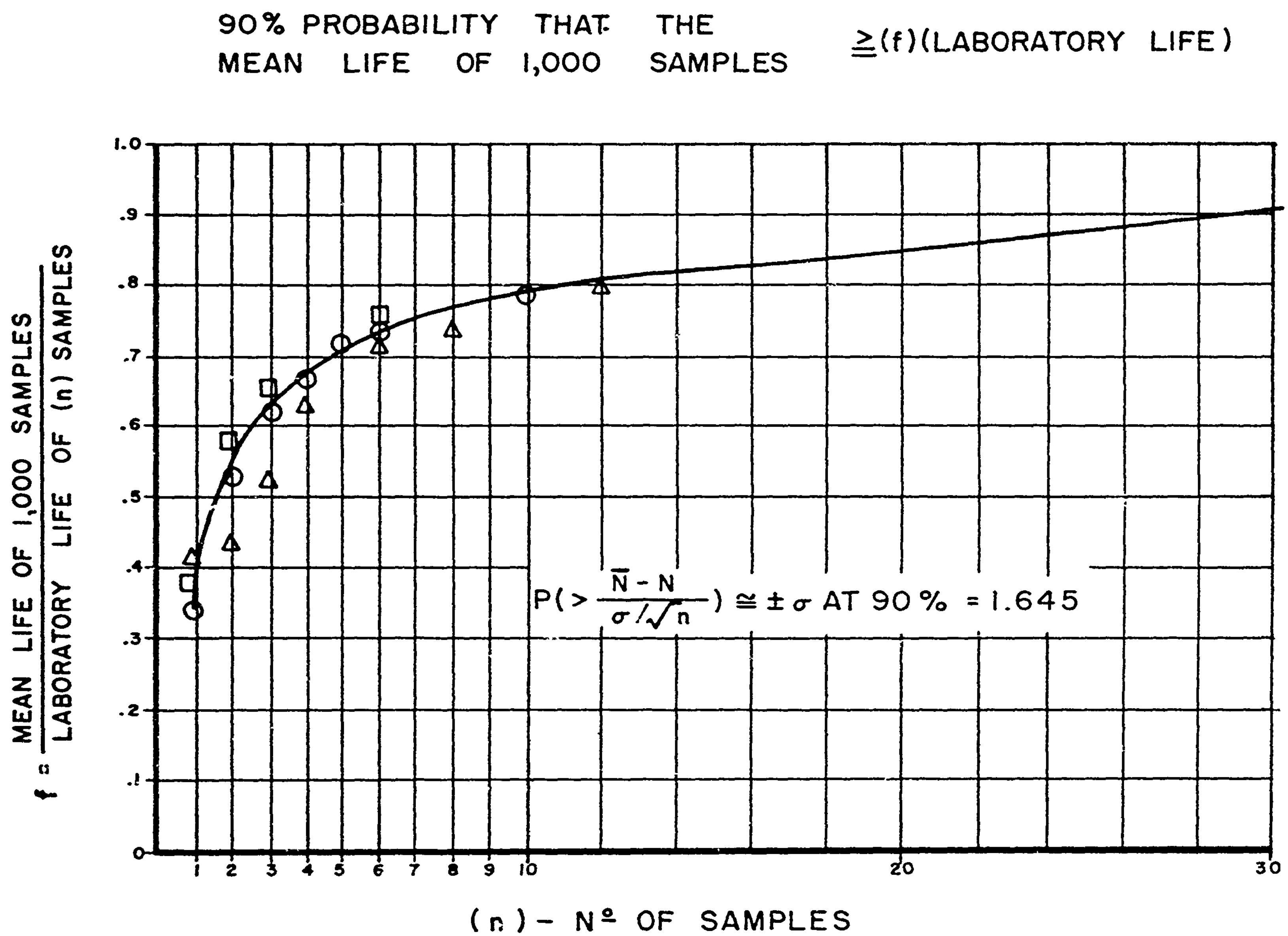

90% PROBABILITY THAT THE MEAN LIFE OF 1,000 SAMPLES $\geq$(f)(LABORATORY LIFE)
f = MEAN LIFE OF 1,000 SAMPLES / LABORATORY LIFE OF (n) SAMPLES
$P(> \frac{\overline{N} - N}{\sigma / \sqrt{n}}) \cong \pm \sigma$ AT 90% = 1.645
(n) - Nº OF SAMPLES
1.0
.9
.8
.7
.6
.5
.4
.3
.2
.1
0
1
2
3
4
5
6
7
8
9
10
20
30

FIGURE 1

SCATTER: $\frac{22}{16} \cong 1.40$ TO 1.0

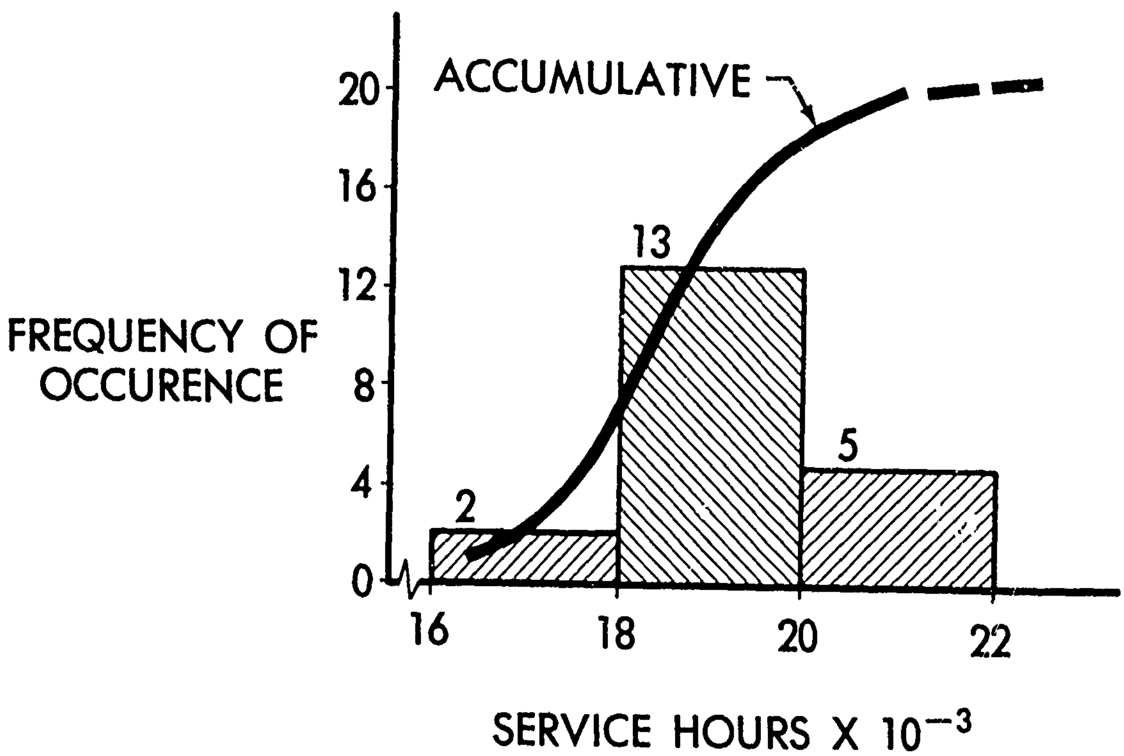


FIGURE 2

Service Fatigue Cracks in Nose Wheel Landing Gear Yokes

6. For pressurized cabins, provide designs which restrict crack propagation (including fast cracking or tearing induced by foreign object penetration) sufficiently to avoid explosive decompression at working stress levels.
7. Maintain design stresses at or below values established as satisfactory by structural service performance when evaluated on a comparable basis.
8. Provide structure which will not be susceptible to high frequency vibration in the vicinity, or in the wake of jet nozzles, propellers, or turbofans.
9. Design for, and adequately evaluate, fatigue resistance.

EVALUATION METHODS

The types of evaluation methods that presently are being used, with varying degrees of success, are outlined in the block diagram of Figure 3. These methods are proposed for fatigue evaluations and requirements of near-future vehicles.

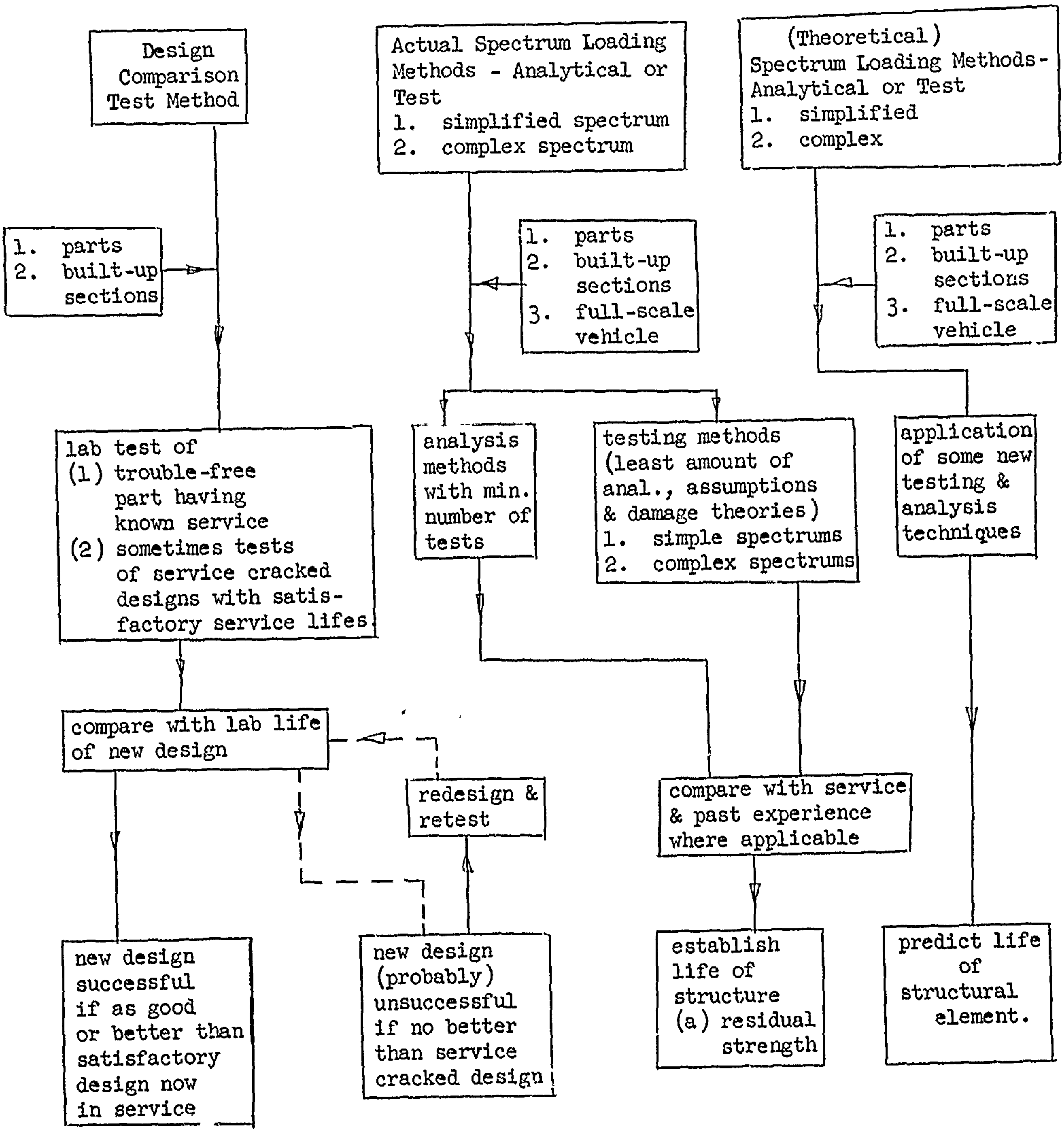
Brief descriptions of the evaluation methods are presented with illustrative and numerical examples given in the appendix.

1. The design comparison method is a technique that combines available sources of pertinent knowledge in a form such that the designer can test a part and then be able to evaluate its probable success in service, (reference 2). The principle is simply stated. A range of S-N relationships will be established employing the fatigue forces to be experienced and the fatigue strength of many parts for which the service experience is known. Designs will be compared with this range in the laboratory. This principle is based on the assumption that if a part is known to be trouble-free in service, and laboratory tests of a new design show it to be as good or better, then the new design will probably be successful in service. Conversely, if a part is known to be subject to fatigue cracking in service, and a new design is no better by test, the new design will probably be unsuccessful. This criterion, established by successful designs, can be represented graphically as shown by the envelope in Figure 4. Laboratory tests should duplicate as nearly as practical the flight loading experience. Tests should be run at a constant mean stress corresponding to the 1G stress level in the airplane with variations corresponding to damage from gust and maneuver induced loads. For some categories, loads other than gusts and maneuvers may be used to define the critical loads and comparison criterion. The application and limitations of the comparative method are discussed in the appendix 1 example.
2. Currently, it is the general belief that environmental spectrum testing is the most rational approach to adequately evaluate structural fatigue strength. Unfortunately, this type of laboratory performance testing is rather time-consuming and costly. For this reason, concerted efforts are being made to devise simplification techniques that will impose the same type and kind of damage in a much shorter period of time. Evaluation of present data indicates conflicting results in the attainment of this goal.

The spectrum testing of programmed cycles of loads taken from flight test data would appear to be a desirable technique to use. However, as it has often been pointed out, it is no more accurate than the simplifying assumptions that have been made in reducing or interpreting the actual flight records. Nevertheless, this method is proposed for future air and space vehicle structure tests whether flight loads are obtained from actual records or by analytical means. Further, it is believed that laboratory spectrum testing properly carried out will be the only valid proof of an assessment of useful structural life for future vehicles. That this is so will become apparent when the complex effect of high temperatures is introduced in combination with the loading of service usage. A numerical example of a laboratory spectrum test is given in Appendix 2.

STRUCTURAL FATIGUE STRENGTH EVALUATION

(METHODS OF APPROACH)



TYPES OF EVALUATION METHODS

Figure. 3

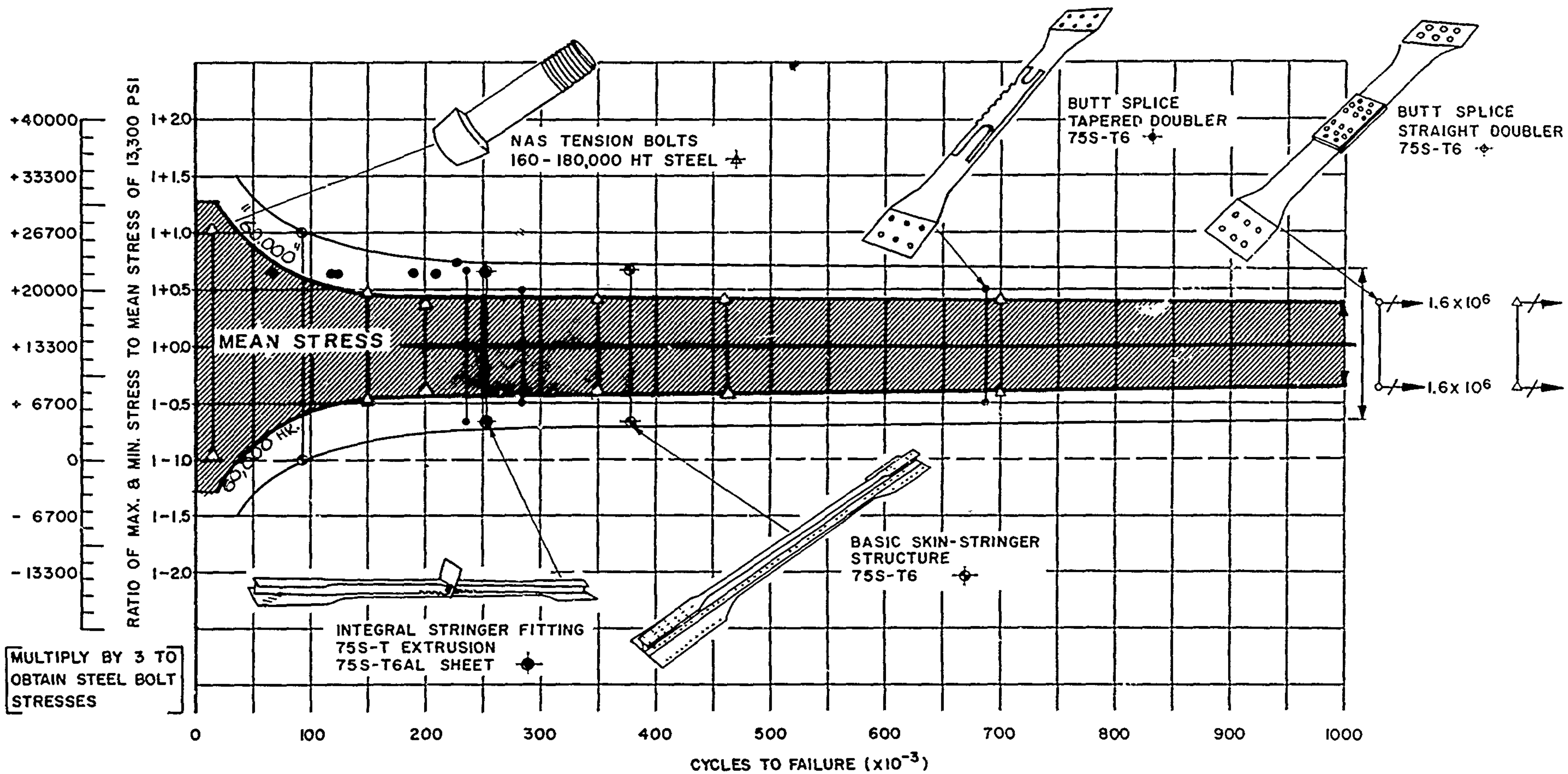


FIG. 4 S-N CURVES FOR COMPARISON OF FATIGUE LIFE OF STRUCTURE COMPONENTS

3. The third evaluation method to be considered is fundamentally the same as the spectrum method diagrammed in Figure 3, with the exception that the service loads are unknown. This method is reserved for those future design categories where the anticipated loads and environments are primarily theoretically derived. The information necessary to provide satisfactory rules for all of the parameters involved in conditions such as these is not yet available. These factors are discussed in the following sections on future fatigue design requirements.

FUTURE FATIGUE DESIGN REQUIREMENTS

It appears that ample criteria and reliable evaluation methods can be made available and that future aircraft, missile and space vehicle structures can be designed to safely withstand the anticipated loads even though fatigue failures may occasionally occur. The fatigue design criteria for vehicles of the future will be, in many respects, the same as they have been in the past. The wide variability in the results of the fatigue evaluation of our structures has been due primarily to our inability to accurately establish the environment. A good example of such variability can be shown by some recent tests conducted at NASA, formerly NACA, (reference 3). On aluminum alloys, it was found that the ratio of fatigue lifes for specimens tested inside the laboratory to those tested in the field are as great as 3 : 1. This difference was found for a testing period extending from one to six months. The difference probably would have been much greater if it had been feasible to conduct the tests over a much longer period - say, one to two years. The continuing deleterious effect of corrosion during the use of a vehicle is believed to have an extremely important influence on its useful life.

Increased effort will be necessary in defining and obtaining the magnitude of the loads, and more attention should be directed to the proper use and inclusion of corrosive mediums in accelerated fatigue testing. Additional emphasis will be placed on such thermally activated mechanisms as creep, stress rupture, thermal stressing, thermal shock, and their combined interaction and/or accummulated effect with mechanical load stressing. More attention will be required in the design of the experiments and in the techniques used to evaluate the results.

The design criteria for the fatigue effects due to isothermal-repeated loading or for cyclic temperatures under sustained loading will reflect the effects of these new environments. For example, long before fatigue cracking can occur in such cases there may have been cyclically accumulated deformation and permanent set at joint bolt holes, Figure 5. Fatigue failure of a structure usually connotes that cracking has occurred; in this case, however, no detectable cracking is present, yet repeated stressing has produced an unacceptable deformation. A basic design criterion for repeated or creep-fatigue loading is that no detrimental permanent set (practically defined) shall develop during the useful life of the vehicle. In other cases, however, thermal cycling can result in fatigue cracking in the usual sense (Figure 11).

There have been a considerable number of high temperature investigations concerning the fatigue of materials. These investigations have included the effects of cycling loads at constant temperatures as well as cycling temperatures with cyclic loads. However, considerably less of this type of data can be found on tests of actual structures. No data can be found on built-up high temperature structures where the effects of speed or rate of cyclical loading was investigated. See example in Figure 6. There are many high frequency applications where this effect may not be significant in design. However, for many vehicle airframe structures it is believed that speed of testing will have to be changed to include the important relaxation and time at temperature parameters. Spectra will have to be

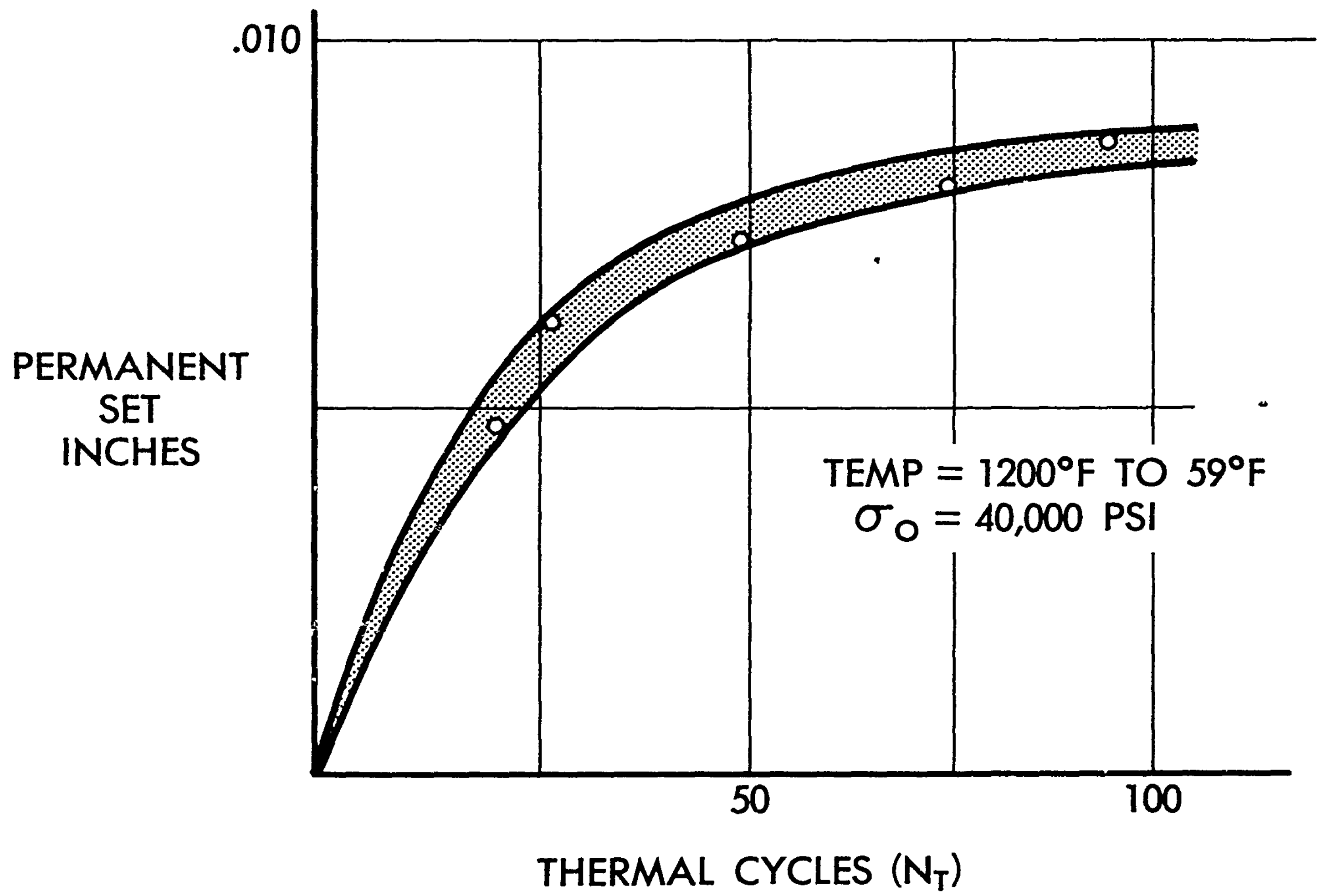


FIGURE 5
Progressive Deformation in Bolted Joint Attach Holes

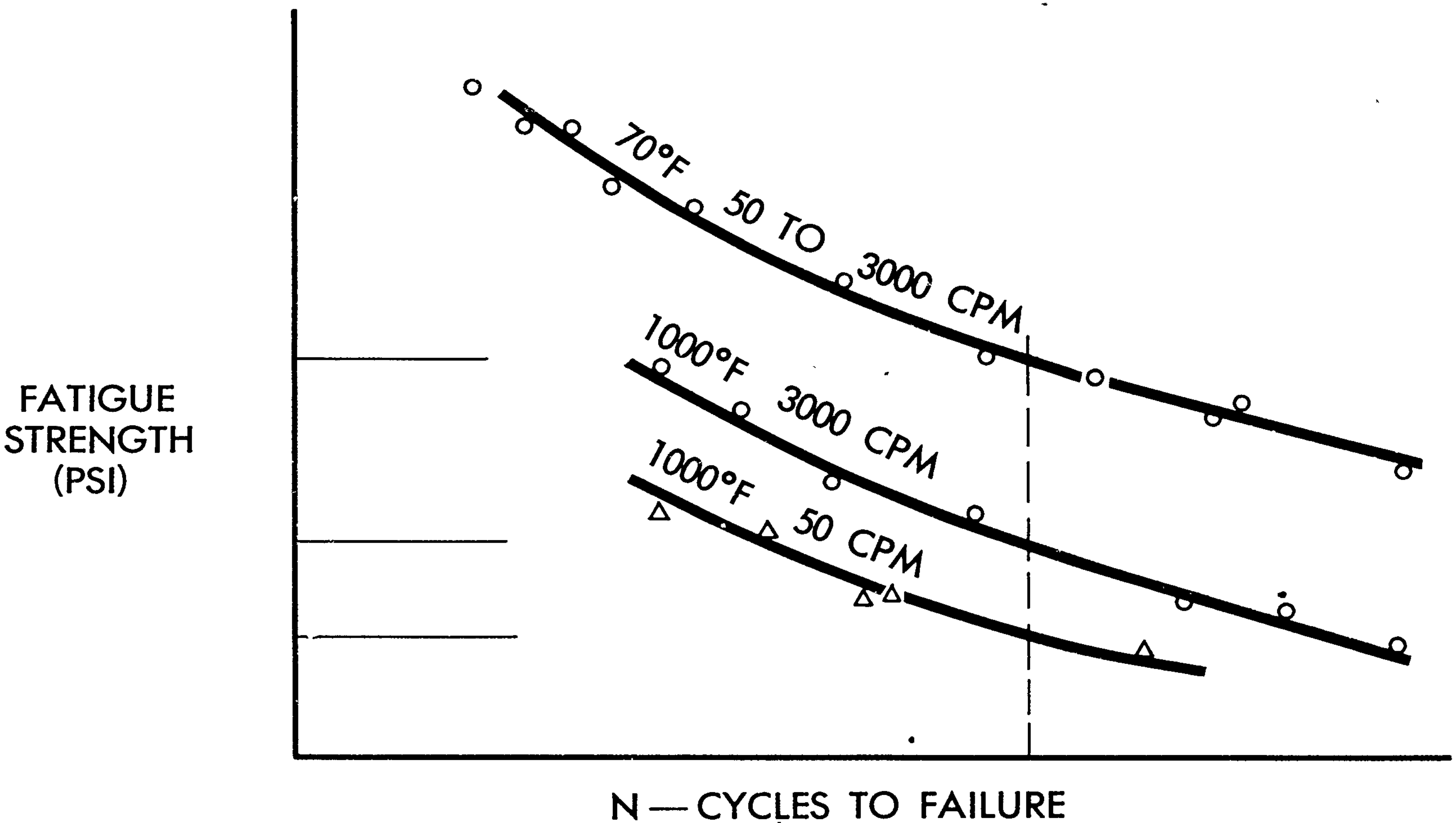


FIGURE 6

Effect of Temperature and Rate of Cycle Loading on the Fatigue Strength of Materials

expanded for the attainment of an efficient design, since the finite life and low cycle region of the SN diagrams are becoming increasingly important particularily for airframe structures. It is suggested that if in thermal fatigue tests of the future more emphasis is given to defining and accounting for the time at temperature during low speed creep-fatigue investigations, more confidence may be gained in the use of the Life-Fraction Rule $L = \sum \frac{t_i}{t_{R_i}}$ as a useable design tool, (reference 4-5).

The adaptation of complex analysis methods to digital machine computation have brought about techniques which have already been standardized and refined to accommodate the effects of temperature, plasticity, and creep. Similarly, the newly developing field in Design of Experiments should contribute much toward providing the needed guidance and organization of testing methods so that complexity will not necessarily mean more difficulty and cost. Careful planning based on mathematical or statisical approaches should do much toward making test programs more effective and productive, (reference 6).

For analysis and test evaluation methods, the type of data that will be most valuable will be similar to that depicted in Figure 6 and 7. There are actually two ways of obtaining the quasi-isochronous SN curves shown by the test data of Figures 7 and 8. One method resolves itself into testing at temperature and at rather wide variations in test speeds. This allows test data to be accumulated at the same number of cycles to failure, but at entirely different cyclic temperature periods. The other method, which may not be reliable for at least some of the precipitation hardening alloys, is to soak the structure specimens for varying amounts of the time under load and then to fatigue test to failure. The most accurate and practical method, or combination of the two, needs to be established.

For high performance type vehicles operating at high temperatures, there is usually a certain amount of deterioration expected in the strength of materials under load. That this phenomenon is a function of time of exposure is evidenced by the well known stress-rupture and creep curves. For this reason, the effects of mechanical stressing on the material strength deterioration due to time dependent high temperature exposure should be determined for these materials suitable in the construction of such vehicles. It is proposed that quasi-isochronous KSN curves for notched coupons be obtained for the purpose of making the appropriate fatigue strength and useful life calculations. It may be sufficient to obtain data such as that outlined in Figure 7 for increments of 100 hours soaking periods and at maximum anticipated temperatures under sustained loads. In this manner, the accumulative load and temperature spectra damage effects for each 100 hour period of useful life may be successively evaluated. It is a little disconcerting to note that as the frequency of loading is decreased, the number of cycles to failure also decreases. In the past, where temperature effects were not present, the S-N-curve was not altered by changing the frequency of loading. In the presence of the effects of high temperature, the life of a part in cycles depends not only on the stress, but also on the frequency of loading. At lower loading frequencies, the adverse effects of creep are greater.

In summary, the proposed fatigue design requirements for future air and space vehicles may be itemized as follows: (The following requirements, however, are to be supplemented by the general requirements, rules, and guides that previously have been set forth).

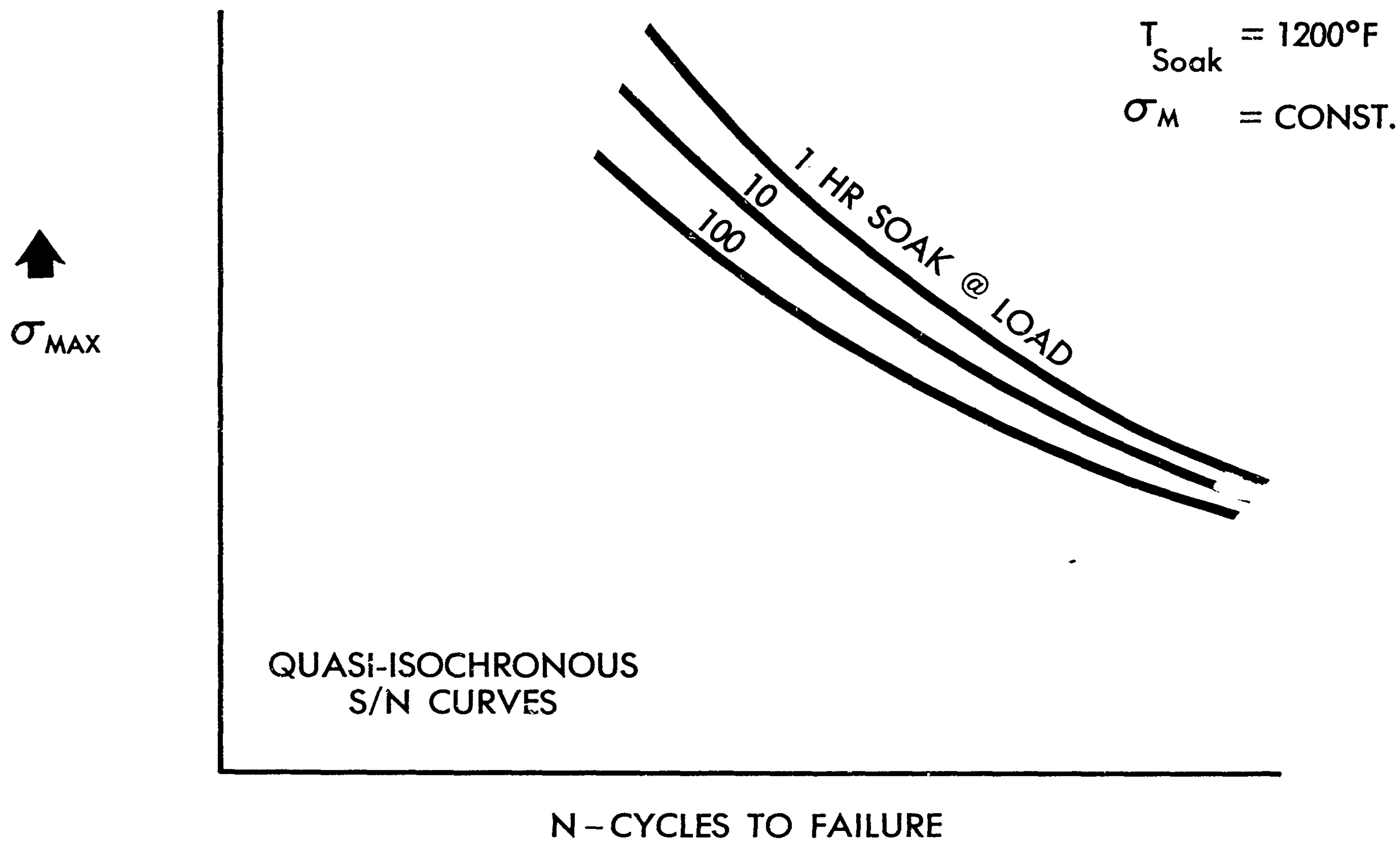


FIGURE 7

Elevated Temperature Tests for Varying Soak Periods Under Load Prior to the Fatigue Test

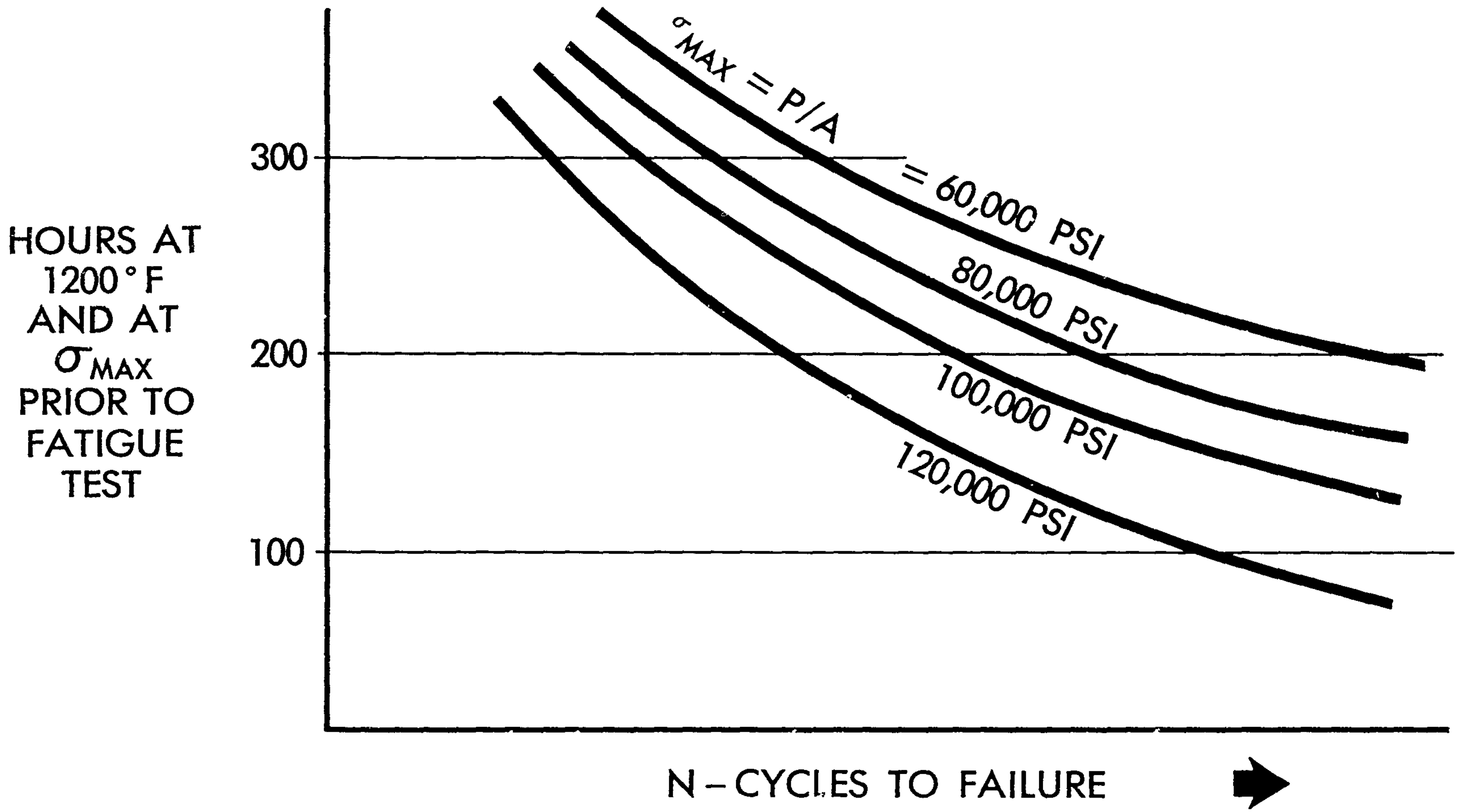


FIGURE 8

Soak Hours vs. Cycles to Failure
(Mechanical Stressing)
Const. Mean σ = 30,000

1. The structure should be shown by analysis and/or test to be capable of withstanding the repeated loads, repeated temperatures, and the environments in or out of the atmosphere as will be expected during its useful life.
2. Cumulative detrimental permanent deformation developed during the use of a structure caused by thermal stressing or sustained loading during cycling of high temperatures should not exceed an amount that will seriously affect the operation, or shorten the useful economic life of the vehicle consistent with replacement costs.
3. No catastrophic structural failure should occur even in the event fatigue cracking, fast cracks or tears, induced by foreign object penetration, occurs.
4. In a greater measure than has been necessary in the past, the strain distributions should be controlled in design and verified in test so as to assure structural reliability for the designated life of the vehicle.
5. Presently the proof for adequate fatigue strength of future high temperature structures can only be substantiated by testing structure at the most realistic loading rates and temperatures attainable.

FUTURE FATIGUE DESIGN CONSIDERATIONS

The efficient design of hypersonic vehicles of the near future, for adequate fatigue strength and service life, will require full cognizance of the mission characteristics, optimum use and fabrication of structural materials, and cautious use of service and test experience.

Mission Characteristics: The wide range of missions for which future vehicles are capable will certainly broaden the overall fatigue problem. Defining and designing to specific mission envelopes will be of far greater importance than it has been in the past. For example, the launch or boost ballistic flight beyond the atmosphere, the re-entry phase, and the capabilities for hypersonic maneuvers for a single vehicle need all be accurately defined if the most efficient structure is to be designed. The broad band of potential fatigue cracking parameters is widened by the variable high temperature environment in the various phases of a flight. The chart of Figure 9 indicates, broadly, the interaction of fatigue with creep. Creep may be a major design factor for some vehicles, whereas fatigue may be the critical factor for others. It is conceivable that both factors may be critical in the design of still other vehicles. Paradoxically, creep in one member may alleviate the thermal stressing in an adjacent member. If only a few members of the structure are subjected to large amounts of creep because of high temperature and stress, these members may be able to shift their load to less highly stressed members of the structure, and the effect of creep and/or thermal stress may be alleviated. A step-by-step application of the Maxwell-Mohr method has been devised to evaluate the effect of thermal stress upon the structure, (reference 7).

Characteristics of the Optimum Structural Materials: A major structural problem associated with very high speed aircraft obviously will be the selection of materials to efficiently carry the loads at the high temperatures resulting from aerodynamic heating. Not so obvious, however, is the selection of the optimum materials to withstand the various and varying environments which can be normal for such a vehicle. To further define the material selection for a particular member design, there will be the inter-relationship between the fatigue strength and the rupture strength at high temperatures. Reference to Figure 10 indicates that for the lower temperatures and high rates of loading, the fatigue strength criteria will dictate the particular element design. On the other hand, for the same case at high temperatures, the short time rupture strength and/or creep strength may establish the design

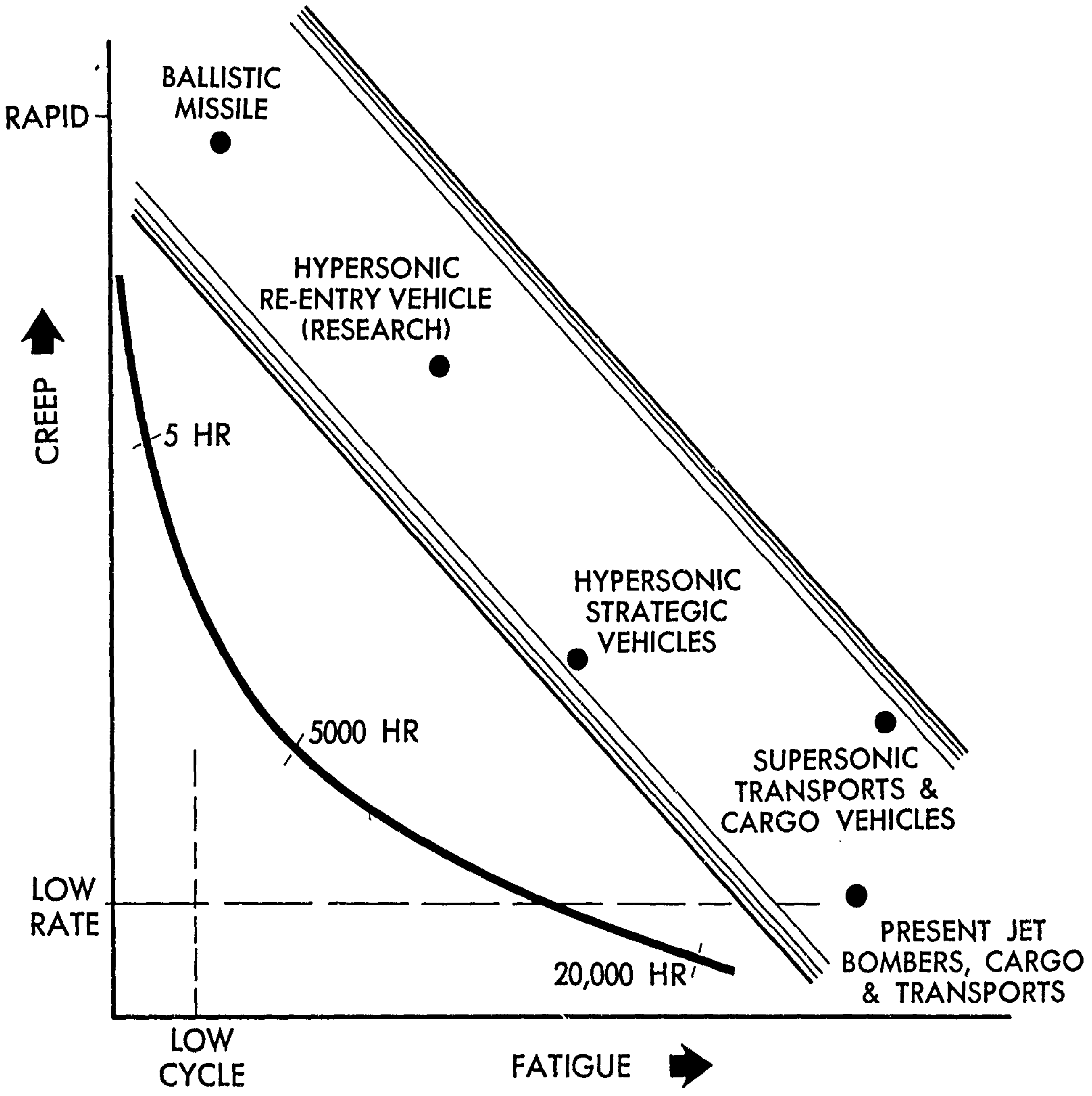


FIGURE 9
Interaction of Creep and Fatigue

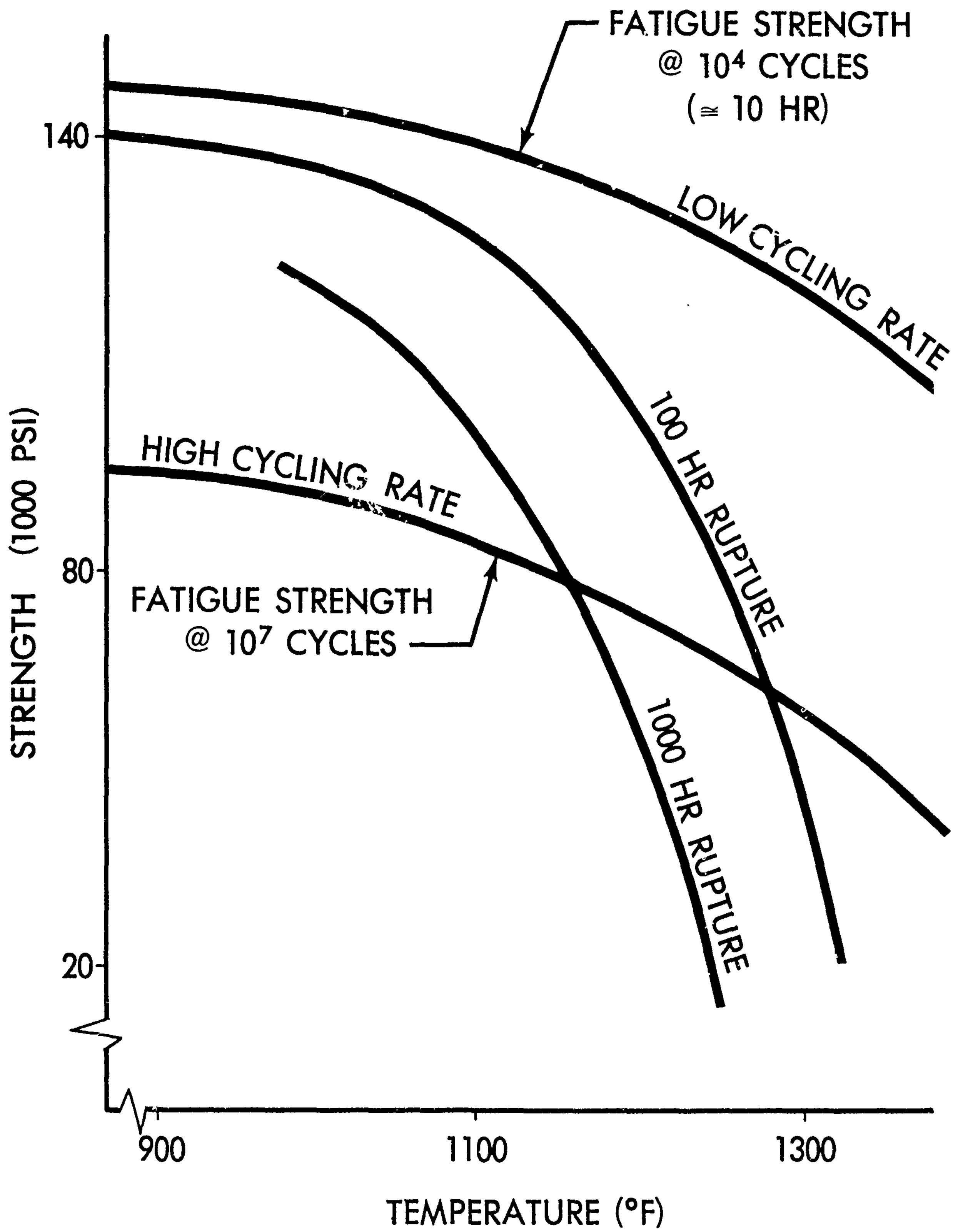


FIGURE 10
Relation Between Fatigue and Stress Rupture

envelope. It can be seen from such curves that the problems in design will be more difficult of solution since these and other parameters, as well as interaction effects, will have to be considered.

Thermal cycling (stressing) and thermal shock are additional physical characteristics of the materials of construction that must be evaluated. The evaluation of these parameters in design will include the effects of various amounts of restraint, see Figure 11. Although some success has been experienced in evaluating most of the above parameters, it is believed that more data are required before complete design criteria can be established.

Fabrication Methods: Although current methods of airframe structure fabrication principally employ riveting and bolting, future trends appear to favor an increase in the use of welding. The fatigue characteristics of airframe structures joined principally by welding are not well known. There is very little service experience with such constructions. It is certain, however, that welded joints are stress raisers and are, therefore, potential fatigue problems. These fatigue problems are inversely related to the ductility of the welded joint. It is known that some of the rare metals which have not yet been used in airframe constructions will be used in these new structures. Molybdenum, for example, will probably be used in these structures where the temperatures exceed 2000° F. It is interesting to note that some of these metals can now be successfully welded. Some of these metals which can be welded and which will probably be used in these new vehicles include Beryllium, Molybdenum, Tungsten, Niobium, Tantalum, and Zirconium. Joining dissimilar rare metals, or rare metals and stainless steels, results almost invariably in joints of low ductility, (reference 8-9). It is clear that the joining techniques to be employed in the fabrication of these new vehicles will require much study to properly assess their fatigue characteristics.

Service Experience: In many respects it will be necessary, as well as desirable, to use the service experience of vehicles of similar structural arrangement in developing fatigue design requirements for the future. Due account, however, will be taken of the differences in the operating procedures and conditions. As in the past, many circumstances will require substantiation of the fatigue requirements by test as well as by analysis. The general techniques used by at least one airframe designer are given in the section on General Fatigue Design Requirements and in the Appendix.

ANALYSIS AND TEST OBJECTIVES

It appears that the useful life evaluation of the structures of either air or space vehicles may be accomplished by analytical means with the support of some structural test data. Such a means might be an expanded life fraction rule which accounts for creep and thermal cycling, as well as fatigue. An application of such an equation, which might be of the following form, is given in Appendix 3.

$$\text{Useful life of member in hours} \cong \frac{1}{\sum \frac{n}{N} + \sum \frac{t_i}{t_{R_i}} + \sum \frac{n\Delta T}{N\Delta T}}$$

In order to establish criteria, it will be necessary first to evaluate the combined effects of fatigue, creep-damage, and thermal stressing with an accurate assessment of the time of loading. In the past, realistic speeds of testing and load wave forms were not important; they are important for future air and spaceframe applications in that the time at load for elevated temperature tests must be accurately

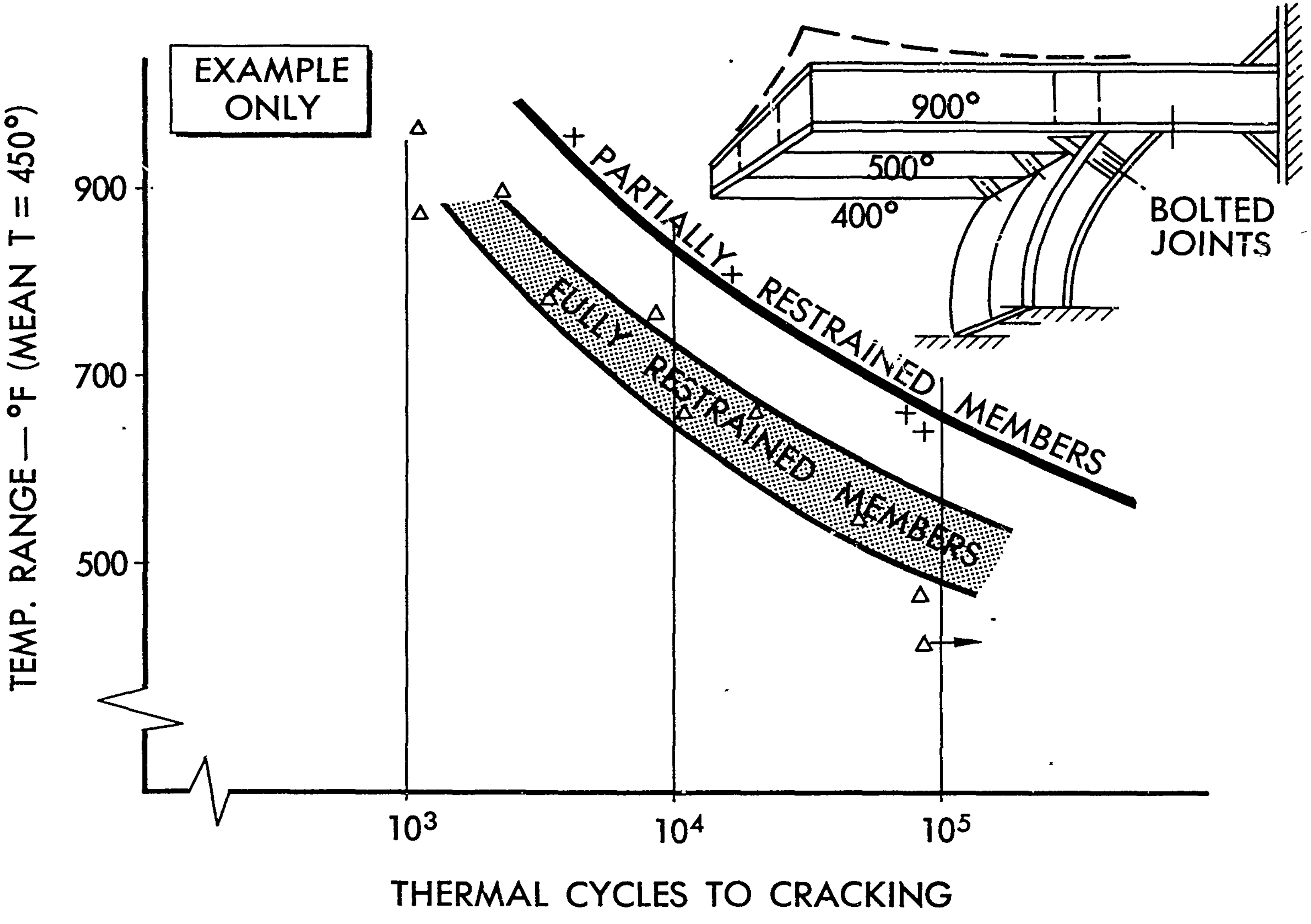


FIGURE 11
Thermal Stressing at Various Fixities.
Restraint Provided by Differing Degrees
of External Loading

measured and assessed. This test criterion is to be observed throughout the extremes of the vehicle categories, that is, from the rapid heating and loading of ballistic missiles to the relatively gradually applied loads of supersonic aircraft and some space vehicles. The attendant problems in rapid testing at elevated temperatures are clearly outlined in reference 10. A schematic sketch of a proposed test setup for applying programmed loads and temperatures is shown in Figure 12.

SPECULATIONS ABOUT THE EFFECT OF SOME SPACE ENVIRONMENTS

Not all mediums should be considered as always damaging to structural materials. For future vehicles operating in partial or near vacuum, there will be less or no oxidation effects. For example, the fatigue test data of Figure 13 indicates the corrosiveness of air with respect to that of a partial vacuum.

Fretting Corrosion: Recently an extensive series of tests have been completed by the Space Research Laboratories of Litton Industries (reference 12) concerning the coefficient of sliding friction between metal surfaces at ambient pressures of 10^{-6} mm of mercury. In these tests it was found that the friction for a large number of metal to metal combinations almost doubled over atmospheric values. It is believed that the greatest significance of this phenomenon is the increase in wear, chafing, or fretting that may occur between faying surfaces of members. A large percentage of service fatigue cracks experienced in present day structures is due to the phenomenon of fretting fatigue.

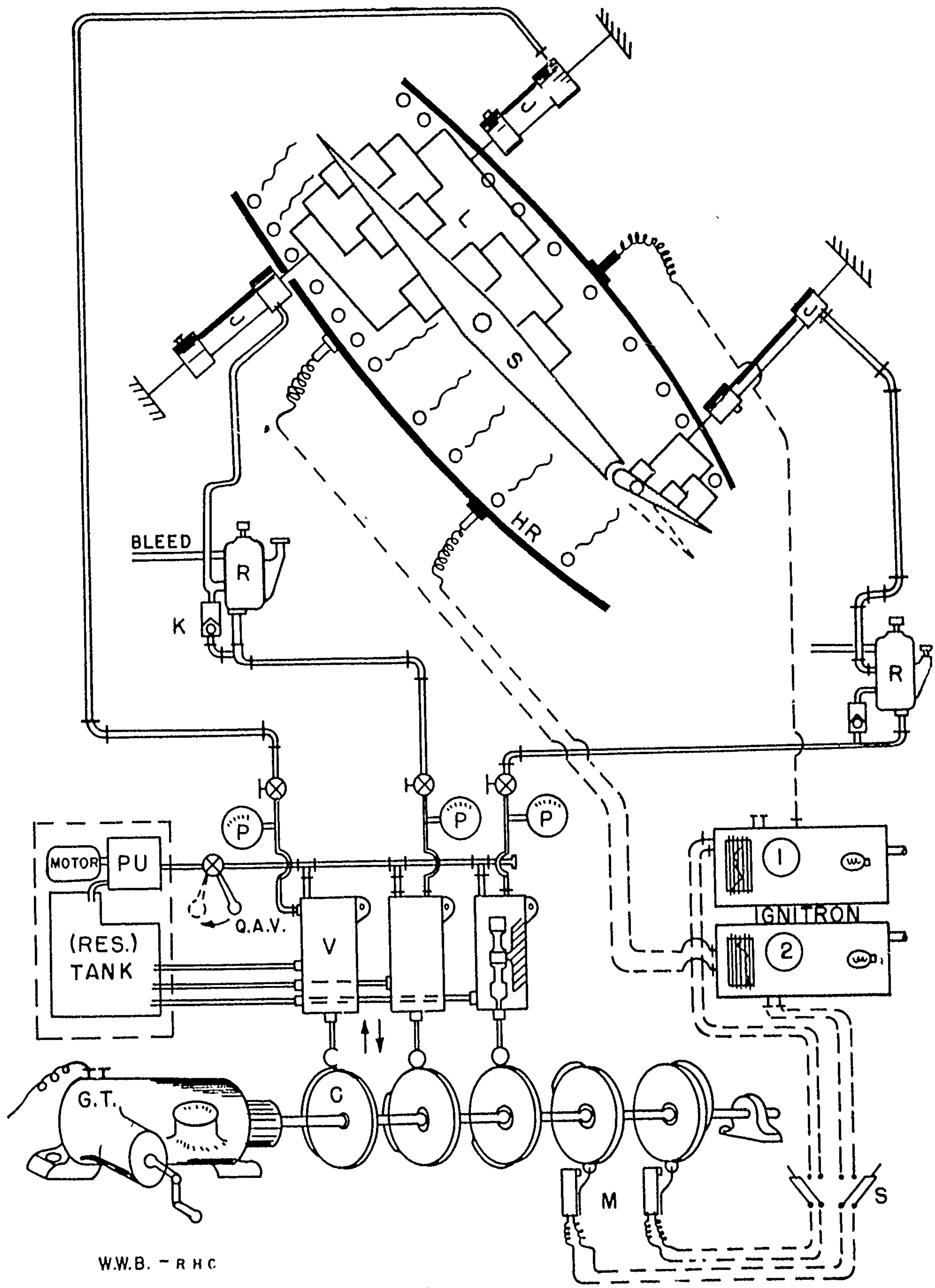
Solar Irradiation: For extremely long period communication-type-satellites, the thermal fatigue action due to the solar irradiation may be a primary problem. It is conceivable that fatigue cracks will occur in primary or secondary structures having restraints and subjected to thermal stressing from 200 to 450° K.

Radiation Damage: Recently there have been many investigations concerning the effect of nuclear irradiation upon the mechanical properties of metals. Radiation induced (reference 13) changes have been termed "radiation damage", since in many cases damage in one form or another has been induced. Such damaging effects as loss in ductility have been noted in many metals. On the other hand, beneficial effects such as increase in surface hardness, yield strengths, and tensile strengths have also been observed. The effects of prior neutron bombardment on the fatigue strength of materials is not well known. Figure 14 shows the results of a few tests on irradiated 7075-T6 aluminum alloy. Although mechanical property changes appear to be a function of the total integrated flux, it is believed that the amount in this test (2×10^{18} fast neutrons/CM^2) was within the range required to show a significant effect on the metal. In these particular tests, the effect of prior irradiation has increased the fatigue strength. However, the slight improvement in life, noted on the graph, may not be significant for the simultaneous action of fatigue and irradiation. It is pointed out that these effects certainly should be considered and studied further.

Meteoroid Penetration: The subject of meteoroid penetration and probability of collision has been discussed in the literature, (reference 15-17). The concern with meteoroid penetration, as far as this paper is concerned, is that of nucleating a starter crack for fatigue crack propagation. If the penetrations are rather large there will be no concern regarding fatigue characteristics. The concern is about critical fracture length of an unstable crack in the specific materials of construction. In some cases, there may be required some additional concepts and philosophies with regard to fail-safe construction, particularily if the working stresses in the future vehicle categories are fairly high; say, 100,000 psi and greater. Nevertheless, it is believed that the reduction in fatigue properties due to small pin holes (meteoroids) can be accurately evaluated for the materials of construction.

FIGURE 12

TYPICAL LOAD AND TEMPERATURE PROGRAMMING SYSTEM FOR FATIGUE OR RAPID LOADING TESTS



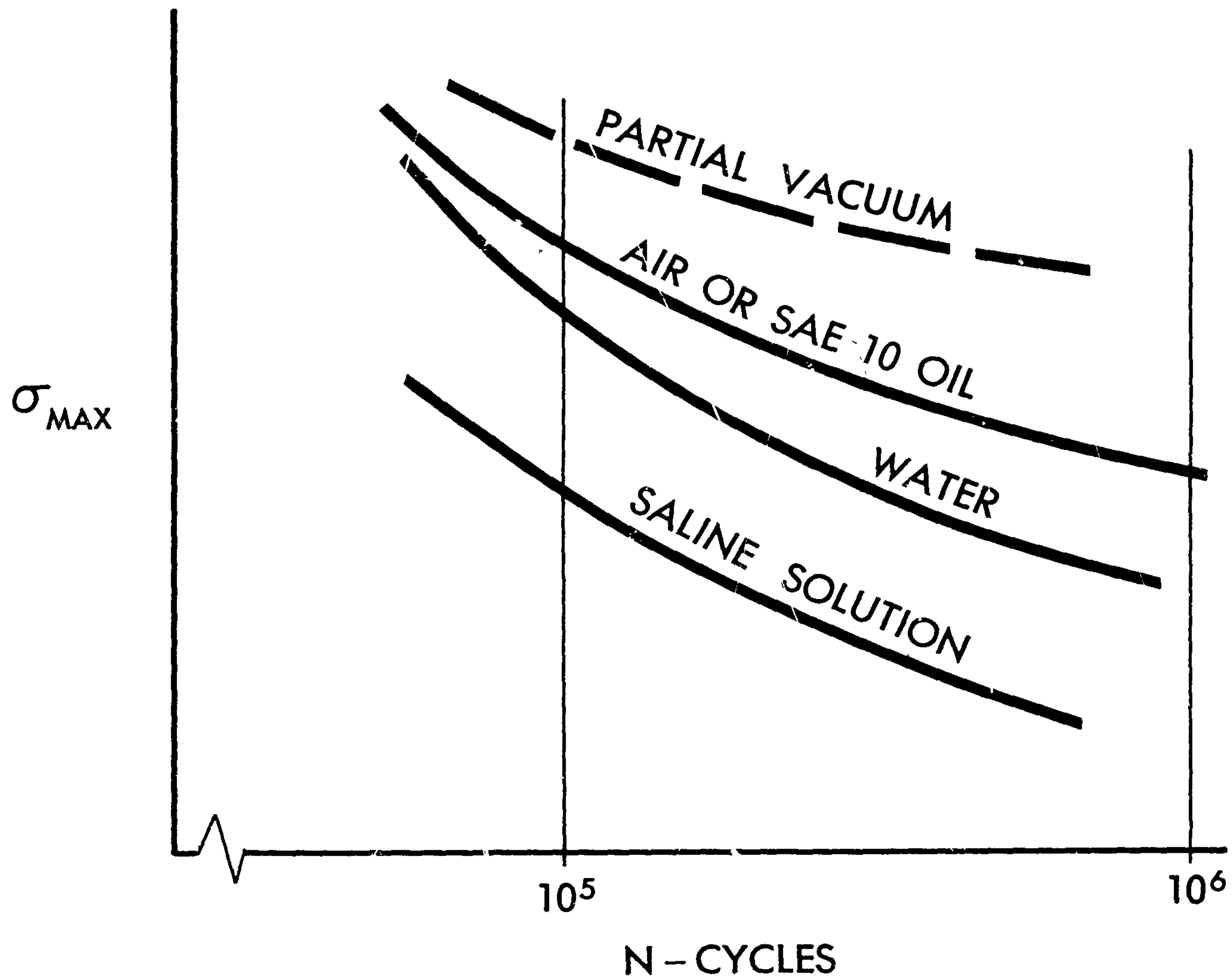


FIGURE 13

Fatigue of Riveted Panels Tested in Various Corrosive Environments

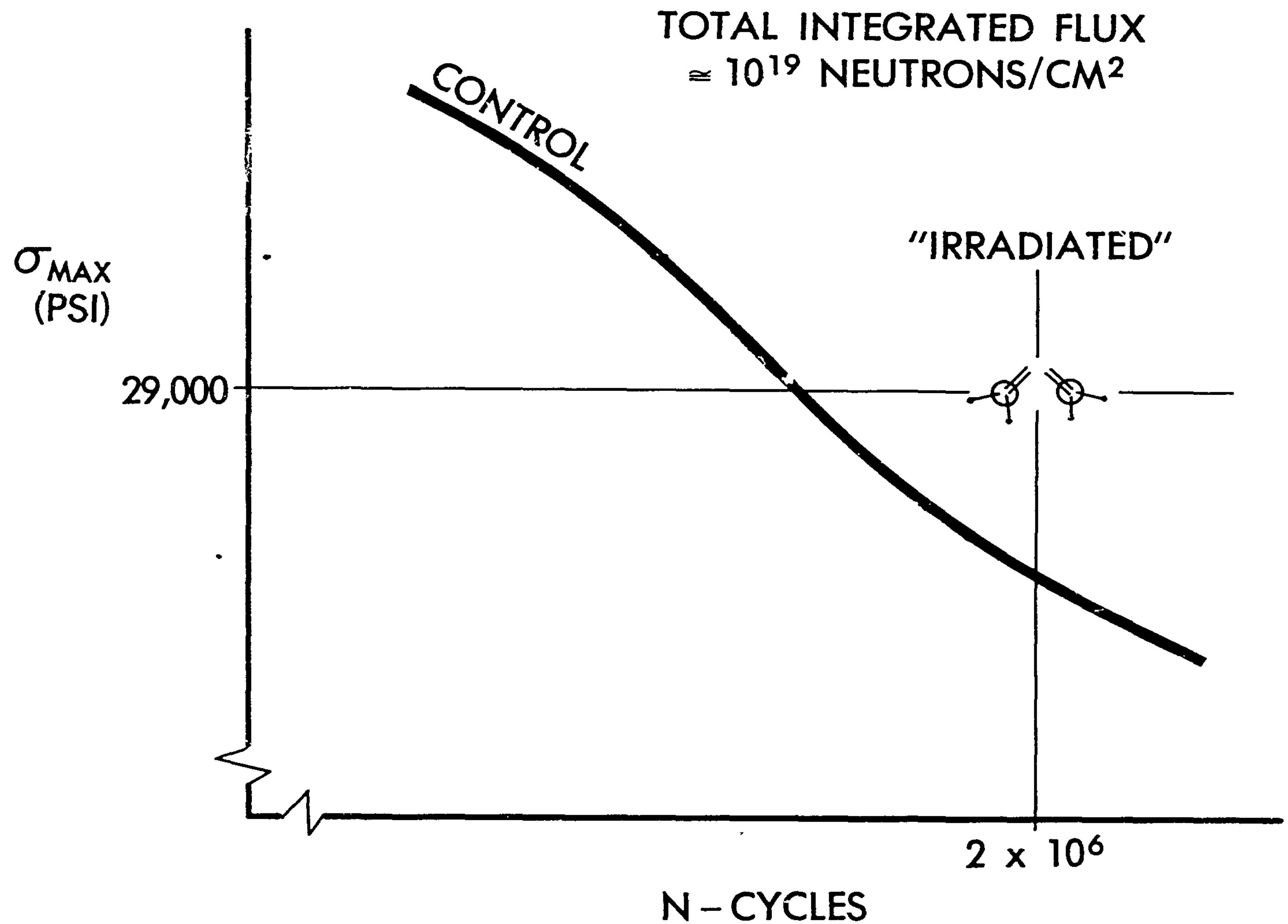


FIGURE 14
Effect of Prior Fast Neutron Bombardment on Rotating Beam Fatigue of 7075-T6 Aluminum Alloy

Loss of Aerodynamic Damping: In general, damping of structural panels, which is largely available from static or dynamic air pressure, is helpful from the standpoint of controlling acoustical and other high frequency vibrational fatigue damage. However, the pressure in space is so low (10^{-6} mm mercury) that the effects of external viscous damping will be lost. This situation in conjunction with a force field of zero gravity will demand a review of the fatigue design criteria for vibrating panels. Sonic fatigue, in this medium, will not present a problem. A problem could arise, however, if the acoustic forces from a rocket motor to the supporting structure are transmitted to adjacent structure through liquid or an insulating air space.

APPENDIX 1

FATIGUE STRENGTH EVALUATION BY PRESENT DESIGN COMPARISON METHODS

For the purposes of illustrating the comparison method, an example will be given of a preliminary design study. The component design is similar to many present air-launch captive system structures, see Figure 15. The carrier system in this example is of the B II - B III bomber category.

The proposed service life for the structural design of such a category is 10,000 hours (5,000 landings and load-factor range from + 2.0 to 0). For this particular case, therefore, the data and curves of Figure 4 can be used; since it is assumed that design techniques, working stress levels, and materials of construction are very nearly the same.

When structural specimens are tested, they normally will fall between the two curves of Figure 4. Those parts which fall close to or inside of the inner curve usually can be improved with no great increase in weight, and should be improved even though there is a weight penalty. Those parts which fall close to the outer pair of lines have good endurance, and further improvement may not be possible without increase in weight. The design effort should be directed towards making all parts come as close to the outer line as possible. Relative position of the part in the band is a measure of the amount of cost, weight, or design time that should go into its improvement.

The elements of the design of Figure 15 were tested with the following results:

Specimen	σ Max.	σ Mean	N - Cyc. to Failure
1	22000	13300	123,778
2	"	"	190,656
3	"	"	212,718
4	"	"	122,110

If these points are plotted on Figure 4, it will be found that they fall above the "60,000 hour inner" curve. In this respect the design is satisfactory since the 10,000 hour design service life has been exceeded.

Limitations of the Design Comparison Method: The comparison method of design evaluation is limited to the extent that loading condition must be known and accounted for in testing. For example, even basic structure may fail prematurely in service if subjected to local normal loads or dynamic overloads which have not been properly accounted for in design. For this reason, care must be taken to ensure that tests include the effects of all loading conditions which the structure may encounter. This requirement puts an added premium on more exact methods of stress analysis.

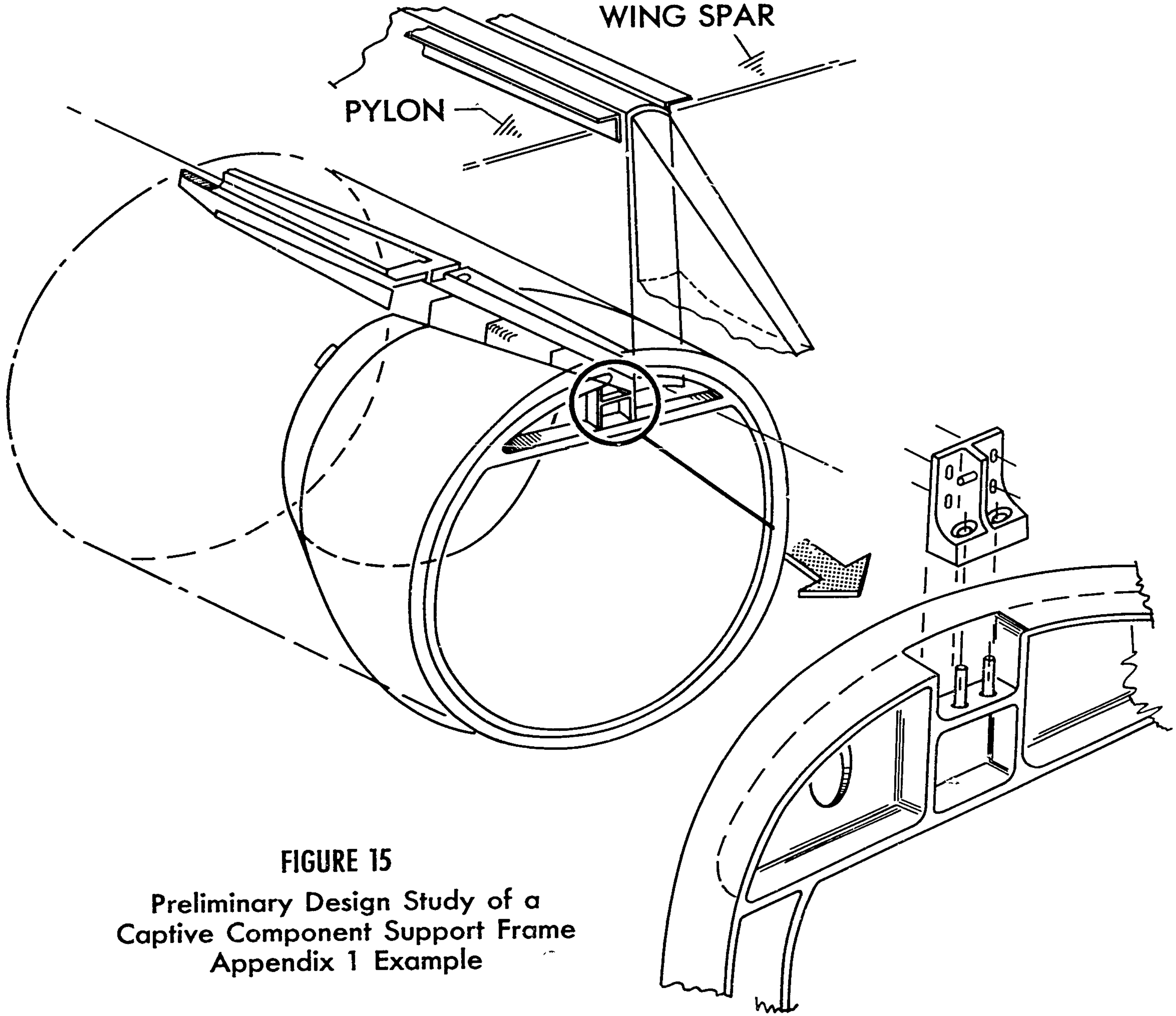


FIGURE 15

Preliminary Design Study of a
Captive Component Support Frame
Appendix 1 Example

Discontinuities must be accounted for when establishing stress levels for test purposes. Test stresses for specimens representing structure in the region of cutouts, for example, must be adjusted from nominal gross stress values to account for the local condition. Specimens must be made large enough to incorporate typical structure working at nominal stress levels, and the discontinuities must be represented.

It is hoped that use of the comparative method of design evaluation in conjunction with fundamental design procedures for providing safety will aid the designer in taking full advantage of materials available and in creating more efficient and reliable designs.

APPENDIX 2

FATIGUE DESIGN EVALUATION BY SPECTRUM TEST METHODS

For the purpose of illustrating this method, an example evaluation is made by the use of programmed load test results of aircraft elements. In this case, the carrier system structural elements are wing spar joints. The design configuration and materials of construction for this particular example are such that the structural member could be considered a retrofit design of a present vehicle to extend its useful life.

The first member-joint tested was taken from an aircraft structure having experienced 20,300 hours of commercial type operation. The part was fatigue tested according to the programmed loading schedule shown in Figure 16. After applying seven (7) programmed cycles, each analytically corresponding to 3,300 useful hours, the joint failed. The fatigue crack life or useful life was then estimated to be

$$H_j \cong (20{,}300) + 7\ (3300) = 43{,}400 \text{ hours.}$$

A slight change in the design details of the joint was then made to obtain some additional improvement in life. A "new" material member-joint was then tested to the same loads and for the same gross area stresses. The results of the retrofit design were as follows:

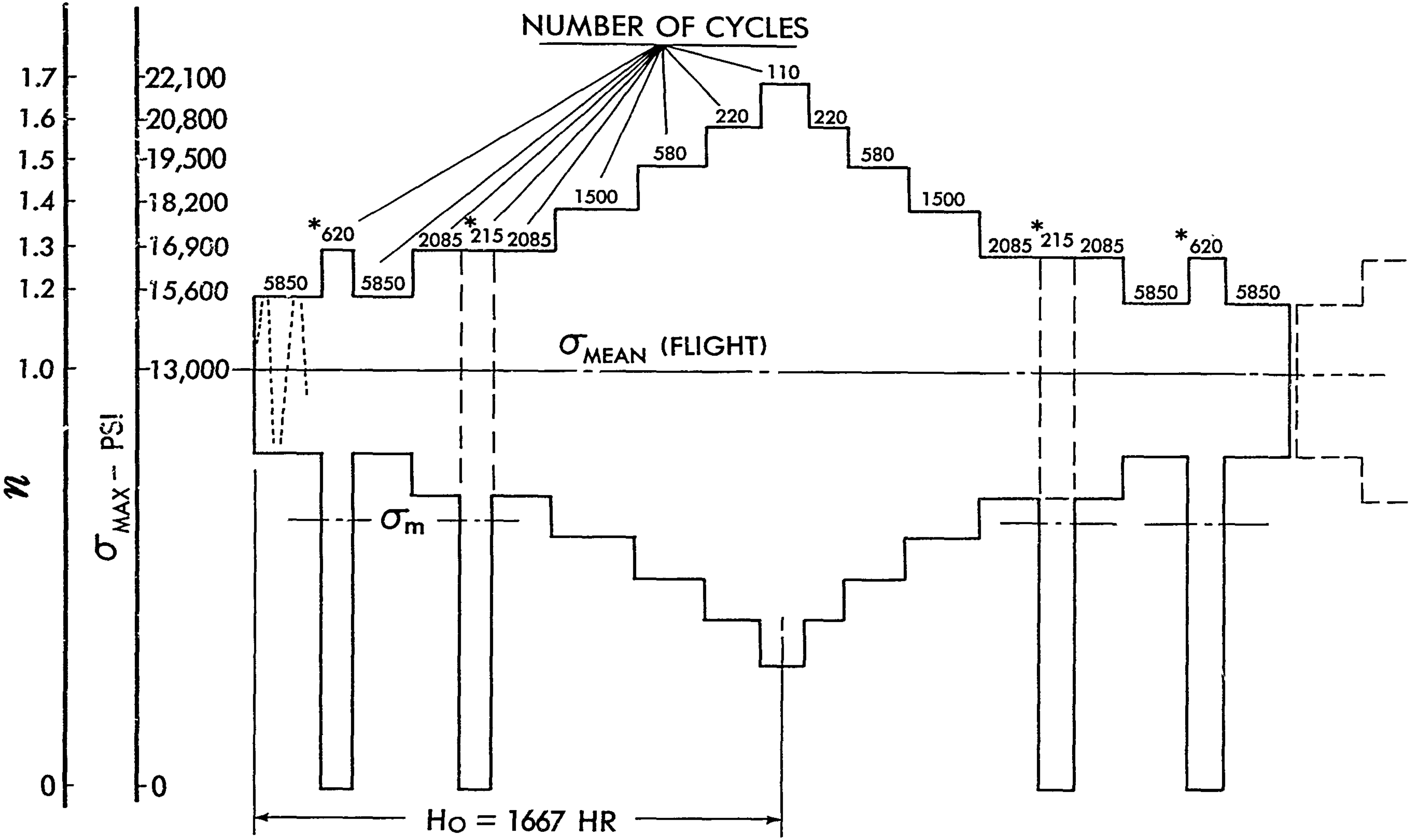
The part endured seventeen (17) programmed cycles of the pattern shown in Figure 16. The fatigue crack pattern is shown in Figure 17.

$$H_{j_{redesign}} \cong (17) \cdot (3300) = 56{,}100 \text{ hours.}$$

It is interesting to note the close agreement of the test lifes of the two joints even though one joint has experienced approximately one half of its damage due to the loads imposed by service usage.

The selection of the stress increments used in this test was purely arbitrary; 1000 psi steps at the high levels and 500 psi steps at the lower levels were used. There does not appear to be sufficient test experience to establish a test criterion at this time. A preliminary analysis is usually made of all pertinent load data, and the findings are arranged in a convenient and practical test pattern for programming.

The program cycle used in this example is representative of a rather large number of hours. For some other categories, the test integration of damage may best be accomplished by 100 to 200 hour program cycles.



*GROUND-AIR-GROUND CYCLE BASED ON ONE CYCLE PER 2 HR. FLIGHT

FIGURE 16

Test Programmed Loading Schedule
of a Wing-Spar Joint
Appendix 2 Example

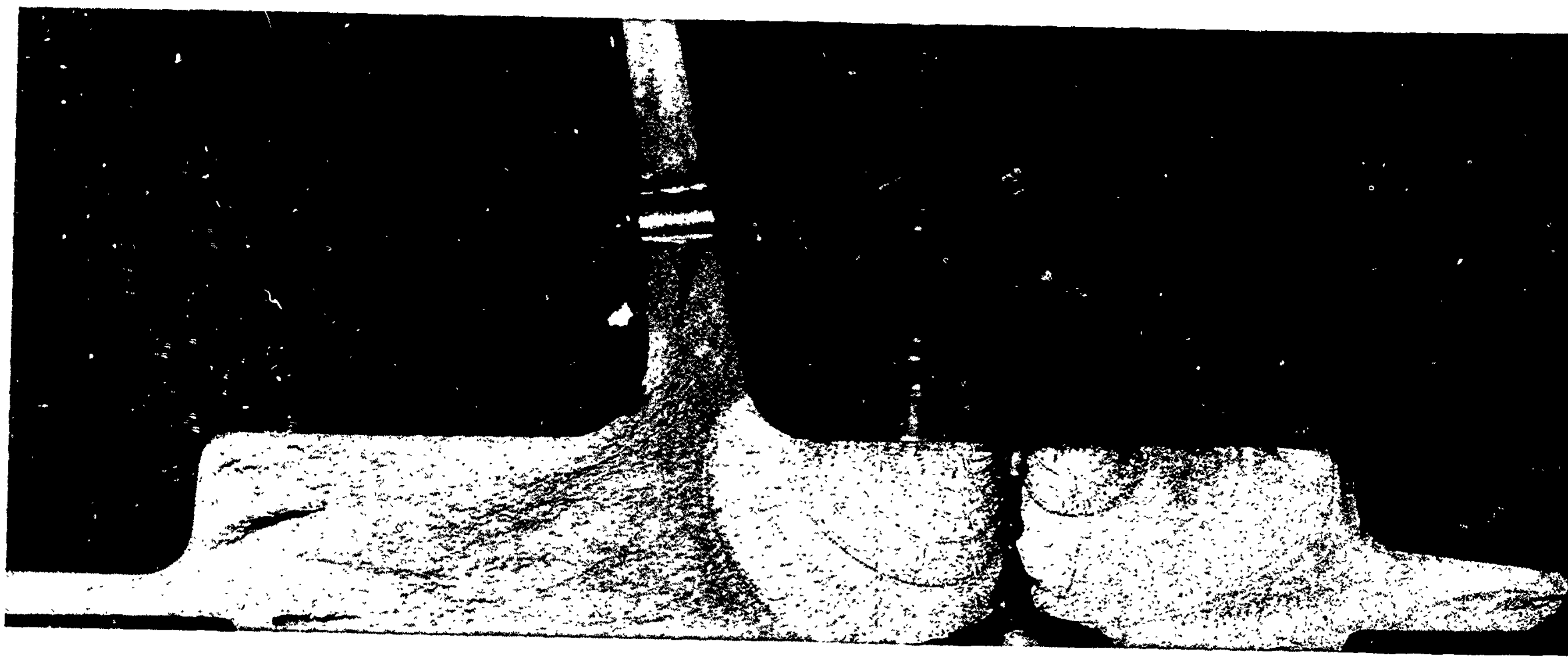


FIG.17

7075-T6 ALUMINUM-ALLOY EXTRUSION UNDER AXIAL LOADING. NOTE REPETITION OF CRACK-GROWTH PATTERN. THIS PART WAS TESTED IN THE LABORATORY THROUGH A SIX-STEP INCREASING AND DECREASING SPECTRUM BLOCK. IT WAS CYCLED THROUGH 17 SUCH BLOCKS BEFORE FAILURE.

It may also be of interest to note that an additional specimen, tested at a single load excursion of $\sigma_{max.} = 22{,}000$ psi and $\sigma_M = 13{,}300$ psi failed in 62,280 cycles. Plotting this data on Figure 4 indicates a design below the "inner 60,000 hr. band" by the Design Comparison Method. It should be noted that in this case the data is compatible with the prior spectrum evaluations.

APPENDIX 3

FATIGUE STRENGTH EVALUATION OF FUTURE HIGH TEMPERATURE VEHICLE STRUCTURES

The following discussion briefly outlines the major steps to be taken for the analysis evaluation of near future structural member designs. This will be done with the aid of numerical examples for a hypothetical structure using a typical, yet realistic, loading and temperature spectra employing test and interpolated data.

It is to be made clear, however, that the spectrum testing of a full-scale vehicle structural component, including the combined effects of all of the factors that should be considered, has not yet been made.

The overall load and temperature spectra are shown in Figure 18. The histogram of operating stresses, temperature ranges, and corresponding periods of duration are listed in Table I. The fatigue damage evaluation for the anticipated occurrences of the various loads considered is given in Table II. The normal life (N) to failure for the various stress levels and temperature ranges has been taken from Figure 19 (the SN diagram of the particular design joint). In this first approximation, analysis of the estimated useful time prior to fatigue cracking calculates to be 6700 hours. It is realized that the fatigue cracking and damage in this high temperature example would, in all probability, be accelerated by the localized creep-damage that would occur simultaneously with the damage imposed by the mechanical straining of the flight loads.

In the absence of laboratory test data wherein these effects have been simultaneously and realistically evaluated as time-dependent parameters, it will be necessary to make separate analyses pertaining to creep-damage and thermal cycling.

From Table I it is estimated that the total cruise time per 100 flight hours for the particular vehicle is 60 hours at high temperature. It is further estimated that the average peak stress value for the cruise stage is 60000 psi for one hour.

A creep test at 900° F and for a nominal tension stress of 60,000 psi for the design joint indicates a permanent deformation of approximately 0.2% at the end of 110 hours. This value may be extrapolated from a 20 hour test, (Figure 20). It is assumed that a 0.2% creep strain has been set up as a design criterion.

An additional creep damage evaluation is made for the infrequent high maneuver and gust loads at the same temperature. One hundred thirty significantly high gusts and maneuvers per 100 hours with times at load and temperature ranging from 0.1 to 7 seconds calculates to be a total accumulative time of 0.145 hours for each 100 hours or 9.7 hours within the useful period of 6720 hours. The creep curve of Figure 20 indicates a safe time period of 20 hours at 75,000 psi.

The thermal stressing of the vehicle joint that occurs in the transition from ground load to maximum probability maneuver L. F. per flight and from approximately 59° to 900° F calculates to be

$$20 \text{ flights} \times \frac{6720}{100}$$

$$= 1340 \text{ cycles.}$$

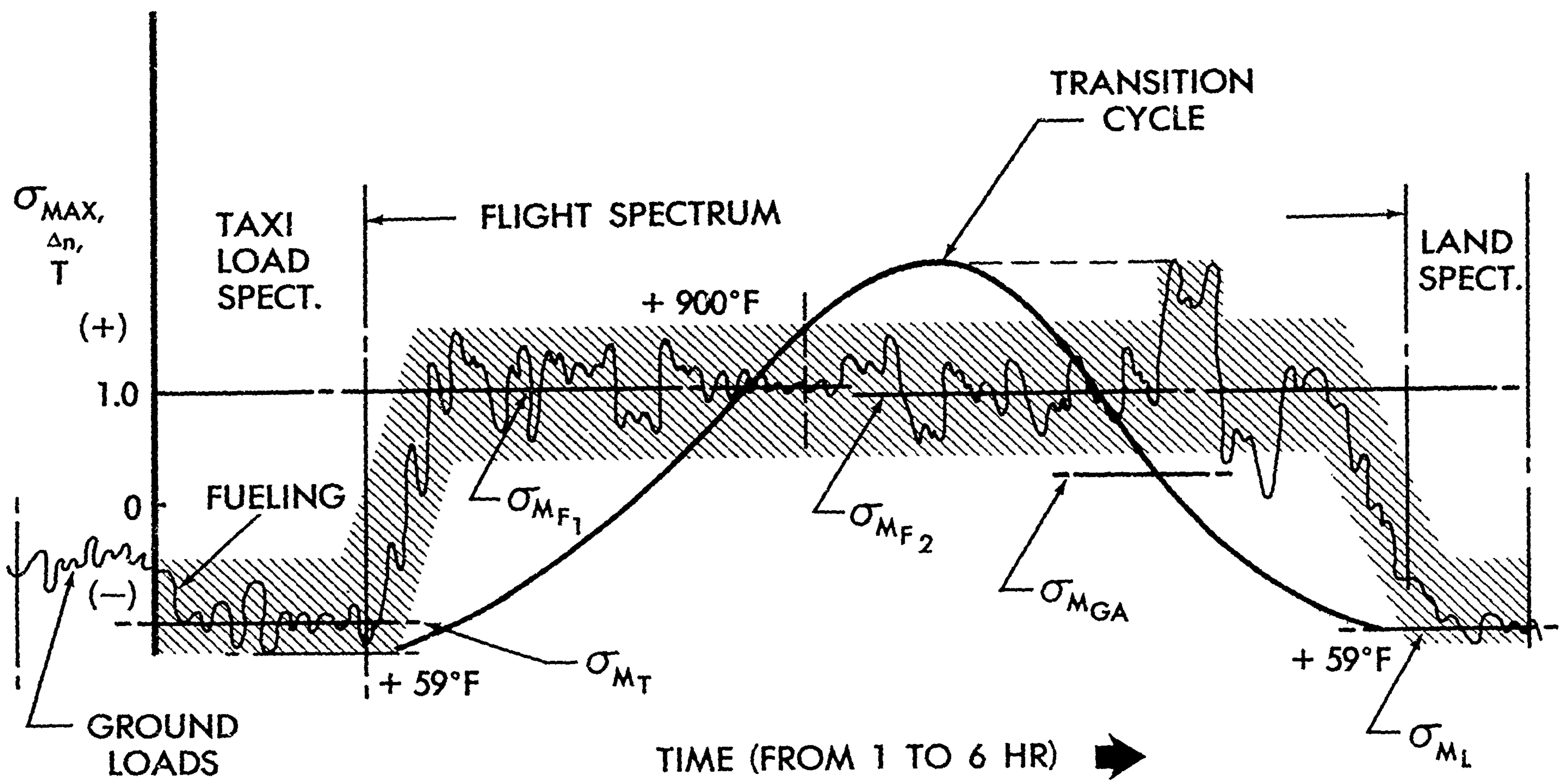


FIGURE 18

Flight, Ground Load and Temperature Spectra
Example No. 3

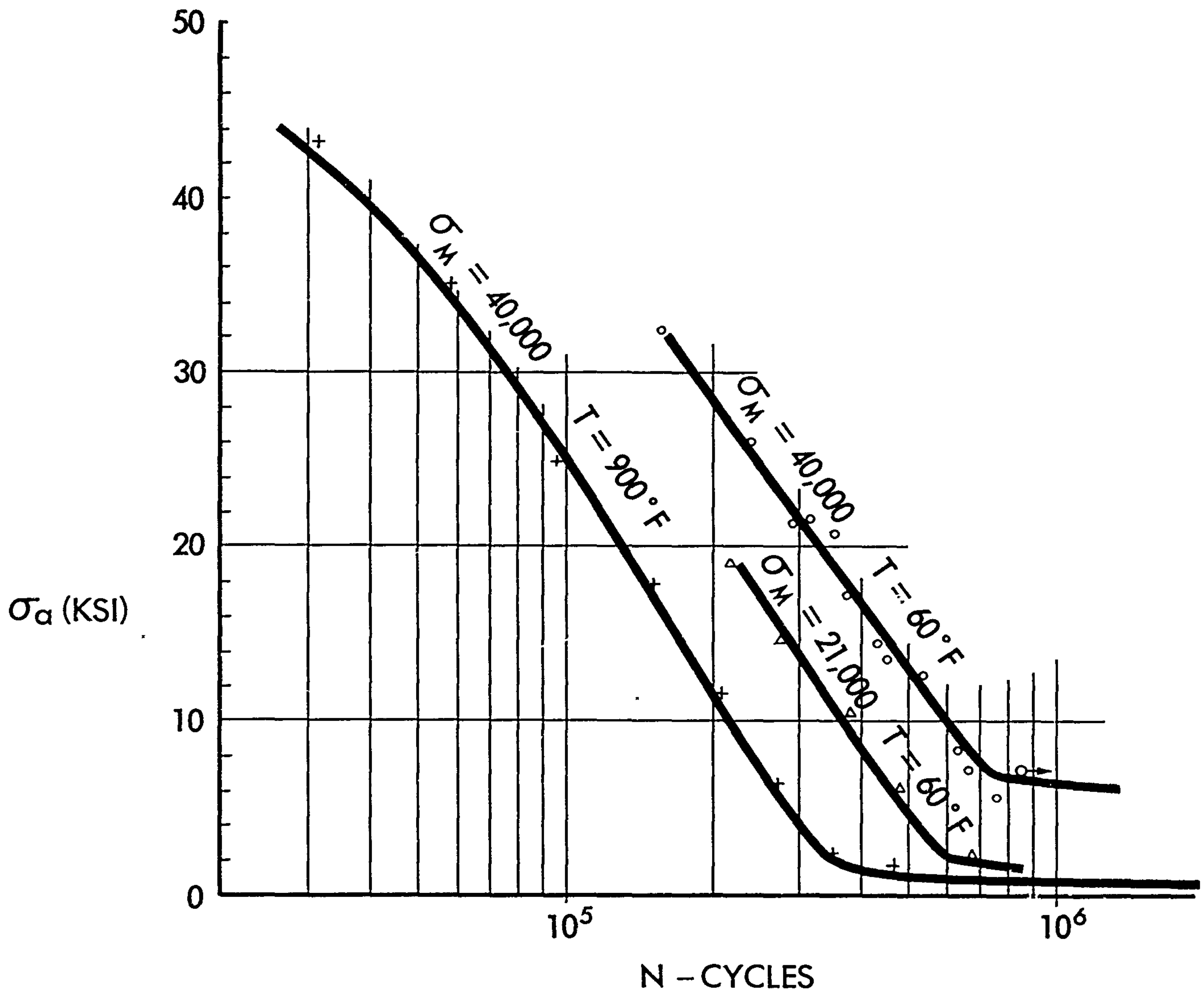


FIGURE 19

S/N Curves of Wing-Fuselage Joint of Appendix 3 Example (See Figure 18)

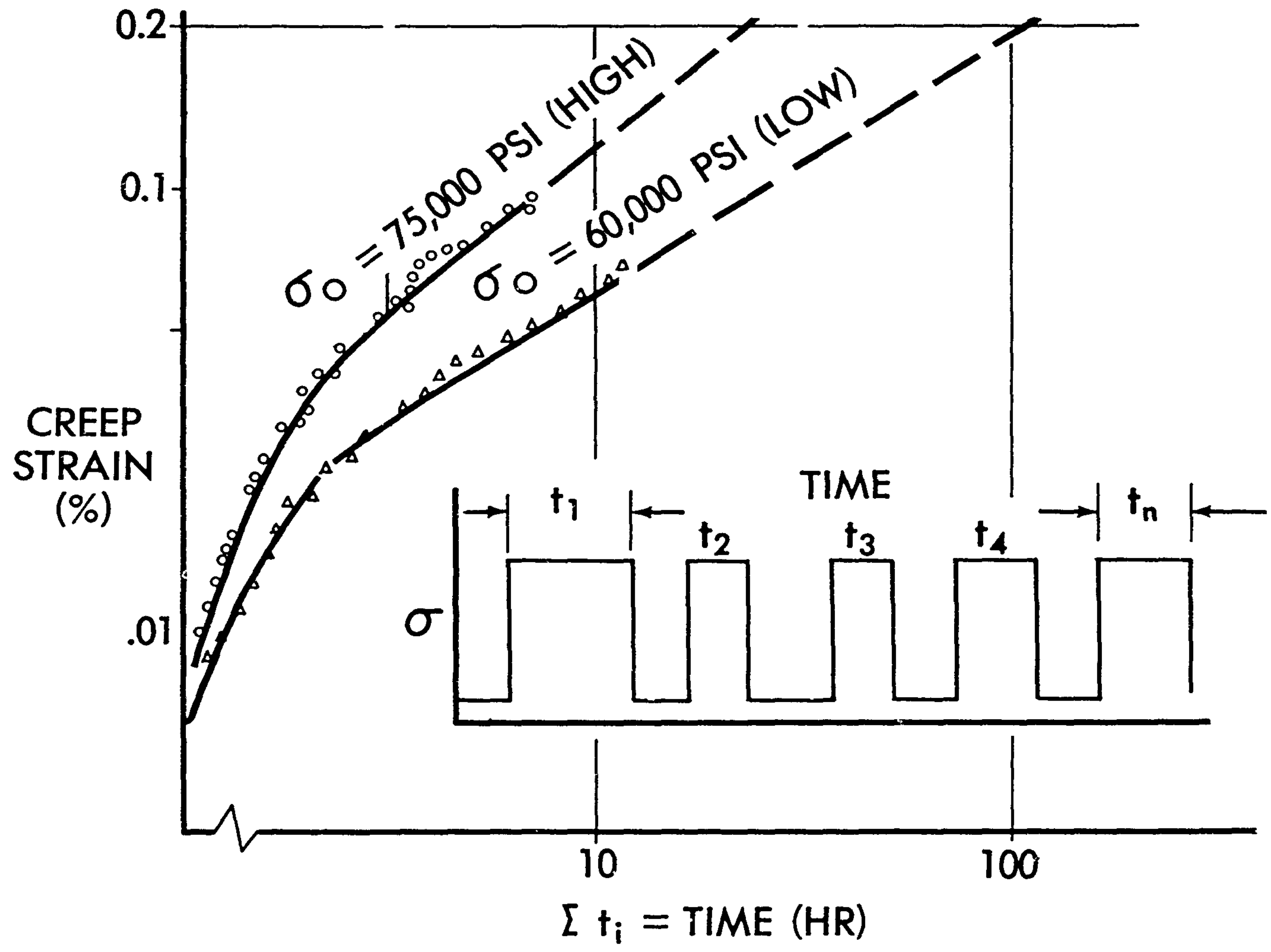


FIGURE 20

Riveted and Bolted Joint
Temp. = 900°F

TABLE 1

EXAMPLE WING-FUSELAGE JOINT

FATIGUE ANALYSIS AND TEST LOAD SPECTRUM

Loading	Mean Stress	Max. Stress	Load Cycles For 100 Hrs. Service (20 flts.)	Temp. Range °F	100 Hr. Service: Total Time For Temp. Anal.	100 Hr. Service: Creep	100 Hr. Service: Thermal Stressing (cycling)
Gusts	40,000	42,000 46,000 54,000 58,000 62,000 66,000 70,000	1,350 400 140 55 21 9 4	Climb 59° to 900° F Enroute at 900° F Descent 900 to 59° F	60 hrs. at cruise maneuvers & gusts avg. σ = 60,000 psi for 1.0 hour 85 gusts in excess of 1.45 G σ = 75,000 for example	low rate high	20-1/2 cyc. 20-1/2 cyc.
	40,000						
Maneuvers	40,000	44,000 52,000 60,000 68,000 76,000 84,000	300 100 30 10 3 1	1/2 enroute = 900° F	60 hrs. (incl. in gusts) 45 maneuvers in excess of 1.5 G σ max. = 75,000 psi	low rate high rate	10 cyc.
	40,000						
Landings	21,000 21,000 21,000	23,100 29,400 35,070	19 10 5	59° F			
Taxi Brake Turn	21,000 21,000 21,000	25,200 27,300 29,400	2,600 260 26	59° F			
Transition G-A-G	33,500 σ_{min} = 7000	60,000	20	59° F to 900°			

TABLE 2

STAGE	n *	N **	$n/_N$ PER 100 HRS.
Gust	1,350	350,000	.00386
	400	270,000	.00148
	140	170,000	.00082
	55	130,000	.00039
	21	105,000	.00020
	9	84,000	.00011
	4	67,000	.00006
Maneuvers	300	300,000	.00100
	100	190,000	.00053
	30	120,000	.00025
	10	74,000	.00013
	3	48,000	.00006
	1	25,000	.00004
Landings	19	570,000	.00003
	10	410,000	.00002
	5	290,000	.00002
Taxi, Brake Turn	2,600	510,000	.00510
	260	460,000	.00056
	26	400,000	.00006
Transition	20	120,000	.00017
			$\sum =$ .01489

* Col. 4, Table 1
** From Figure 19

$$\text{Useful Member Life} \cong \frac{100}{\sum \frac{n}{N}} = 6{,}720 \text{ Hours.}$$

The cycles of thermal stressing for the temperature range and degree of joint restraint is evaluated in the laboratory to be greater than 5000 cycles as shown in Figure 11 at a temperature cycling differential of approximately 840° F (900-59). The limiting life factor in the above example has been evaluated to be due to creep at elevated temperature. Rather than evaluate the relative damage contributed by the various factors it is suggested that the damage in this case be evaluated on the basis of total damage imposed for each 100 hours of service. In the absence of the full-scale hi-temperature vehicle structural test wherein the various damaging parameters are timewise and simultaneously interacting it may be necessary to make separate calculations for fatigue, creep, and thermal cycling. The separate damage calculations are then additively combined. The calculated net useful life from all of the damage contributing factors may be conservatively estimated to be

$$
\begin{aligned}
\text{Damage}/_{100\ \text{hrs.}} &\cong \text{Fatigue} + \text{Creep} + \Delta T\ \text{Cycling} \\
&\cong .0149 + \frac{1}{110} + \frac{.145}{20} + \frac{20}{5000} \\
&\cong .0351 \\
\text{Useful life} &\cong \frac{100}{.0351} = 2{,}850\ \text{hours.}
\end{aligned}
$$

This calculated life for a designated category life of 4000 hours is unsatisfactory and a redesign of the joint is necessary. It is proposed that a conservative estimate of the useful life of a hi-temperature structure may be made for the interaction of the damaging parameters by the simple equation:

$$
\text{Useful life of member in hours} \cong \frac{1}{\sum \frac{n}{N} + \sum \frac{t_i}{t_{R_i}} + \sum \frac{n\Delta T}{N\Delta T}}
$$

Design rules such as these are usually empirical and since there are not enough data available, rules which presently exist must be regarded as tentative. The immediate need for design guides, however, forces the use of some method of extrapolating data until full-scale tests have been made under the simulated conditions of service.

ACKNOWLEDGMENT

Appreciation is expressed to Mr. George E. Bockrath for his helpful suggestions, particularly in regard to the contributions relating to high temperature creep.

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WADC VIEWS ON ESTABLISHMENT OF STRUCTURAL LIFE CRITERIA

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The fatigue evaluation portion of the Air Force structural integrity program is outlined. The desire of the Air Force for a statistical approach to predicting the structural is discussed. Special emphasis is given to the various features and to the variances that exist in each aspect of an analytical fatigue analysis. Several cases of these variances are supported with factual data.

INTRODUCTION

The most significant major objectives of the fatigue life evaluation portion of the structural integrity program are: 1) To obtain, for specified operating and environmental conditions, a first order estimate of the fatigue lives of our existing aircraft fleet and a built-in desired life for all future designs, 2) To determine now what is necessary to correct for any life deficiencies which may exist, in lieu of such a determination under crash conditions, 3) To follow-up with a program to monitor the life integrity status of each aircraft type throughout the types operational life, so that the programed life will be consummated within our maximum capability to do so.

In establishing the life of a given system, a statistical approach is desired such that the predicted life will be expressed in terms of a

statistical quantity. Non-quantitative assertions, implying that the design is such that the risk of failure is extremely remote is unacceptable, even though in reality this may be true. There is no reason to believe that all that would be accomplished in the way of providing life integrity through a non-quantitative approach would not also be provided in a statistical approach. Fail-safe features are recommended and desired but are insufficient by themselves for Air Force purposes. In order to satisfy the requirements for logistic planning, the proposed finite fatigue life certification procedures are mandatory. It is also insufficient to merely establish the mean life of a system when what is most desired is a knowledge of when the shortest lived aircraft will probably experience difficulties.

The question might arise as to the weight penalties a statistical approach might have because of the large statistical factors which might have to be employed. It is contended that the weight penalties will not be unduly severe. Most fatigue problems are concentrated in local areas of joints, in abrupt changes in load paths and the like. Tremendous improvements can be made at critical locations for small weight increases prior to the necessity for gross section area changes to reduce stress levels.

As described by Colonel Taylor in his presentation, the approach to establishing structural life integrity is through the steps, which are given again here:

a. Perform a comprehensive structural fatigue analysis for the entire airplane.

b. Conduct element tests of critical areas to verify their fatigue characteristics. The selection of these areas will be based on stress analysis and on results of the static test.

c. Conduct a full scale cyclic fatigue test of the complete airframe. The loading spectrum for this test will be derived from the findings of the fatigue analysis.

d. A flight test to verify the structural dynamic response to flight and ground loading conditions.

e. Monitor, through a service load recording program, the operational and load experiences of each aircraft type.

f. Monitor the integrity through a structural inspection program and correct anticipated deficiencies on an orderly basis. The where and when to inspect and repair to be derived from the results of the preceding steps.

The aspects of sonic-induced fatigue have not been included above. This is not intended to minimize the problem but is done so only because the subject is not within the scope of this paper.

In spite of the elaborate program outlined here, there is no delusion that a finite life can be predicted for an individual and specific aircraft within the present state of the art. However, it is believed that a reasonable estimate can

be made of the statistical life for a fleet. Further, it is not believed that this program will completely prevent the occurrence of some fatigue problems. It is hoped, however, that for those problems which remain they will have been relegated to the category of no consequence.

ASPECTS AFFECTING LIFE PREDICTION

Since the fatigue analysis, which is being required, will be used in the design analysis stage, in providing the laboratory test spectrum, in evaluating or modifying the test results, and in evaluating the effects of changes in the loading experience during the operational life, let's look then at what is required to conduct a fatigue analysis and particularly at some of the statistical variances that exist in the various steps. The following are the essential ingredients to an analysis: 1) A life requirement in years of hours of flight, 2) The proposed or programed operational utilization of the aircraft, 3) A predicted loading experience based on the missions of (2) preceding, 4) A stress analysis to transfer loads into stresses, 5) The fatigue characteristics of the structure in the form of S-n data, 6) A method or technique for assessing the damage inflicted by the loading history, and 7) Finally, a criterion philosophy which fixes the required level of safety or an acceptable probability of failure within the required service life.

DESIGN LIFE REQUIREMENT

In establishing the first item, a design life requirement, the goal should be to set this value as close to the actual anticipated required service life as possible. A guide line can be obtained through using procurement and replacement schedules, combined with purposed utilization rates in hours per year. It is an absolute necessity that the Using Commands be involved in establishing these utilization rates. These values must be further tempered by the useful life experience of past and present systems. This is exemplified by the remarkable extended life of the DC-3. Since foresight is substantially shorter than hindsight, the statistics of past experience may provide a better guide. Also, it is axiomatic that utilization of the full service lives must be anticipated for all designs. For combat type systems, consideration must also be given to the required back-up support role that retiring systems play and in the case of fighters, to the added operational usefulness by their being frequently passed along to another command or in their being delegated to fulfilling other combat roles. Also, as a system nears the end of its required peacetime service, some margin on endurance should be considered for the needed reliability in fulfilling actual combat missions. These end of life, or secondary roles, should not be entirely forgotten when proposing a design life based on assumptions pertaining only to present procurement needs. For those structural elements which are not reasonably accessible for inspection or for a modification to extend the life if desired added factors on life to reduce the probability of failure should be considered.

OPERATIONAL UTILIZATION (MISSION PROFILES)

The operational program, i.e., mission planning, that is to be assigned a given system must be furnished primarily through the assistance of the Using Commands. Since the major covering objective of the total effort is to maintain a trouble-free peacetime force, suitable for combat at any instance, it is the

peacetime training and proficiency flying which will consume the structural endurance and is, therefore, the most essential to be established. Consequently, procurement mission performance charts should not be used as the basis for these peacetime training mission profiles.

If there is one aspect with the largest possible variance, which will effect the damage incurred by an individual aircraft or by the entire fleet of a given system, it undoubtedly will be the kind of missions and the location or environment in which these missions are flown. Table 1 shows a very limited list of these mission variables.

It will not always be possible to fully cover all of the mission variables in a life evaluation. However, for those variables which have the greatest effect, a conscientious effort should be made to provide the closest estimate to what will occur. It will not always be possible to determine in every case the distribution of each mission parameter or the relative distributions between the parameters. However, a conscientious effort should be made to provide the closest estimate to what will occur. It is not unlikely that some missions, which will be flown, will be restricted to aircraft operating out of one base. Also it may be that only a limited percentage will operate out of a base, which due to its location, will produce a severe environment effecting the fatigue life of the given system. A prime example of this was the unusual and severe gust environment experienced in Korea. In these cases, when it is known that critical areas in the structural are predominantly effected by the varancies in the loads experienced, a separate evaluation of assigned aircraft should be made and submitted. These missions can therefore be deleted from the general fatigue evaluation. Since, the distribution of missions can have large variations between individual aircraft as can the parameters connected with a type mission, a further investigation should be made of the increase in damage rate for critical areas which are particularly sensitive to the increase in the mission distribution or mission parameter. This type of an evaluation should reasonably be expected to form an essential feature of any fatigue analysis. From this information the services can therefore be appraised of the operational aspects which are most damaging.

Although there will be a service load recording program monitoring a moderate percentage of each aircraft type, if any change to the mission planning or a shift of the system to another command is to be made, it is not only desirable but essential that the using command submit these to the cognizant engineering agency for evaluation of its effect on the projected vehicle life.

LOADING SPECTRUMS

Prediction of the loading experience, which is based on the established mission profiles, will always be difficult particularly for some of the major airframe components. Fortunately, the two major loading conditions, contributing to fatigue in transports and bomber classes, are probably the least difficult to predict. These are turbulence and the ground-air-ground cycle. Although, a deficiency of information on low altitude turbulence exists at the moment, the existing B-66 low altitude flight test project should now supply the needed data quickly. Where maneuver loads are critical, they will probably be the subject of the most debate. Past maneuver experience must be respected nevertheless for the knowledge it can yield. Where exact and applicable statistical data is not available, the distribution and the frequency of the maneuver levels relative to

Table 1

MISSION VARIABLES

MISSION OBJECTIVE	TRANSITION TRAINING	WEATHER NAVIGATIONAL NIGHT LANDING APPROACH
	NORMAL SERVICE TRAINING	
	BOMBER	HIGH ALTITUDE GROUND ALERT AIRBORNE ALERT
	FIGHTER	SCRAMBLE AIRBORNE ALERT GROUND SUPPORT
		ROCKET DELIVERY BOMB DELIVERY MISSILE DELIVERY
	COMBAT	
CONFIGURATION	WEIGHT	TAKE-OFF WT. NO. OF REFUELINGS LANDING WT. FUEL DISTRIBUTION & MANAGEMENT
	STORES	NUMBER & LOCATION
	CARGO	FUEL TO CARGO WT. RATIO CARGO DENSITY
MISSION PROFILE	SPEED	% OF TIME
	ALTITUDE	% OF TIME
	LANDINGS	NUMBER: FULL STOP TOUCH & GO
	WEAPON DELIVERY TECHNIQUE	
	DISTANCE OR FLIGHT TIME	
LOCATION OF BASE	CLIMATIC ENVIRONMENT GEOGRAPHICAL ENVIRONMENT	INFLUENCE ON TURBULENCE

the design maneuver value determined from reasonably similar types can be applied.

All statistical loading data available today presents information for the average airplane experience of a measured group. As was brought out earlier, the major interest is in determining the worst aircraft life. Knowledge of the deviation, that the worst airplane experience is greater than the average or mean airplane, is therefore an absolute necessity.

An analysis was made of some Air Force data to investigate the variance that individual aircraft of a measured group had to each other and to the composite average. These aircraft were operated concurrently out of Nellis Air Force Base and were subjected to the same routine operational flight planning. Figure 1 shows the average composite experience of 14 F-86F aircraft, indicated by the symbol delta (δ). The individual experience of eight of these 14 aircraft is indicated by the plotted points. The hours to reach or exceed (cumulative frequency) each of three load factor levels (3, 4, and 5 g's) is shown for the plotted aircraft and for the composite. The brackets shown inclose an interval within which 99 per cent[a] of the sampling variation would be expected to occur in estimating a "True" or general hours to exceed, if the samples were normal samples taken from one homogeneous maneuver load population. The fact that most of the individual aircraft plot outside of the bracket means that normal sampling variations are not sufficient explanation for the plots exceeding these estimated limits.

Figure 2 is a similar maneuver experience chart for the F-100A and compares 10 aircraft with the composite consisting of 19 aircraft. Figure 3 shows the gust experience for the same group of F-86F aircraft. It can be seen that many points plot outside of the 99 per cent limits.

An analysis of this same data converted to frequency of occurrence for values within specified loading intervals is shown in Tables 2 and 3. Ratios are given for the severest experience to the lowest and to the composite average aircraft experience. Looking at the maximum high to low ratios, the spread in experience is considerably greater at the higher load level than at the lower loads. The range in the maximum high to composite ratio is 1.3 to 2.7. To be more rational, the two high values were averaged, as were the two lows. These ratios are also shown. It is seen that the high to composite values are not significantly reduced. Although this limited analysis is not suggested as being conclusive, it does seem to validate a frequency factor of 1.5, which has been proposed by the British.[1] A proposal to range the frequency factor from 1.5 at the lowest load values to 2.0 or greater at the largest loads would appear to be entirely reasonable. Whatever the factor nevertheless, in order to predict the minimum airplane life, the frequencies of loadings obtained from averaged spectra must be increased to account for severe individual aircraft use or load experience. This holds even when all aircraft are subjected to identical mission profiles. Further study of this variance is intended to be made.

Another feature, which also contributes significantly to the loading spectrum for some components of the airframe, is the structural dynamic response to transient load inputs, where both magnitude and cycles of load are effected.

(a) Calculated from the composite probability using samples of size N equal to 1000. The smallest number of peaks for a plotted aircraft is about 1000.

Table 2

SCATTER IN MANEUVER EXPERIENCE
OF SINGLE AIRCRAFT

	Load Factor Range	Points in 99% Limits	Occurrences per Hour			Max Ratio		Ratio - Ave 2 Hi & 2 Lo	
			Com-posite	Max High	Max Low	Hi/Lo	Hi/Comp	2 Hi/2 Lo	2 Hi/Comp
F-86F[a]	3.0 - 4.0	1	10.9	15.5	5.1	3.0	1.3	2.1	1.3
	4.0 - 5.0	2	4.2	4.9	2.8	1.6	1.16	1.6	1.1
	5.0 - 6.0	1	1.4	2.9	.3	9.6	2.0	4.6	1.6
F-100A[b]	3.0 - 4.0	2	8.0	19.4	3.0	6.3	2.4	4.2	2.0
	4.0 - 5.0	2	2.7	5.2	.7	12.1	1.9	6.0	1.9
	5.0 - 6.0	5	.6	1.3	.07	58.4	2.2	16.0	2.0

(a) 8 out of 14 aircraft in the composite

(b) 10 out of 19 aircraft in the composite

Table 3

SCATTER IN GUST EXPERIENCE
OF SINGLE AIRCRAFT

F-86F - 8 A/C out of 14 A/C in composite

Gust Vel Range (U_{d_e})	Points in 99% Limits	Occurrences per Hour			Max Ratio		Ratio - Ave 2 Hi & 2 Lo	
		Com-posite	Max High	Max Low	Hi/Lo	Hi/Comp	2 Hi/2 Lo	2 Hi/Comp
10.2 - 17.0	4	9.52	14.0	3.1	4.5	1.5	3.2	1.4
17.0 - 23.8	6	.98	2.33	.47	5.0	2.4	2.5	2.1
23.8 - 30.6	8	.12	.32	.03	10.7	2.7	6.9	2.0

When power spectral dynamic response techniques are properly employed, rational accounting of magnitude and frequency of loading is achieved. When the dynamic response to discrete load inputs is used, the magnitude of loads may be reasonably correct, but the cycles of load at all points in the structure may not have been correctly assumed. For every major cycle of load that a wing root feels, an outboard wing station may count several or more major load cycles. Figure 4 shows the relative effects of the normalized bending strain power spectrum at an inboard and an outboard wing station.[(2)] It is seen that the contribution of the response at the higher frequencies, above 2 cps, to the total power spectrum is much larger at the outboard station than it is at the inboard station.

Table 4 shows the percentage that the root mean square value for the frequency range from two - ten cps contributes relative to the zero - two cps frequency range. For the inboard station the percentage is 27 per cent and a significantly large value of 71 per cent for the outboard station. To show the contribution at specific frequencies, a frequency interval of .2 cps was used at each frequency to determine the root mean square value and is shown in Table 5. The per cent that the RMS value at 1o cps is relative to the peak RMS value is four per cent for the inboard station and 40 per cent for the outboard station and this same situation is seen to exist also at the other lower frequencies shown. Due to the relatively high magnitude of load response at the higher frequencies significant damage increase can occur at the outer stations. Therefore, when employing a discrete load input, an increase in the number of loading cycles, when the response is relatively large at the higher frequencies, must be considered for all locations in the structure.

STRESS ANALYSIS

Further refinements of stress analysis calculations may be required to furnish additional information on extreme fibre stresses. Nominal stresses will, in many instances, not be sufficient to perform a fatigue analysis. This will be true for instance at spars, areas around large cut-outs, and other locations where there are abrupt changes in the direction of the load paths. Further comments will not be made, as the subject and the variance which could exist in the predicted result is well known.

S-N VALUES AND THE APPLICATION OF CUMULATIVE DAMAGE THEORIES

Since it is believed that some or most contractors will rely on the application of Miner's Theory in its basic form in spite of the more sophisticated procedures expounded, it is necessary to consider the statistical aspects and the variances associated therewith, when this theory is employed in life predictions. Mr. Butler of Boeing in his paper also covered this in much detail.

An analysis of some test data is presented in figure 5 showing the computed damage value at failure in relation to the groups to failure which represents the number of times a fixed spectrum was applied until failure.[(3)(4)] Mean S-n values were used. Large scatter in damage exists for low group numbers, which reduces as the group number increases. Note where the nominally accepted damage constant of 1.0 falls across the diagram. Since an infinite number of groups to failure is probably more representative of the loading history encountered by an aircraft, the information presented at a group number of 1,000 is of most interest.

Table 4

RELATIVE EFFECT OF FLEXIBILITY AT TWO WING STATIONS

	Range CPS	Approx Mean Sq Value	RMS	% of 0-2 CPS MS	% of 0-2 CPS RMS
	0 — 10	16.67	4.06	—	—
Station	0 — 2	15.50	3.94	—	—
54	2 — 10	1.16	1.08	7.5	27
	8 — 10	.067	.08	.4	2
	0 — 10	19.22	4.38	—	—
Station	0 — 2	12.74	3.57	—	—
572	2 — 10	6.48	2.55	51	71
	8 — 10	1.70	1.30	13	36

Table 5

CPS	RMS (a)		% of Peak RMS	
	Sta 54	Sta 574	Sta 54	Sta 574
.5	Peak 2.25	—	—	—
1	1.10	1.10	49	68
1.5	1.18	Peak 1.61	52	—
2	.33	.33	15	39
5	.085	.88	4	55
10	.085	.63	4	40

(a) For a Frequency Interval of .2 cps

It is believed that most of the variance in estimating damage, as indicated, is due to the scatter which is basically inherent in the S-n data. If this scatter were accounted for by the use of increased survival limits on the S-n values in lieu of the mean (50%) values, what Miner's damage constant would be better suited in prediction of the worst specimen?

To correct the mean S-n values, a normal distribution of the logarithm of the cycles to failure is used and a standard deviation equal to .176 (log 1.5) has been assumed. This standard deviation value has been proposed by the British in their most recent proposed fatigue requirements and is concluded to be typical for aluminum alloys.[1] This number seems rather reasonable as several sources indicate the number can vary from approximately .1 to 0.6.[5][6] In light of what will be shown, it is well to note that the new proposed joint Navy-Air Force Military Specification for Airplane Strength and Rigidity will require the use of 5% probability of failure S-n values.[7]

In table 6, the damage values for mean S-n data shown in the first column were corrected by the values shown at the bottom of the chart to get the damage values when using 5% and 1% probability S-n data.

To correct the mean S-n values the following procedure was used:

$$\text{Log } N_{5\%} = \text{Log } \bar{N} - 1.645\,\sigma$$

$$(\text{Assuming } \sigma = .176 = \text{Log } 1.5)$$

$$\text{Log } N_{5\%} = \text{Log } \bar{N} - 1.645 \text{ Log } 1.5$$

$$N_{5\%} = \frac{\bar{N}}{1.5^{1.645}}$$

$$\frac{n}{N_{5\%}} = \left(\frac{n}{\bar{N}}\right) \times 1.5^{1.645}$$

Also it has been assumed that the distribution in the damage value is log normal, since the scatter is to a large degree related to the basic variance in the S-n data.

To predict 2 5% probability of failure using mean S-n data the damage constant is .38; for a 5% probability using 5% S-n data the constant is .75; for a 1% probability using 1% S-n data the constant is also .75. Therefore, it appears justified to require a damage constant value of much less than 1.0 when predicting the worst specimen .

EFFECTS OF TEMPERATURE ON AIRCRAFT LIFE

Before completing the run down on a fatigue analysis, an additional item is interjected concerning the effects of temperature on aircraft life. Professor Gatewood in his paper made an excellent and comprehensive review of this subject. As he shows many new design modes will supplement present modes for unheated structures. Only a few of the 8 or 10 he outlined are covered. Since material property reductions are a result of both temporary and cumulative heating, it is

Table 6

VARIANCE OF $\Sigma \frac{n}{N}$ WITH SELECTION OF S-n VALUES

Based on 1000 Groups to Failure

	Value of $\Sigma \frac{n}{N}$ where N is based on		
	Mean S-n Values	5% Probability Values[a]	1% Probability Values[b]
Arithmetic Mean Damage (Approx)	.75	1.46	1.93
-1 σ (Normal)	.62	1.21	1.59
-1 σ (Log-Normal)	.50	.98	1.28
5% Probability (-1.645 σ)	.38	.75	.99
1% Probability (-2.33 σ)	.29	.57	.75
Lower Scatter Band	.50	.98	1.28

(a) Mean column x $1.5^{1.645}$ = mean x 1.95; assuming log-normal distribution

(b) Mean column x $1.5^{2.33}$ = mean x 2.57; assuming log-normal distribution

essential that an end of life or residual strength be provided. Creep-fatigue and thermal stress fatigue and all the other, Professor Gatewood has shown will supplement room temperature fatigue, and creep deformations will supplement elastic and possibly plastic deformations.

Figure 6 is used to illustrate the reduction in tensile strengths of 7075-T6 for various temperatures and times and since it is already sufficiently familiar it will not be discussed. But it does clearly illustrate both the temporary and the cumulative effects of temperature on tensile strength and the need for an end of life strength requirement.

In addition to the end of life strength consideration, the effects of creep and fatigue and how the two interact to cause failure must be considered in design. When considering fatigue in the design of room temperature structures, in general only stress magnitude and cycles are necessary to assess the life. But because of the metallurgical instabilities resulting from heating and loading with time, the material behavior is strongly influenced by the sequencing, magnitude, and duration of the applied load and thermal environment. In addition, the previous time, temperature, and load history will greatly effect the damage imposed by any particular heat and load application. Therefore, for heated structures, greater detailed consideration must be given to the true history of load, temperature, and time.

This discussion points up some of the added problems which confront the procuring agencies in providing adequate structural design criteria, and the designer in the selection of materials, operating stress levels, and, ultimately, the structural configuration.

Therefore, it is necessary that we add to our statistical loading knowledge information on load-temperature-time relationships.

DESIGN CRITERIA PHILOSOPHY

Without a criterion which sets the required level of safety or an acceptable probability of failure at the design life value, the fatigue analysis can not be completed.

In considering the choice of a standard, there are several alleviating conditions, which ease the need or desire for a large factor to be used in predicting the shortest life. For one, the laboratory test program will give a tremendous insight into what is required for an adequate structural inspection and retrofit program. Secondly, the service loads monitoring program should keep a reasonably close tab on the fleet service usage and load experience. These surveillance features offer considerable hope that the worst specimens will be caught prior to any failure.

To set the level of safety two criteria items are needed to be selected: values that the worst aircraft deviates from the mean or average aircraft load experience and fatigue strength; and a value for the probability of failure at the design life.

Let me close this presentation by saying that as of this moment the Air Force has taken no firm stand on either of these two items. However, a 5% probability of failure would appear to be the absolute minimum value acceptable for those structural items which are reasonably inspectable and provided that the life monitoring program as described is adhered to.

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Figure 1

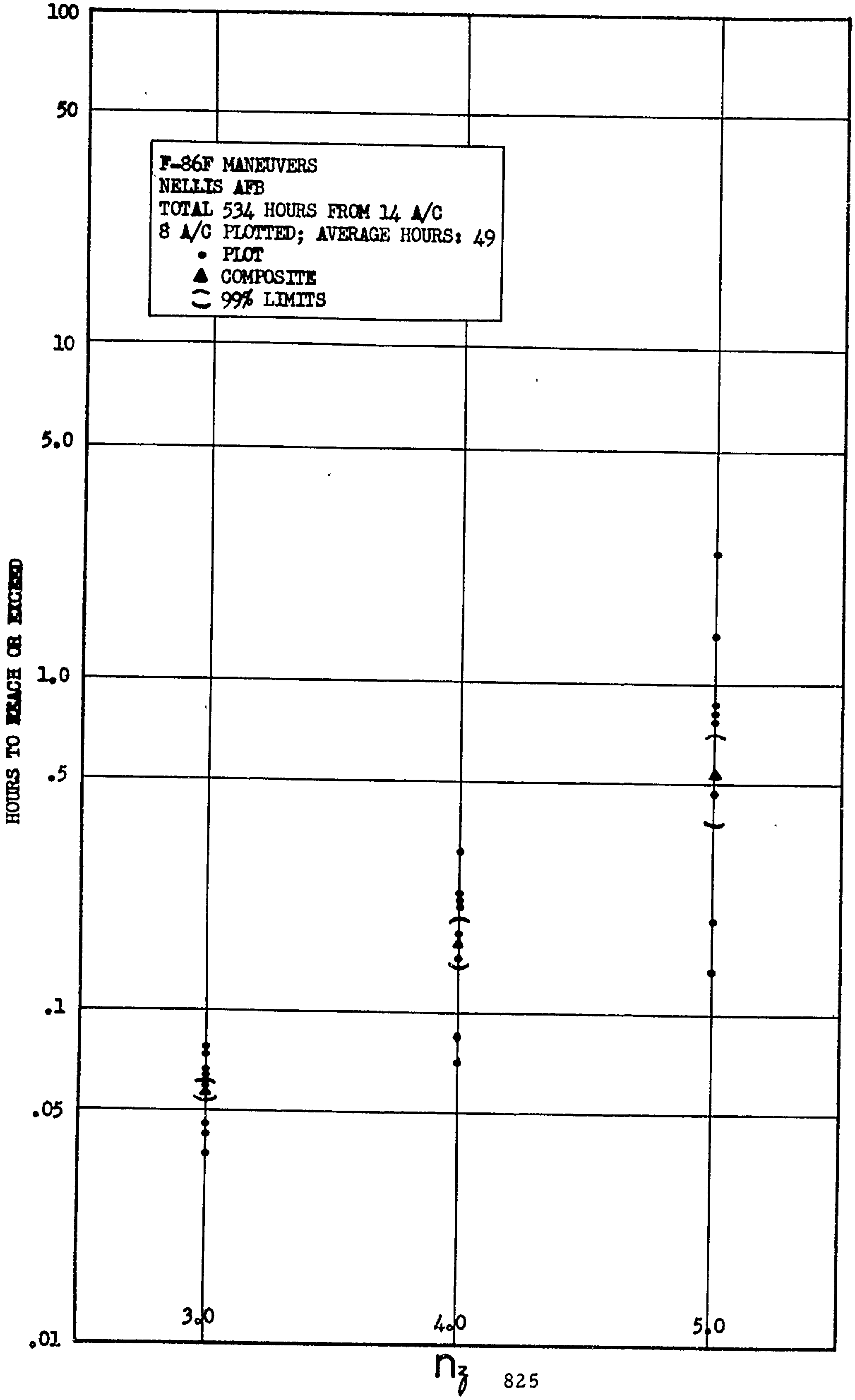
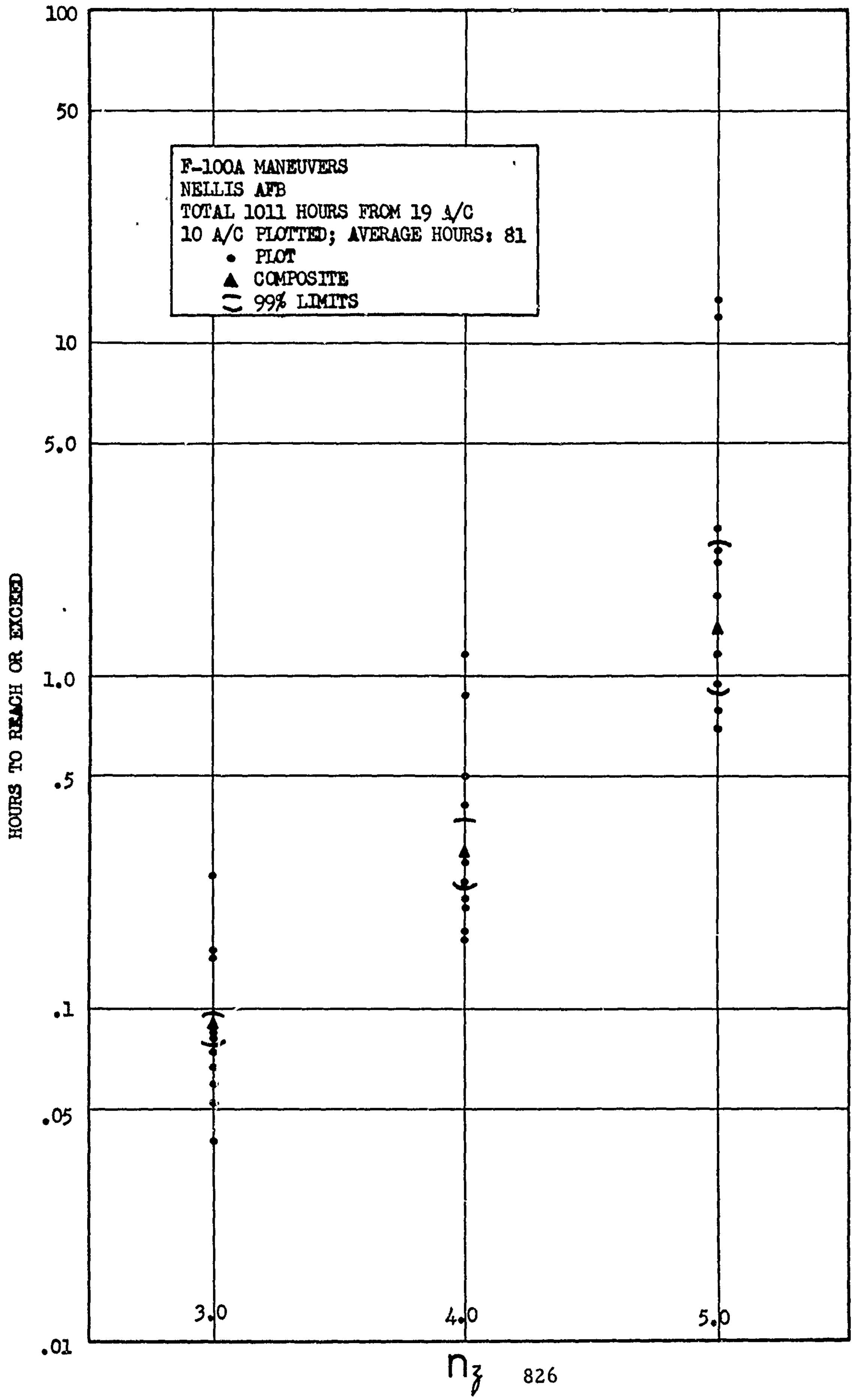
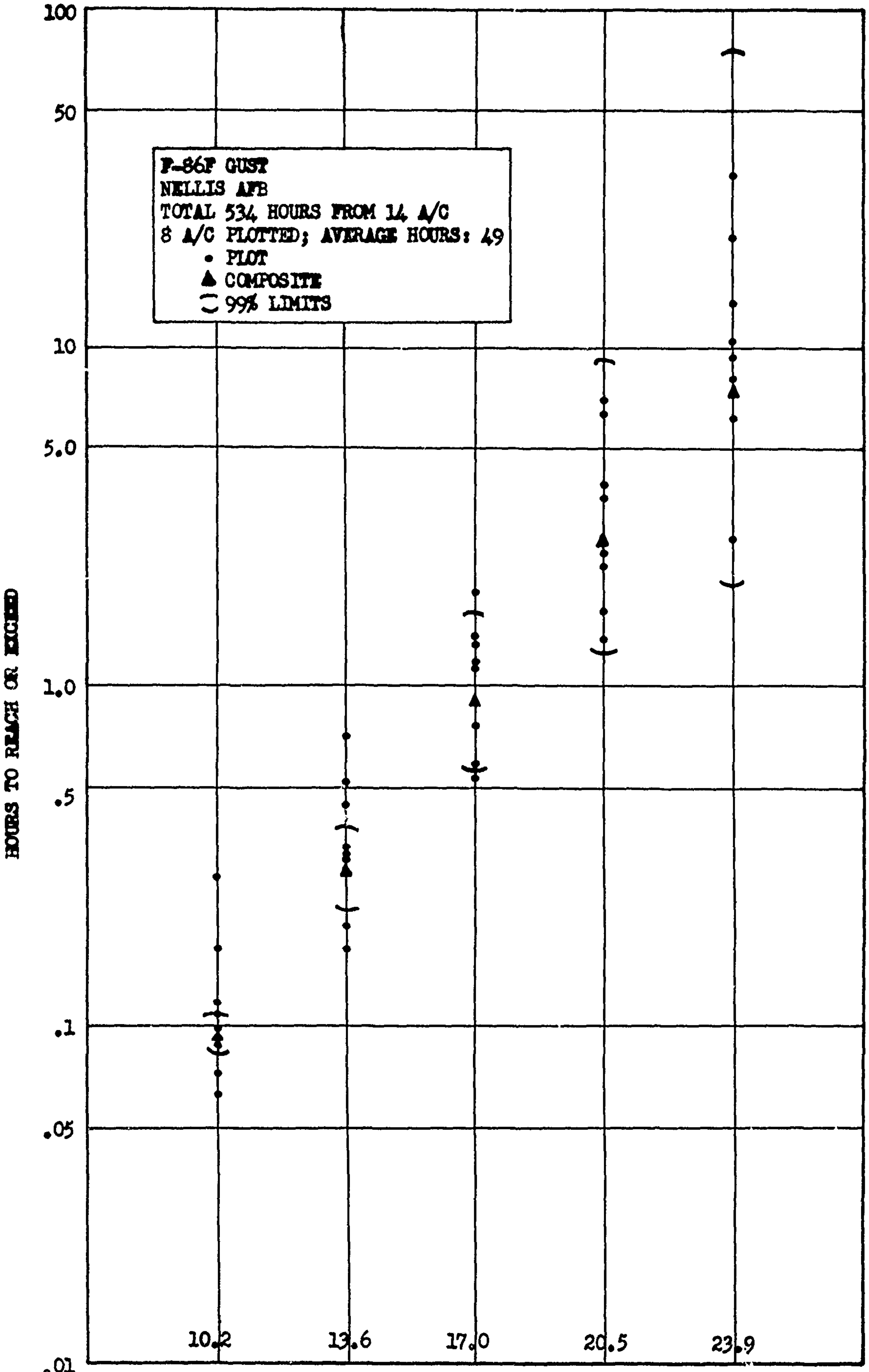

F-86F MANEUVERS
NELLIS AFB
TOTAL 534 HOURS FROM 14 A/C
8 A/C PLOTTED; AVERAGE HOURS: 49
• PLOT
▲ COMPOSITE
99% LIMITS
HOURS TO REACH OR EXCEED
100
50
10
5.0
1.0
.5
.1
.05
.01
3.0
4.0
5.0
n_z

Figure 2



100
50
10
5.0
1.0
.5
.1
.05
.01
HOURS TO REACH OR EXCEED
F-100A MANEUVERS
NELLIS AFB
TOTAL 1011 HOURS FROM 19 A/C
10 A/C PLOTTED; AVERAGE HOURS: 81
PLOT
COMPOSITE
99% LIMITS
3.0
4.0
5.0
n_z

Figure 3


F-86F GUST
NELLIS AFB
TOTAL 534 HOURS FROM 14 A/C
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• PLOT
▲ COMPOSITE
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HOURS TO REACH OR EXCEED
100
50
10
5.0
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17.0
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U_{d_e}

Figure 4

EFFECTS OF WING FLEXIBILITY

FRONT SPAR B-47

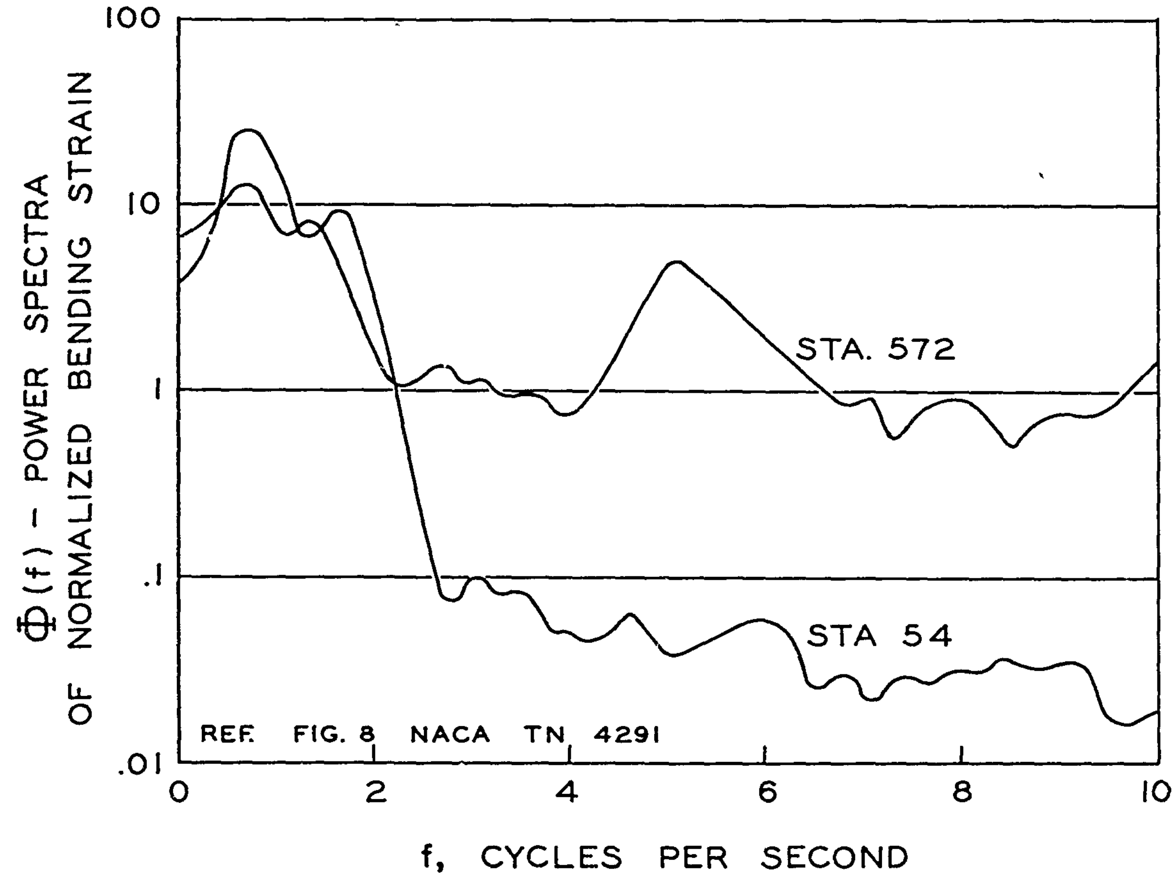


Figure 5

DAMAGE VS. GROUPS TO FAILURE

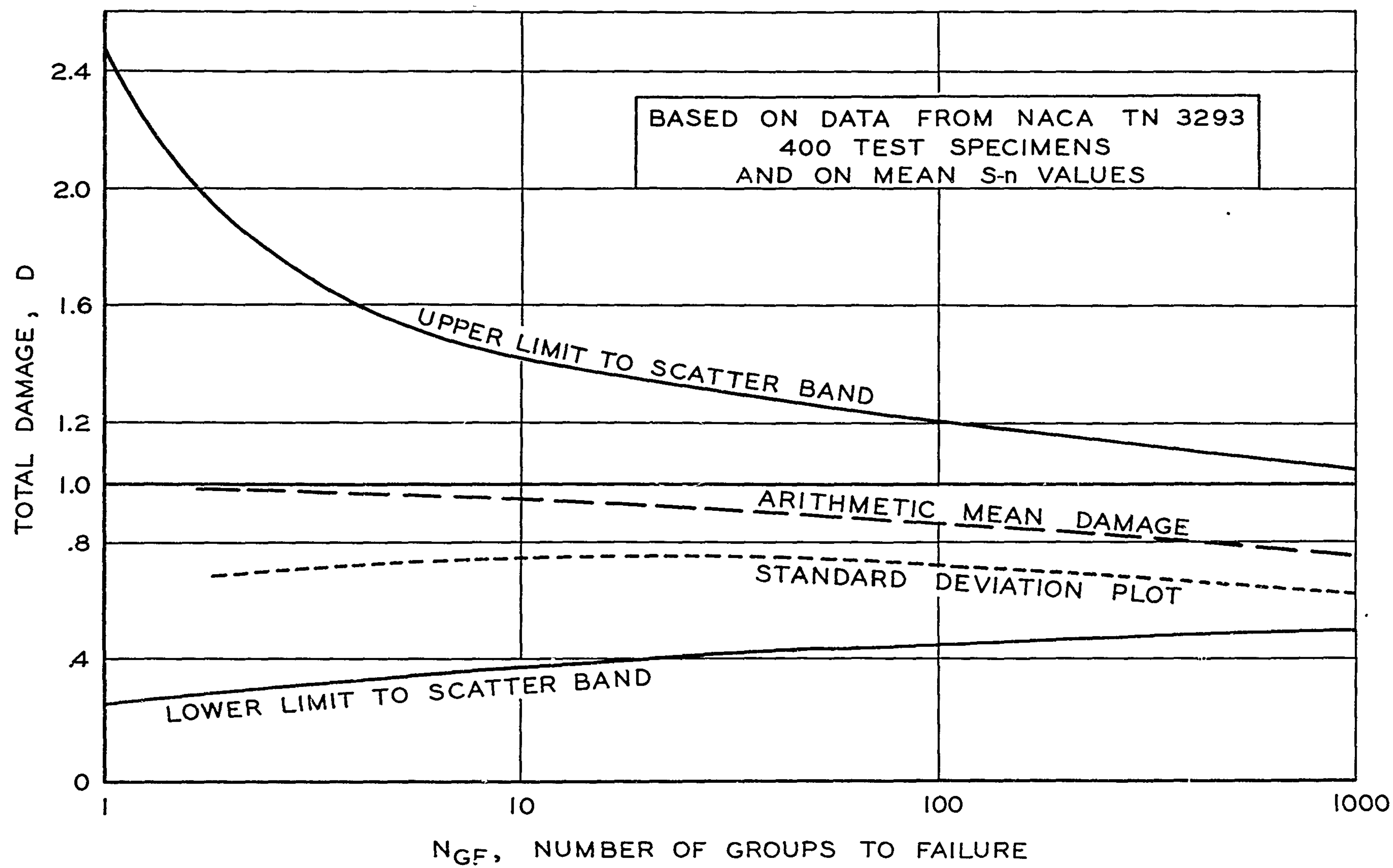
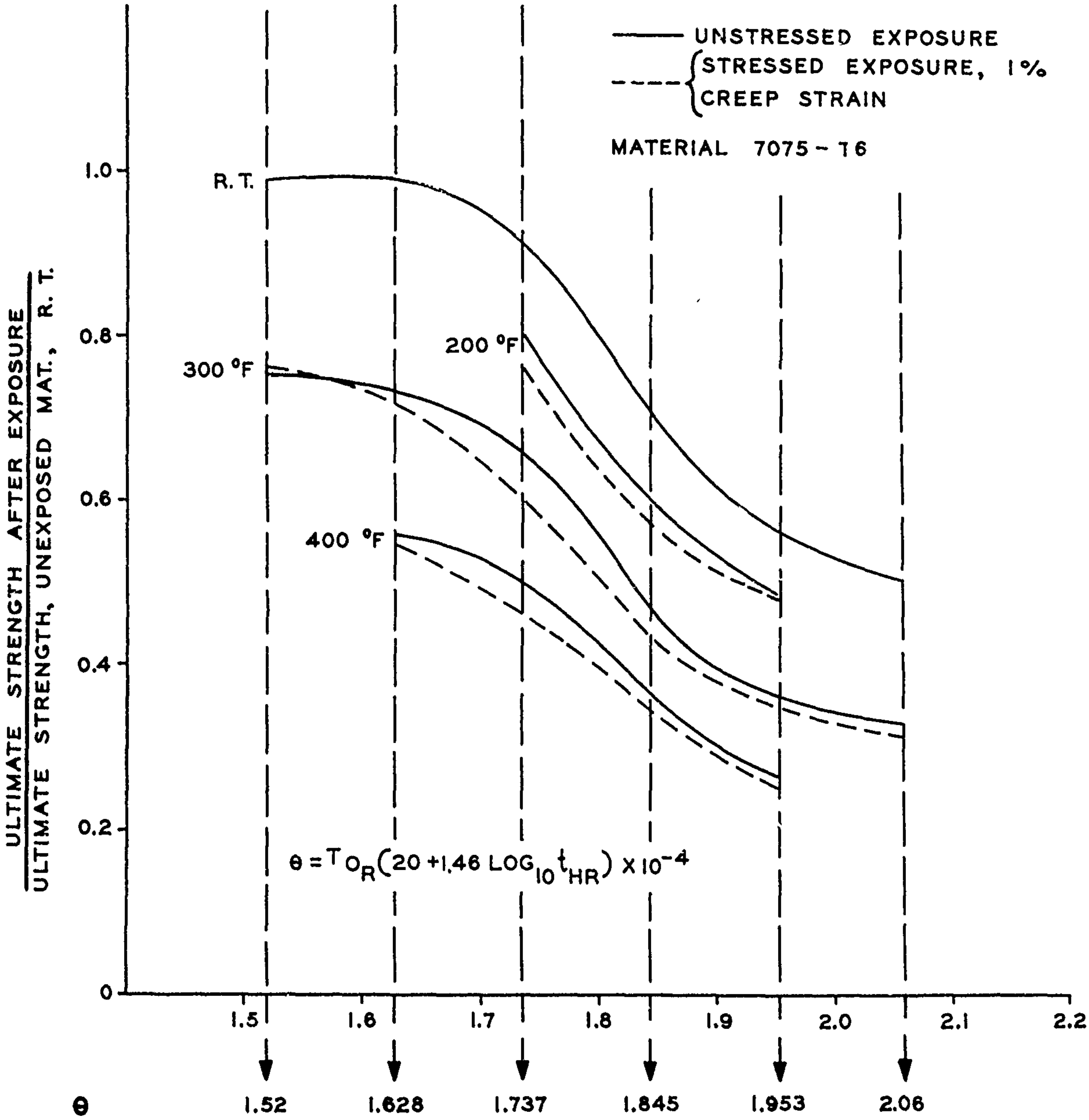


Figure 6

TIME - TEMPERATURE - LOAD EFFECTS ON RESIDUAL STRENGTH



TIME EQUIVALENT OF θ (HOURS)						
R. T.	10^6		10^9			
250 °F	10	10^2	10^3	10^4		
300 °F	1	10	10^2	10^3	10^4	
350 °F	.01	1	10	10^2	10^3	10^4

STRUCTURAL DAMPING AND ITS IMPORTANCE IN RESONANCE AND ACOUSTICAL FATIGUE

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The effect of system damping on the dynamic response of a mechanical system under periodic excitation, particularly near resonance is reviewed. Equations are presented for the resonance amplification factor involving the parameter D_0, the total damping energy dissipated by the system. The special case of random excitation, characteristics of acoustical fatigue, is discussed and the role of system damping explained. The various component parts of system damping are classified within the framework of (A) hysteric damping within the structural materials and (B) structural damping associated with (1) interface slip or Coulomb friction; and (2) shear strain in an adhesive layer at an interface. Each of these damping mechanisms is analysed to emphasize the factors important in the utilization of damping as an engineering property. Particular attention is paid to analytical concepts for maximizing the shear damping in an interface adhesive by design optimization procedures.

I. INTRODUCTION

The current trend towards lighter weight, reduced factors of safety, and higher speeds has greatly intensified the dynamic loading encountered by a structure in service. This same trend has also brought the natural frequencies of structural components and assemblies into a region where near-resonant vibrations are becoming increasingly prominent in service. The deleterious effects of the fatigue stress associated with near-resonant vibration has thus become a major cause for service failure.

In former years resonance conditions could often be avoided by properly controlling the natural frequencies in a system, that is by adjusting the stiffness and mass factors in the structure such as to safely separate the natural and the exciting frequencies. However, this approach is no longer effective in many situations, particularly in aero-space vehicles, because an excessive weight penalty is often involved. Furthermore, random excitation, either of mechanical or acoustical origin, is becoming more prevalent in service. For example, the typical noise spectrum shown in Figure 1 for the jet engine (1)* contains a range of frequencies which is wide enough to encompass most of the resonance frequencies encountered in an aircraft structure. Thus, whatever the natural frequency of the structural components and assemblies may be within the wide range of frequencies shown in Figure 1, resonant vibrations can still be excited.

Experiences in recent years have made it abundantly clear that resonant vibrations can no longer be avoided by clever design. Thus, modern structures, particularly aero-space vehicles, must be so designed as to withstand the resonant excitations characteristic of service. This paper is concerned with the factors involved in (a) the analysis of resonance amplification, (b) the fatigue stress associated with this condition (resonance fatigue), and (c) the importance of damping in the design of structures for high resonant fatigue strength. Since the factors important in conventional fatigue strength (stress concentration, loading range, etc.) are amply covered in other papers this paper shall be concerned with only those factors which are of unique interest in resonance fatigue.

By way of introduction and in order to clarify the role of damping in resonance fatigue the elementary resonance curves for a simple single-degree-of-freedom system excited by a simple <u>sinusoidal</u> force shall be reviewed. Referring to Figure 2, Mass M is attached to a

* Numbers refer to bibliography at end of paper.

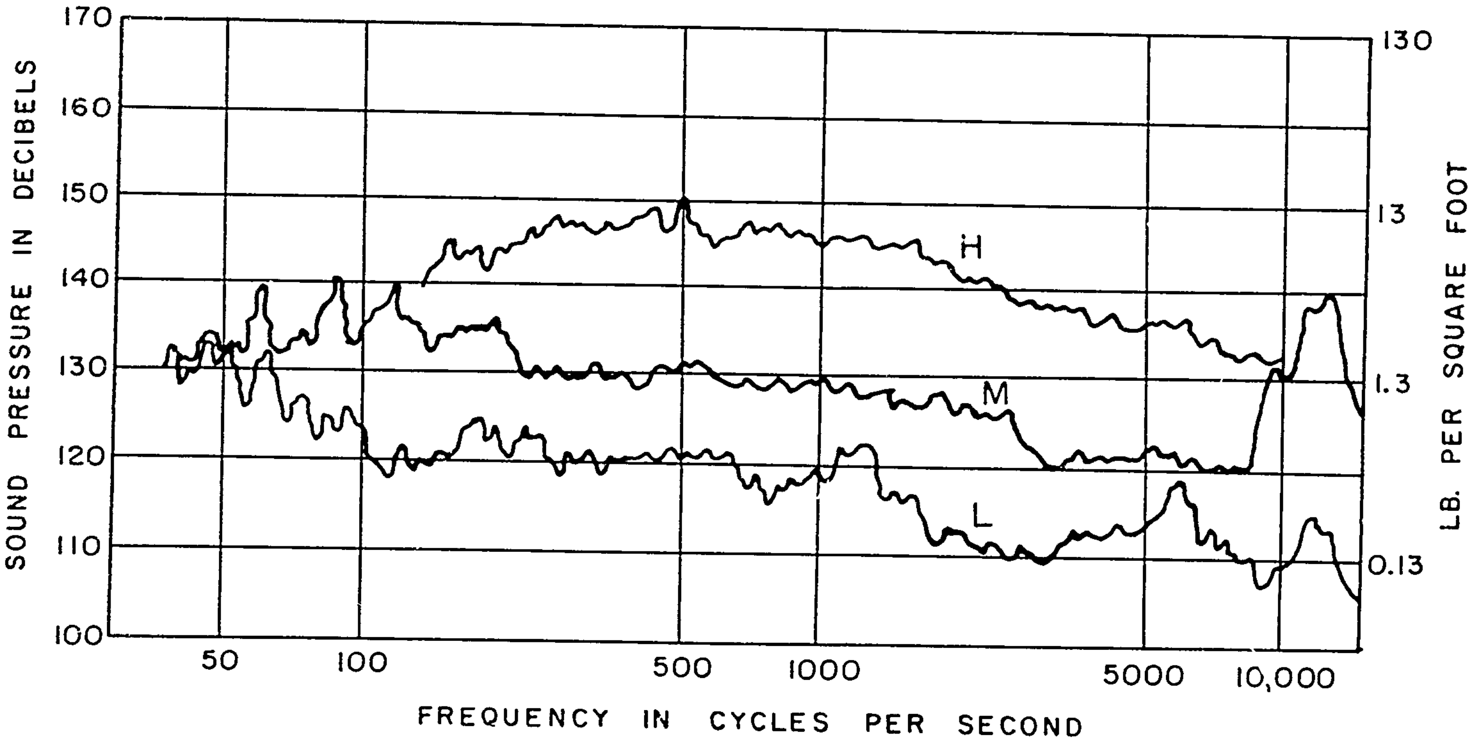


Fig. 1 Frequency Content of Typical Jet Exhaust Noise at High (H), Medium (M), and Low (L) Exhaust Velocities.

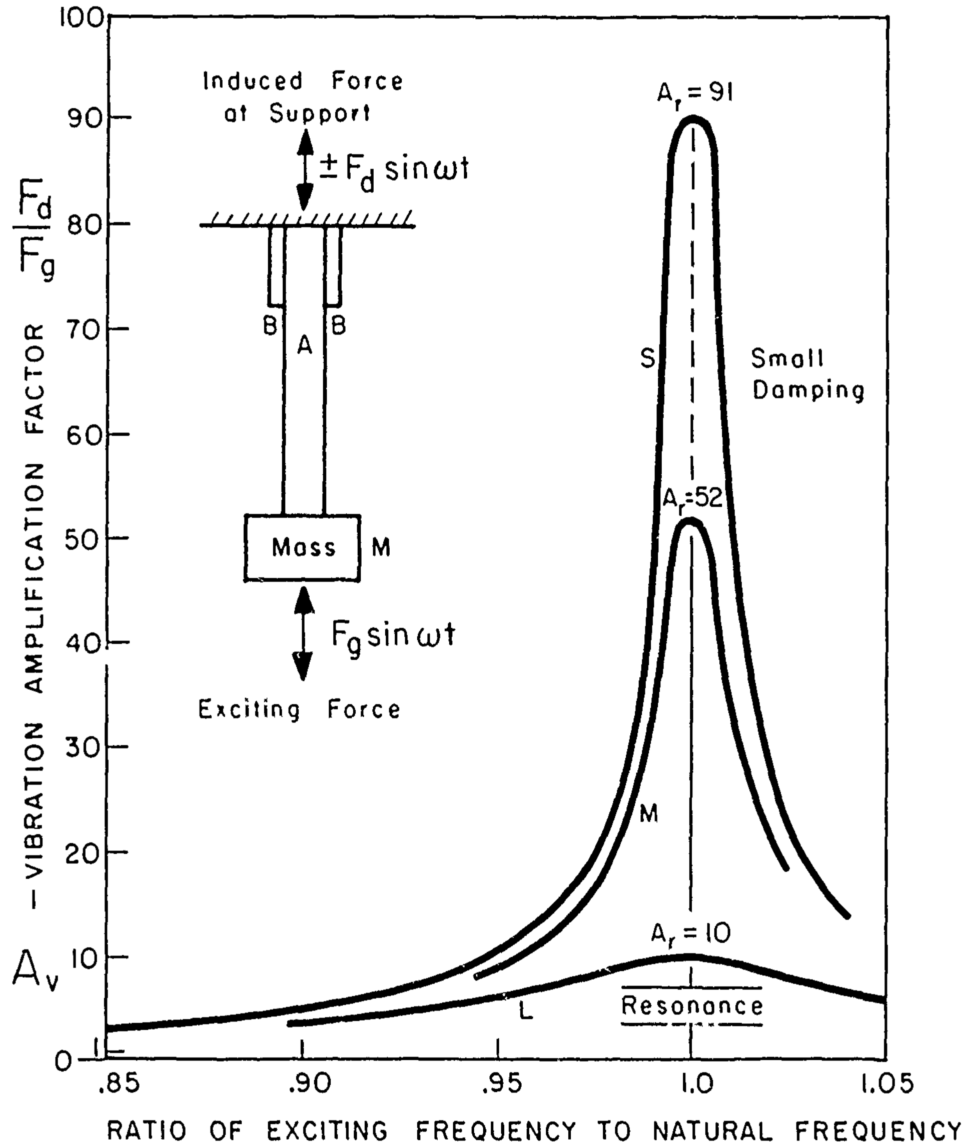


Fig. 2 Effect of System Damping on Vibration Amplification.

light structural member A which is held at its top by the support and cover plates B. The exciting force imposed on the mass is $F_g \sin \omega t$ and the force felt by the support, the induced force, is $F_d \sin \omega t$. A very low exciting frequencies ω, well below the resonant condition, induced force F_d is equal to or only slightly larger than exciting force F_g. However, as the exciting frequency approaches the natural frequency of the system (ratio of two frequencies shown in abscissa approaches unity) a vibration amplification is observed, the magnitude of which (A_v in Figure 2) depends on both the proximity to resonance and the damping in the system. The maximum amplification occurs at resonance when the vibration amplification factor A_v is called the resonance amplification factor A_r. Thus, under sinusoidal excitation:

$$A_v\Big]_{\phi = 90^\circ} = A_r = F_d / F_g \qquad \text{Eq. (1)}$$

where ϕ = vibration phase angle = 90° at resonance

For the case illustrated the resonance amplification factor A_r is 91 for a system with small damping ("S" curve), 52 for medium damping ("M" curve), and 10 for large damping ("L" curve). The great importance of the system damping in defining resonance amplification and controlling the associated resonance fatigue problems is thus apparent.

It can be shown (11) that for a linear system under sinusoidal excitation:

$$A_r = 2\pi \frac{W_e}{D_o} \qquad \text{Eq. (2)}$$

where W_e = total elastic or strain energy in the system partaking in the vibrations at its maximum deflection, in-lbs

D_o = total damping energy dissipated by the system per cycle of vibration, in-lbs per cycle

Combining equation (1) and (2):

$$F_d = 2\pi \frac{W_e}{D_o} F_g \qquad \text{Eq. (3)}$$

The above equation may be interpreted as follows in a fatigue analysis problem. Induced force F_d is the force actually felt by the structure (member A in this case) and provides a basis for the conventional fatigue analyses. That is, knowing the induced force and such structural details as shape of members, notch geometry, etc., one can compute the fatigue stress at the maximum locations and determine fatigue life. In the non-resonance problem the induced force is equal to the

exciting force F_g so the conventional fatigue analyses may be applied directly to the exciting force. However, in the resonance fatigue problem the exciting force may be greatly amplified since W_e / D_o in equation (3) may be a large number. Thus, in a resonance fatigue analysis of a system under sinusoidal excitation one is concerned not only with the conventional fatigue problems but also with the degree of amplification. As shown by equation (3) the total damping D_o in a system is an important parameter in such an analysis.

The total damping is equally important in a system under random excitation of acoustical origin. Even though such excitation is (a) produced by fluid pressure rather than structural forces, (b) not periodic, and (c) variable in amplitude at any discrete frequency (the resonant frequency) rather than of constant amplitude, the role of damping in controlling random resonant vibration is similar to that indicated above for periodic resonant fatigue.

In spite of its great importance in a variety of types of resonant fatigue problems damping has received relatively little attention to date. Several years ago a program of study on damping was initiated at the University of Minnesota, first on materials damping and more recently on structural interface effects. The purpose of this paper is to review some of the results of this study.

II. DEFINITION OF DAMPING AND CLASSIFICATION OF DAMPING MECHANISMS

The term damping as used in this paper defines the energy dissipation properties of a material or system under cyclic stress. In most cases a direct conversion of mechanical energy to heat is involved. This definition specifically excludes such energy transfer devices as dynamic absorbers or so-called dynamic "dampers". Within the context of this definition energy must be absorbed and dissipated within the specified system before the term damping is applicable.

Damping may be classified in various ways. For convenience in this paper damping is broken down into two major headings which shall be identified in this discussion as: (a) material damping and (b) system damping.

Material damping, sometimes called internal friction, internal damping, or hysteretic damping, is related to the energy dissipation in a volume of macro-continuous media. The term macro-continuous is intended to exclude the damping in a configuration originating at interfaces between recognizable parts, yet include the types of micro- and submicro-interface effects which might constitute an important mechanism in the volume or bulk damping of materials not homogeneous on a microscopic or submicroscopic scale. In general, material damping is associated with the energy dissipation which takes place when a more or less homogeneous volume is subjected to cyclic stress and the damping mechanisms are associated with the internal microstructure of the material. The two types of materials of interest in structural analyses to be discussed in this paper are (a) structural metals and non-metals and (b) viscoelastic adhesives.

Whereas material damping occurs in a volume of a macro-continuous medium system damping involves configurations of distinguishable parts or the interaction among various phenomena. Among the types of systems in which damping under cyclic stress may be important are:

a. Structural systems in which energy is dissipated in various types of joints, interfaces, or fasteners

b. Electro-mechanical systems in which energy conversion and dissipation may take place through the interaction between electrical or electromagnetic phenomena and physical bodies. Examples of this type of energy loss is the system damping associated with magnetic hysteresis and eddy currents.

c. Hydro-mechanical and acoustical systems in which damping occurs through fluid flow. Acoustical damping and radiation, oil flow through orifices, and dash pot effects are examples of this type of damping.

Since this paper is concerned primarily with structural damping further reference will not be made to electro-mechanical, hydro-mechanical or acoustical damping.

In order to clarify the types of damping of interest in aero-space vehicles it is desirable to ask the following question.

> Beyond the damping energy dissipated internally by the materials in the structure what is the principal mechanism by which structural joints dissipate energy?

The unique characteristic of a joint insofar as damping is concerned is its interfaces or mating surfaces which are maintained on contact. Thus, it is the damping associated with various interface effects that should receive close scrutiny.

For purposes of this paper two types of interfaces shall be identified: (a) a dry or lubricated contact surface (metal* to metal contact or metal to liner to metal contact), or (b) an adhesive type interface (metal to adhesive to metal joint). Next, let us consider the simple types of relative motions which can take place between mating surfaces at an interface to produce damping effects. A review of the general behavior of typical joints in typical load environments indicates that two types of relative motion should be considered: (a) a separation of mating surfaces (motions perpendicular to the interface) and (b) interface shear effects (relative motions in the plane of the interface). Of these two types of motion the one which appears to offer the greater potential for dissipating energy is relative shear at an interface. For example, if a member A in the system shown in Figure 2 is vibrated axially the cyclic axial strain in member A will tend to produce a cyclic shear displacement (slip) at the interfaces between member A and cover plates B. This slip provides a mechanism for dissipating energy.

For the case of dry interfaces (metal to metal contact) coulomb friction provides the mechanism for dissipating energy under cyclic shear displacement. Several types of joints have been investigated considering this mechanism. It is found, as discussed later that this mechanism is capable of dissipating very large energy if the joint design is properly optimized (5) (6) (7).

It is generally found, however, that a design optimized for maximum dry slip damping may sometimes develop serious fretting and corrosion effects in the interface regions subjected to large cyclic slip. Such interface surface deterioration may not only cause the joint to drift from optimum conditions but may also initiate fatigue cracks, the very thing that high damping in a system exposed to resonant vibration is intended to mitigate. In short the cure may lead to a condition worse than the original disease.

*The term "metal" is used here for conciseness. Actually the term structural material, which includes not only metals but plastics, ceramics, etc. is implied.

In view of this difficulty with dry interface slip consideration was given to the possibility of lubricated interfaces (2) and inserts of plastic and other types of "non-fretting" materials at interfaces (3). However, it was still difficult to avoid fretting and maintain optimum conditions for maximum damping in practical joints which dissipate large damping. This led to the consideration of an adhesive type interface of sufficient thickness to permit the relative shear motions between metal surfaces to be absorbed as shear strain within the adhesive itself (no relative motion between adhesive and the metal adherents). This is the principle of sound deadening tape, which has been in use for about a decade. However, there was until revently very little interest in the utilization of adhesive damping in structural joints. The potential contribution of this approach became more apparent recently when it was found that the damping capacity of viscoelastic adhesives (as a material) in shear is very large as discussed later. Furthermore, it was also found that viscoelastic adhesives can withstand very large cyclic shear strain without deterioration. This combination indicates that a properly optimized adhesive joint (thickness, for example, adjusted properly in accordance with its stiffness and other properties) can dissipate very large damping energy. Furthermore, this damping mechanism not only avoids corrosion and fretting problems but also simplifies the problem of maintaining optimum conditions during service. This damping mechanism and its use and optimization in systems or configurations is discussed later.

III. MATERIAL DAMPING

Materials are not perfectly elastic even at very low stress levels. Inelasticity in materials manifests itself in a variety of different ways. Under cyclic stress, for example, inelastic behavior takes the form of a stress-strain hysteretic loop, such as that illustrated in Figure 3. Although such loops are always present at stress of engineering interest they are often too narrow to be observed by conventional methods. The shape of the loop depends on the damping mechanism operative.

Many different types of inelastic mechanisms and hysteretic phenomena have been identified. For purposes of this paper the damping classification given in Table I is appropriate. Referring to the main heading of Table I the various damping phenomena and mechanisms may be identified under two main headings: "dynamic hysteresis" and "static hysteresis". Dynamic hysteresis is discussed first.

Dynamic hysteresis is sometimes identified as viscoelastic, rheological, and rate-dependent hysteresis. It is observed in materials having essentially linear stress-strain laws which are describable by a differential equation containing stress, strain, and time derivatives of stress or strain. These differential equations need not be linear. Furthermore, they can include terms which allow for permanent set such as OB after cycle OAB in Figure 3.

One important type of dynamic hysteresis, a special case which Zener (8) has labeled anelasticity, does not include provisions for permanent set after a long time. This means that if the load is suddenly removed at point B in Figure 3, after cycle OAB, strain OB will gradually reduce to zero as the specimen recovers (or creeps negatively)from point B to point O.

The terms anelasticity and internal friction (damping in anelastic materials) have been reasonably well accepted by physical metallurgists for over a decade. However, the more general types of damping identified here as viscoelastic and rheological hysteresis do not have a well-accepted name.

A distinguishing characteristic of internal friction and the more general case of viscoelastic damping is its dependence on time derivative effects. Thus, the hysteresis loops tend to be elliptical in shape rather than pointed as in Figure 3. Furthermore, the loop area is definitely related to the dynamic or cyclic nature of the loading and the area of the loop is dependent on frequency. In fact, the stress-strain curve for an ideally viscoelastic material becomes a single value curve (no hysteretic loop) if the cyclic stress is applied slowly enough to allow the material to be in complete equilibrium at all times (oscillation period very much longer than relaxation times). Thus, no hysteretic damping is produced by these mechanisms if the material is subjected

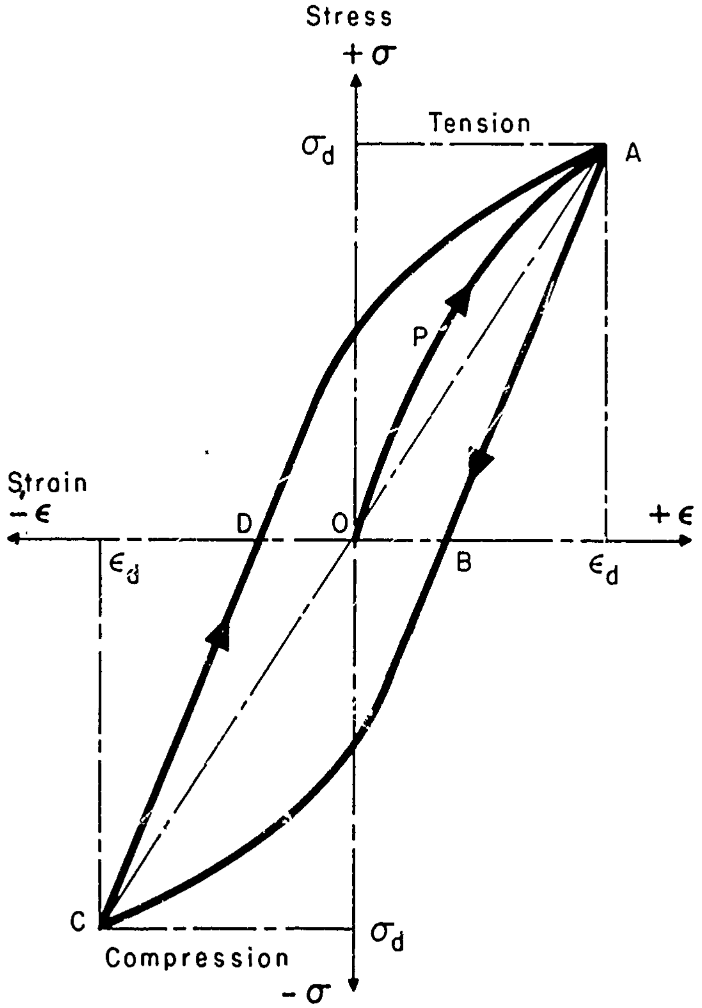
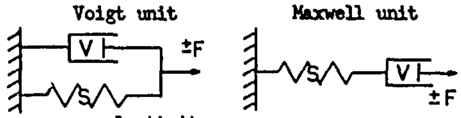
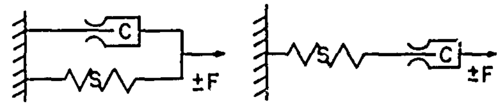


Fig. 3 Typical Stress-Strain Hysteresis Loop for a Material Under Cyclic Stress.

TABLE I CLASSIFICATION OF TYPES OF HYSTERETIC DAMPING OF MATERIALS

	TYPES OF MATERIAL DAMPING		
Name Used Here	DYNAMIC HYSTERESIS	STATIC HYSTERESIS	
Other Names	Viscoelastic, rheological, and rate-dependent hysteresis	Plastic, plastic flow, plastic strain and rate-independent hysteresis	
Nature of Stress-Strain Laws	Essentially linear. Differential equation involving stress, strain, and their time derivatives	Essentially nonlinear, but excludes time derivatives of stress or strain	
Special Cases and Description	<u>Anelasticity</u>. Special because no permanent set after sufficient time. Called "internal friction"		
Simplest Representative Mechanical Model	Voigt unit Maxwell unit  anelasticity		
Frequency Dependence	Critically at relaxation peaks	No, unless other mechanisms present	
Primary Mechanisms	Solute atoms, grain boundaries. Micro- and macro-thermal and eddy currents. Molecular curling and uncurling in polymers.	Magnetoelasticity	Plastic strain
Value of "n" in $D = JS^n$	2	3 - up to coercive force	2-3 up to σ_L 2 to >30 above σ_L
Variation of η with Stress	No change, since n-2=0	Proportional to σ since n-2=1	Small incr. up to σ_L Large incr. above σ_L
Typical Values for η	Anelasticity: < .001 to .01 Viscoelasticity: < 0.1 to >1.5	0.01 to 0.08	.001 to .05 up to σ_L .001 to >0.1 above σ_L
Stress Range of Eng. Importance	Anelasticity - low stress Viscoelasticity - all stresses	Low and medium. Sometimes high	Medium and high stress
Effect of Fatigue Cycles	No effect	No effect	No effect up to σ_L Large changes above σ_L
Effect of Temperature	Critical effects near relaxation peaks	Damping disappears at Curie Temp.	Mixed. Depends on type of comparison
Effect of Static Preload		Large reduction for small coercive force	Either little effect or increase

to essentially static loading. Stated differently, the static hysteresis is zero.

"Static hysteresis", in contrast with dynamic hysteresis, involves stress-strain laws which are insensitive to time, strain or stress rate, or other derivatives. Thus, in a material in which static hysteresis dominates the value of strain is attained almost instantly for each value of stress and prior stress history (direction of loading, amplitudes, etc.), independent of loading rate. Under cyclic loading pointed loops similar to Figure 3 are formed and if the stress is reduced to zero (point B) after cycle OAB, then OB remains as a permanent set or residual deformation. Furthermore, the shape of the hysteretic loop is independent of frequency.

The two principal mechanisms which lead to static hysteresis are magnetostriction and plastic strain. Thus, Table I shows two headings under static hysteresis, not only to identify the mechanisms operative, but also to provide a name for describing their characteristic behavior discussed later.

Table I also shows the simplest representative mechanical models for each of the behaviors classified. In these models S is a spring having linear elasticity (linear and single-valued stress-strain curve), V is a linear viscous dashpot which produces a resisting force proportional to velocity, and C is a coulomb friction unit which produces a constant force whenever slip occurs within the unit, the direction of the force being opposite to the direction of motion. More sophisticated models have been found to predict reliably the behavior of some materials (9), particularly polymeric materials.

The various inelastic mechanisms indicated in Table I have been discussed in prior publications (8) (9) (10) and will not be reviewed in this paper. However, in order to facilitate comparisons to be made later with other damping mechanisms it is desirable to indicate magnitudes of damping which can be expected from structural materials. Such data are given in the bottom half of Table I, which is self explanatory. The units and symbols used are defined below

σ_d = induced stress in material; psi

D_d = specific damping energy of a material at induced stress σ_d; in-lbs per cu-in per cycle

E = modulus of elasticity; psi

η = loss factor of material $= \dfrac{E}{\pi}\,\dfrac{D}{\sigma_d^2}$ Eq. (4)

σ_e = fatigue strength of the material; psi

σ_L = a limiting stress approximately 80% of σ_e; psi

Various criteria for comparing the damping properties of material have been used (10) (11). It has been shown that a convenient and significant method for comparing the damping properties of structural materials is to plot damping energy D_d versus ratio of induced stress to fatigue strength (ratio σ_d/σ_e). Such a plot is shown in Figure 4 for a variety of common structural materials. For the large group of structural materials which were not particularly selected for high damping (excludes materials having large magnetoelastic or plastic strain damping) and for a variety of test conditions (10) the data lie within a fairly well established band shown in Figure 4. The approximate "geometric" mean curve is shown as the "dot-dash"in Figure 4. Even though this is a two segment line it can be defined with sufficient accuracy by the following single equation:

$$D = (\sigma_d / \sigma_e)^{2.3} \qquad 6 (\sigma_d / \sigma_e)^{8} \qquad \text{Eq. (5)}$$

Also shown in Figure 4 for comparison purposes are four materials having especially high damping. Materials 1 and 2 are the magnetoelastic alloys Nivco 10 and 403 alloy. Nivco 10 (14) retains its high damping up to the stresses shown (data not available at higher stresses). However, the 403 alloy reaches its magnetoelastic peak at a stress ratio of approximately 0.2 and increases less rapidly beyond this point, up until plastic strain damping becomes dominant (at stress ratio of approximately 0.8), beyond which damping increases very rapidly. By contract, material 3 , a Manganese copper alloy (12) with large plastic-strain damping, retains its high damping up to and beyond its fatigue strength. Material 4 is a 'typical' viscoelastic adhesive (3M tape No. 466) for which a cyclic shear strain of unity is assumed to lie within the fatigue strength (experiments show that a shear strain well above unity does not cause deterioration in the adhesive even after millions of cycles).

Since these damping data are plotted to a logarithmic scale the superiority of the high damping materials is not dramatically revealed by Figure 4. However, Nivco 10, for example, has a damping thirty times as large as the average structural material in the stress range shown in Figure 4, and the viscoelastic damping is over ten times as large as Nivco. These observations on viscoelastic adhesives will be utilized later.

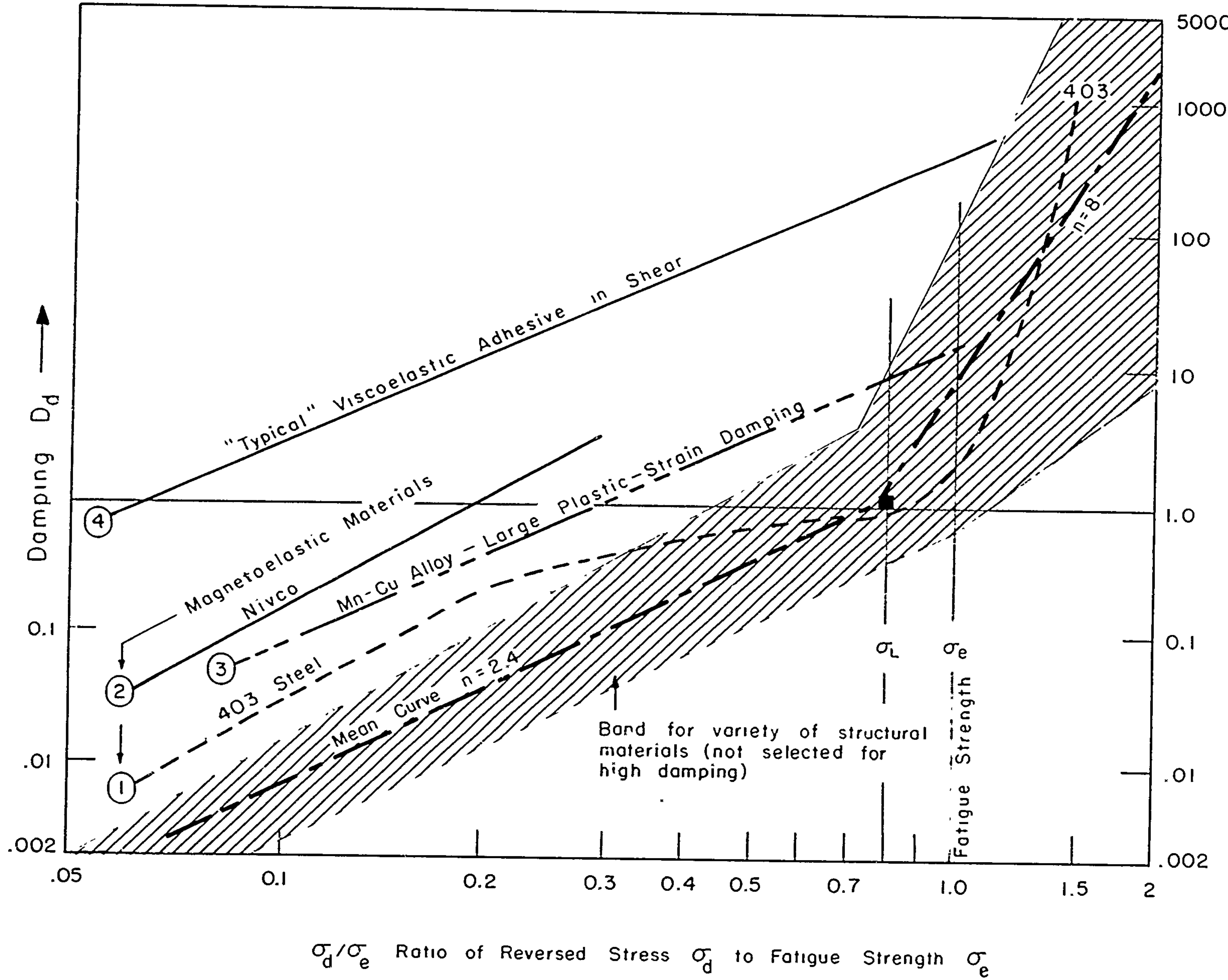


Fig. 4 Range of Damping Properties for a Variety of Structural Materials.

IV. SYSTEM DAMPING AND COMPARISONS WITH MATERIALS DAMPING

1. Interface Slip

It has been shown both theoretically and experimentally (5)(6)(7) that interface slip has the potential for dissipating large damping energy. Still, engineering designers have made little or no use of slip damping. This has been due partially, and in many cases justifiably, to the fear of fretting and fretting fatigue which may in some cases be a greater evil than low damping. However, in many cases the failure to realize the potential in slip damping is due to unfamiliarity with its basic nature and design parameters. Although some slip damping is present in practically all structures this damping mechanism has not been adequately studied in the past.

Reasonably general equations for slip damping have been developed recently (6). However, since the purpose of this paper is to convey general damping concepts for engineering guidance only one special case shall be considered to illustrate the important parameters.

The case to be discussed is the bileaf cantilevel beam AB under uniform pressure P as shown in Figure 5. At very small values of exciting force F (or at very large interface pressure P) the friction at the interface between beams A and B is sufficient to prohibit slip, and the bileaf behaves as though A and B were one solid beam without an interface. However, if the exciting force is increased (or if pressure P is reduced) slip will occur causing an offset at the end of the beams labeled S. Under cyclic exciting force, slip S at the end of the beam and also along the entire interface will also be cyclic and provide a mechanism for dissipating damping energy. This energy is a function of both the slip and shear stress at the interface. Under large pressure P the shear stress is large but slip small; whereas under small pressure P the shear stress is small but the slip is large. Since damping energy depends on a product of a shear stress function and a slip function, maximum damping will occur at some intermediate or optimum pressure. The equations for defining the damping energy dissipation associated with this interface slip have been developed and are plotted in Figure 5 (b) for one special joint considered theoretically and experimentally (5). This figure shows the total damping D_o^s due to interface slip only as a function of interface pressure P and ratio of induced stress σ_d in the beam to its fatigue strength σ_e. The maximization of damping at an optimum pressure of 80 psi is clearly shown in this figure. Another feature of this figure (to be contrasted later with material damping) is that the damping does not change abruptly with stress.

The relative magnitudes of slip and material damping cannot generally be compared directly due to the basic differences in the mechanisms involved. The material damping in a part depends not only on its material composition and constitution but also on stress amplitude, the volume-stress function of the part, and other factors discussed in prior

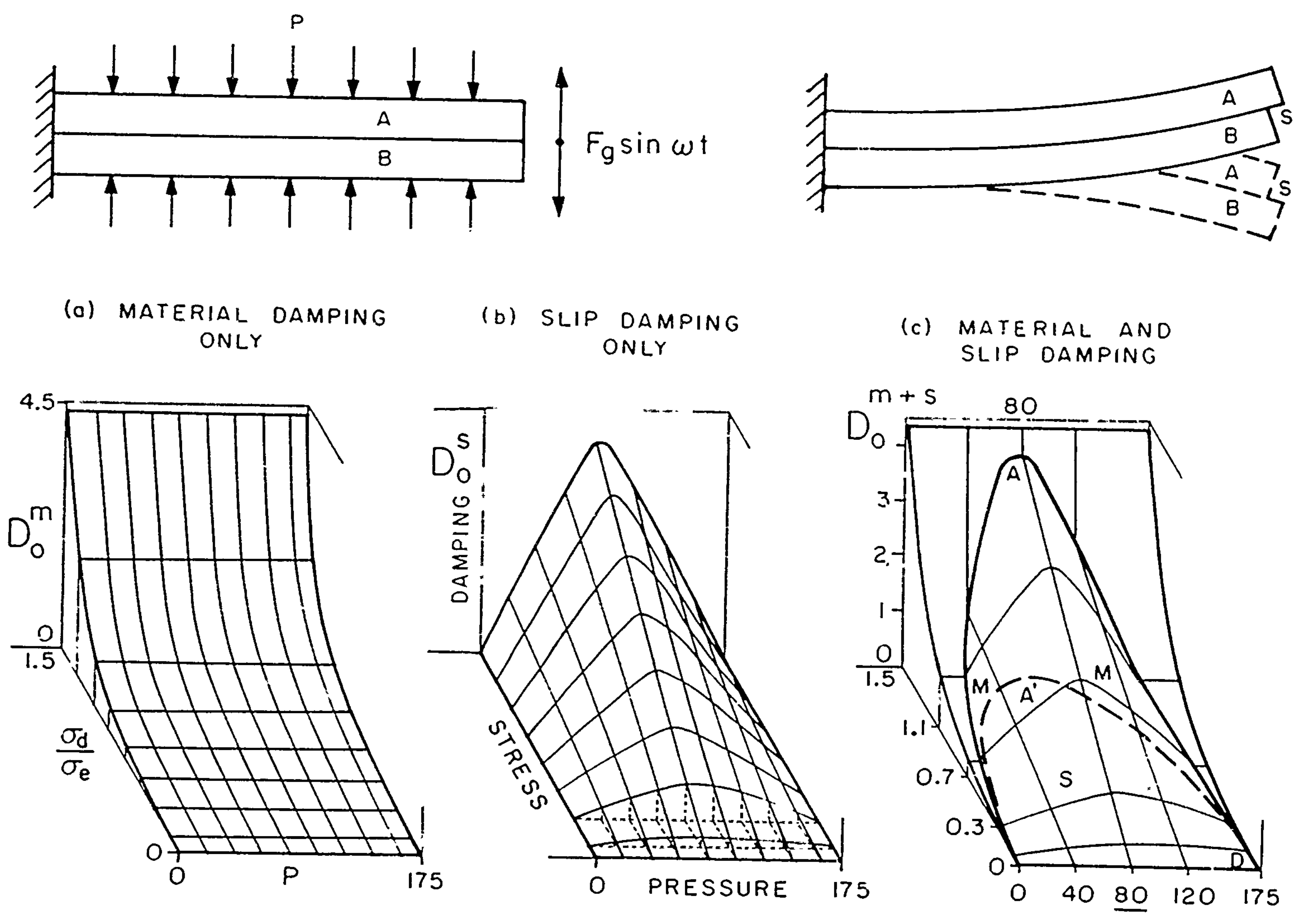


Fig. 5 Material and Slip Damping of a Bileaf Cantilever Beam Under Uniform Interface Pressure.

publications (10). Slip damping at an interface depends on another set of parameters, which include the coefficient of friction, pressure, shear stress, and strain distribution. Furthermore, material damping occurs throughout the volume of a part (a volume integral), whereas slip damping occurs at an interface surface only (a surface integral). Thus, the two types of damping are not directly comparable. Parts and joints can easily be conceived in which either slip or material damping will dominate. Nevertheless, in order to clarify the important parameters in damping it is desirable to make simplified comparisons between material and slip damping. In these comparisons only two important parameters shall be considered, interface pressure and maximum stress. Other factors such as member geometry, coefficient of friction, and types of vibration, are held constant.

For comparison purposes material damping for the same beam assembly, computed for a "representative" material defined by Eq. (5) (also see Figure 4) is shown in Figure 5 (a). The main features this figure has are: (a) damping increases rather abruptly at a stress in the vicinity of the fatigue limit, and (b) interface pressure does not significantly affect material damping.

Both material and slip damping are plotted in Figure 5(c), the line OAD indicating the intersection between the two surfaces. The projection of this line on the basal plane OA'D indicates the combinations of stress and pressure in which each type of damping dominates; in region S at near optimum pressure and at intermediate stress the slip damping is the larger, whereas in region M at extremely high or low pressure or at high stress the material damping dominates.

A second criterion for judging resonance behavior is resonance amplification factor A, which varies inversely with damping as shown in Equation 2. Considering material damping only A_r^m decreases rapidly with increasing stress as shown in Figure 6(a), from 300 to 10 percent of the fatigue strength, to 12 at 150 percent of the fatigue strength. Interface pressure is not a significant factor in material damping, as observed previously. Considering next slip damping alone both interface pressure and induced stress are important factors as shown in Figure 6(b). A minimum A_r^s is attained under the optimum pressure of 80 psi and at low induced stress. If both material and slip damping are included in the computation, then the resultant resonant amplification factor curve A_r^{m+s} combines the features of both curves as shown in Figure 6 (c).

Finally, a criterion based on the permissible exciting force F_g (see Equation33) which produces various amplitudes of cyclic stress at resonance is shown in Figure 7. In order to place the F_g criterion on a dimensionless basis, the ratio $\frac{F_g}{F_e}$ is plotted, F_e being that alternating force which if applied statically will produce a stress σ_e (the fatigue strength) in the beam. Thus, F_e is an indication of the conventional fatigue strength of the beam and F_g at a value of $\sigma_d/\sigma_e = 1$

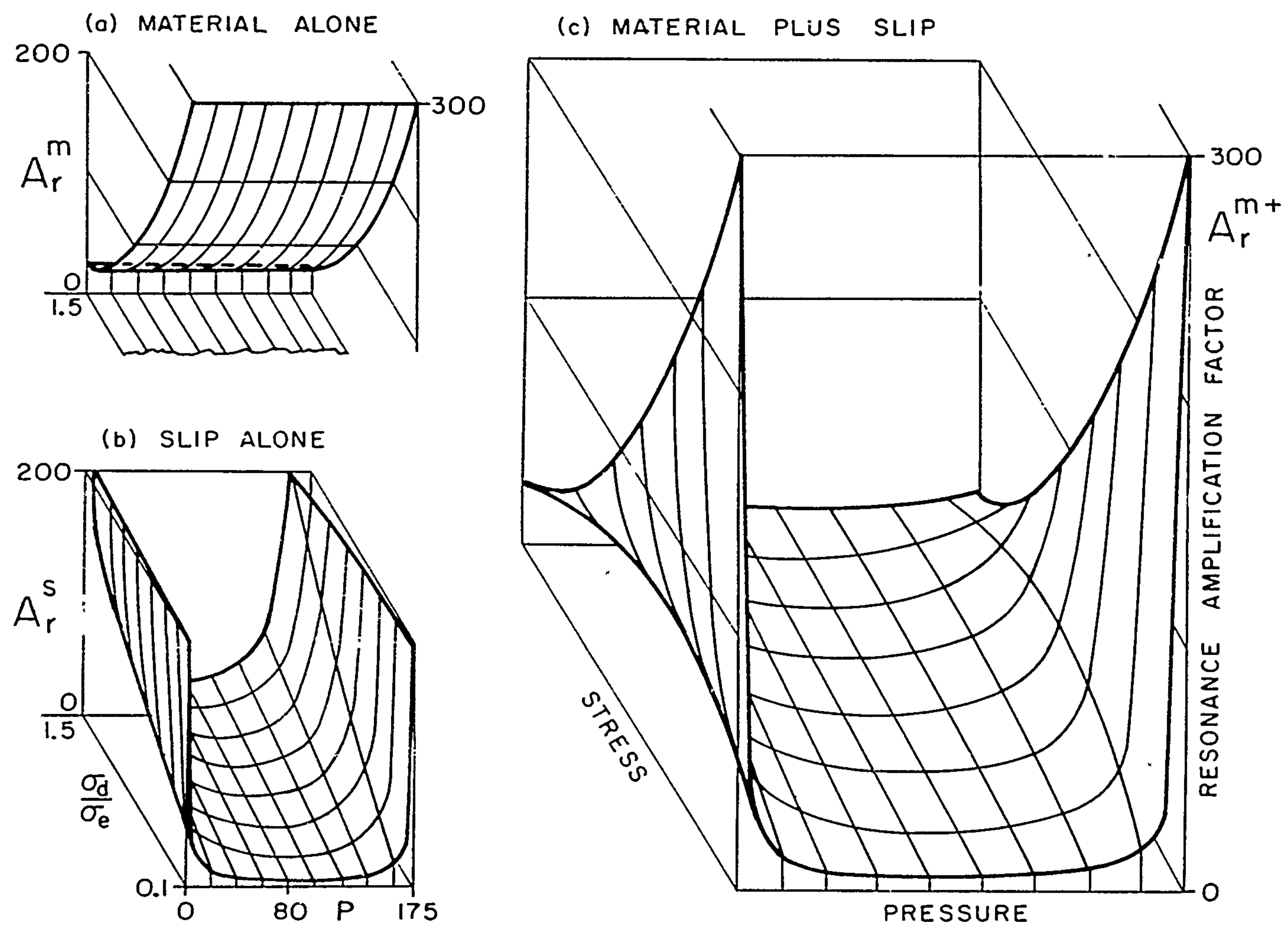


Fig. 6 Resonance Amplification Factor of a Bileaf Cantilever Beam Considering Both Material and Slip Damping.

indicates the resonant fatigue strength. In a system with low damping the exciting force F_g can only be a small fraction of force F_e (ratio F_g / F_e small) before dangerously high fatigue stress is developed. The shape of the exciting force curves shown in Figure 7 are essentially the same as the damping curves shown in Figure 5. Force F_g is large at low stress under optimum pressure (where slip damping is large) and at high stress (where material damping is large).

2. Shear in An Interface Adhesive

As discussed previously the fretting and surface corrosion associated with dry slipping (and developed even in lubricated surfaces) under optimum pressure has led to the study of another damping mechanism: shear hysteretic damping in a viscoelastic adhesive layer located at the interface between rigid members. No shear slip between adhesive and adherents is assumed in such a configuration and all of the relative shear displacement between adherents is taken up as shear strain within the adhesive.

The shear properties of viscoelastic materials are generally specified in the complex notation:

$$G_r = G_1 + iG_2 \qquad \text{Eq. (6)}$$

where G_r = total, resultant, or absolute modulus of rigidity, or complex rigidity

$$= \left[G_1^2 + G_2^2\right]^{1/2}$$

G_1 = real part of modulus, or <u>storage modulus</u>

G_2 = imaginary part of modulus, or <u>loss modulus</u>

The specific damping energy D, in terms of the above notation is:

$$D = \pi G_2 \gamma^2 \text{ in-lbs/cu.in./cycle} \qquad \text{Eq. (7)}$$

where γ = unit shear strain amplitude in the adhesive layer

For illustrative purposes let us consider one rather typical adhesive* and compute its energy dissipation potential as compared to structural metals. At room temperature at 50 cycles per second sinusoidal frequency this adhesive has the following moduli (4): G_1 = 95 psi,

*Minnesota Mining and Mfg. Co. adhesive #466.

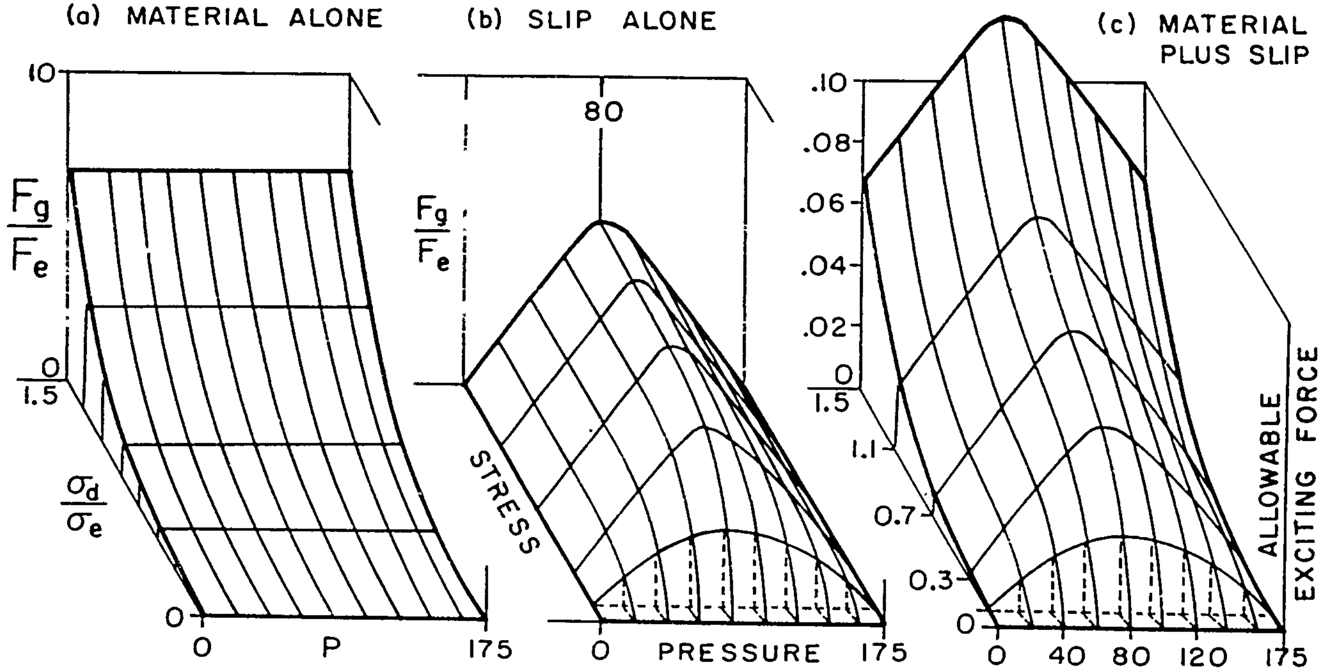


Fig. 7 Allowable Exciting Force at Resonance for a Bileaf Cantilever Beam Considering Both Material and Slip Damping.

G_2 = 110 psi, and G_r = 145 (moduli values are about twice as high at 250 cycles/sec). Furthermore, this adhesive can withstand large cyclic shear safely; shear strains greater than unity at low frequency cause no apparent deterioration in millions of cycles. At unit shear strain damping $D = \pi(110)(1)^2 = 345$ in-lbs/cu.in/cycle. This damping energy is about 100 times larger than the D values for a "typical" structural metal at its fatigue limit. Furthermore, through proper design, as illustrated later, the adhesive layer can be subjected to uniform shear strain, and thus be capable of dissipating maximum energy at all locations. By contrast, most structural parts have rather severe stress gradients and only a very small percentage of their total volume is at maximum stress and dissipating maximum unit damping energy (see reference (10) for quantitative treatment of this factor through use of volume-stress functions). It is thus apparent that viscoelastic adhesive, if properly used in a configuration can dissipate very large damping energy.

In order to clarify the general concepts involved in adhesive shear damping and to illustrate how they can be utilized a specific example shall be briefly discussed. In this case it will be helpful to recall the general concept (10) that in order to maximize damping it is necessary to maximize the "effective" stress (or strain) on the interface layer. Having high shear stress locally only on an interface, while allowing most of the interface to remain at a relatively low shear stress, does not produce a high "effective" stress; most of the interface must be at reasonably high stress. Stated differently the integral of the product of local stress (or strain) and interface volume must be maximized.

Let us examine first a relatively simple case of optimizing a design for maximizing damping. Consider the circular plate (13) shown in Figure 8(a) held so that its edge is always at zero slope but free to move radially as shown. As the plate vibrates with a center amplitude δ, the periphery or edge of the plate will move radially with an amplitude s. If now viscoelastic material is placed between the edge faces of the plant and its support, as shown in Figure 8, the radial motion s provides a mechanism for dissipating damping energy through shear in the viscoelastic interface layer. It can be shown that the amplitude s of radial motion increases with the square of plate deflection δ. The equations for the damping energy dissipated by shear in the viscoelastic material in this configuration were recently developed by Mentel (13). These equations may be expressed in terms of a dimensionless damping energy D_o^v dissipated per cycle by the viscoelastic material. Damping D_o^v may be maximized by optimizing any one of several different design or material parameters. In order to illustrate this concept and its application let us consider the problem of selecting the most effective viscoelastic layer by maximizing the damping equation. The manner in which the total energy dissipated by a viscoelastic layer varies with its G_1 and G_2 values (as defined in Equation 6) is shown in Figure 8. The damping surface shown indicates that the damping energy

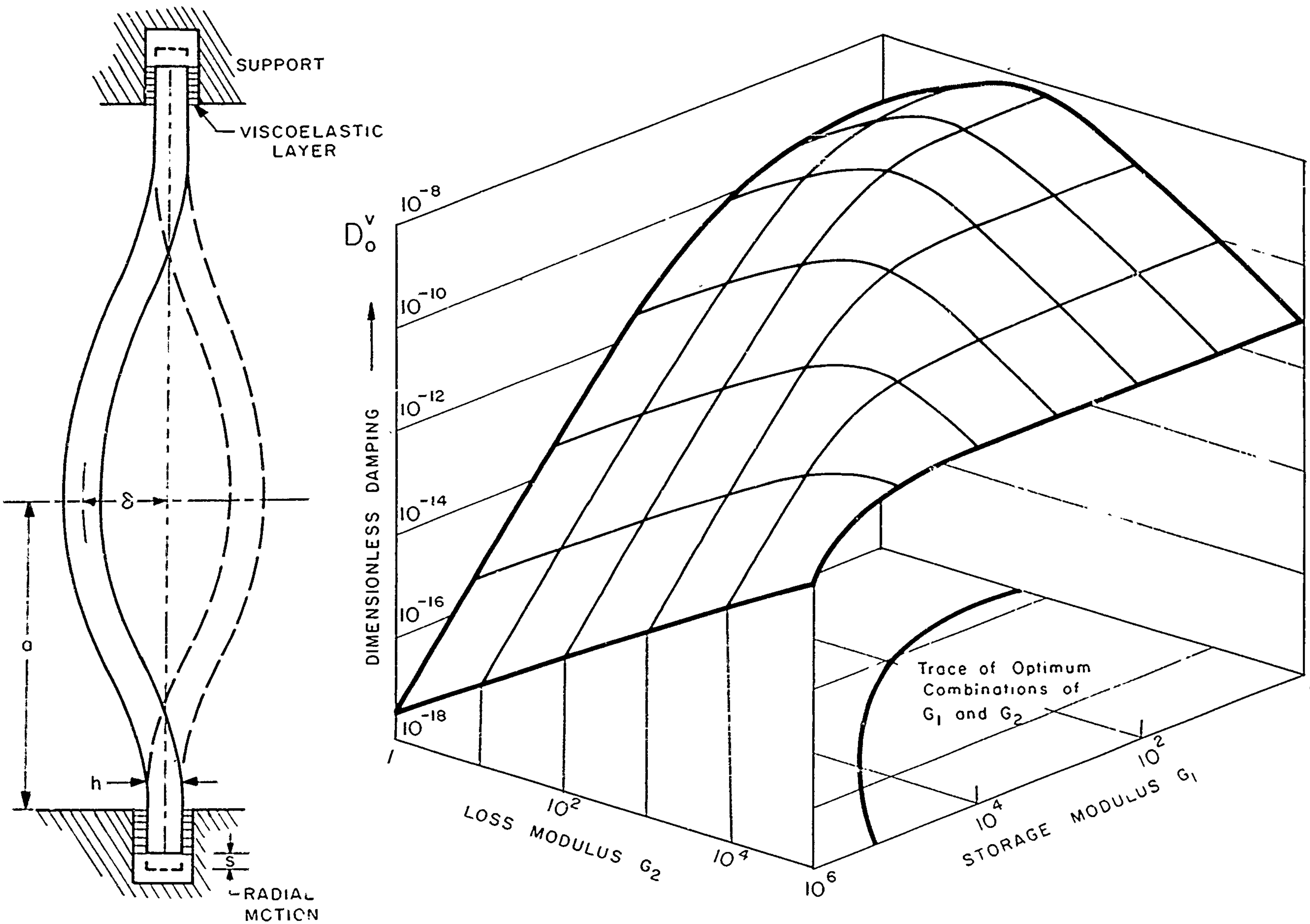


Fig. 8 Energy Dissipation at The Supports of a Circular Plate for Different Properties of Adhesive Supports.

dissipation changes as follows: (a) it increases with decreasing values of G_1, and (b) first increases with increasing values of G_2, generally reaches a maximum at some optimum value of G_2, as shown by the hump in the damping surface, and then decreases. The optimum combinations of values for G_1 and G_2 which result in maximum damping energy dissipation for this special case is shown by the curve traced in the $G_1 - G_2$ base plane. This trace shows that the desirable optimum value for G_2 is approximately 9 and this in combination with a value of G_1 less than 10^2 maximizes the damping.

It should be emphasized that scales of Figure 8 are logarithmic and that the differences in damping energy dissipation for different combinations of G_1 and G_2 may be very large. For example, for the combination of G_1 equal to 10^6 and G_2 equal to 1 the damping energy dissipation is less than 10^{-17}, whereas under the optimum condition of G_2 equal to 9 and G_1 equal to 10 the damping energy dissipation is more than 10^{-9}. The ratio of these two values is 10^8. The potential gain which can be realized through the optimization process is therefore large. This comparison does not of course consider material damping in the panel itself.

It would be instructive for general guidance to compare for this case the energy dissipated by the hysteretic effects in the structural material of the panel with the viscoelastic interface damping discussed above. This is done in Figure 9 which shows the damping as a function of panel dimensions h/a and panel center deflection δ. The damping of the viscoelastic interface, optimized as discussed previously and shown in Figure 8, is indicated by the D_0^v plane. The er .ry which can be dissipated through hysteretic damping in the structural material itself is shown on a similar basis by the D_0^m plane. Observe that at low values of h/a for the panel the viscoelastic interface damping is large compared to the material damping. The two damping values are approximately equal for instance at h/a of 10^{-2} and of 10^{-4}, point A on the intersection line AB. For large values of h/a, and particularly for small values of δ, the material damping is larger than the viscoelastic damping.

The total damping of the panel is, of course, the sum of the two contributions. The differences between the two in some regions of the h/a and δ parameters is very large (observe logarithmic scales). It is thus important, before deciding on how to increase the total damping in a system, to know which of the two contributors offers the greater potential. For example, in those regions where viscoelastic damping dominates relatively little can be gained in total damping by material substitution, heat treatment, or mechanical processing for higher damping. If, for example, the material damping is only one percent of the total, a 1000% increase in material damping will increase the total damping by only ten percent. It is thus apparent that before total damping can be increased significantly it is first necessary to decide, by analyses such as indicated above, wherein the greater potential gain lies. Then a decision can be made on whether to concentrate on improving the materials or the design.

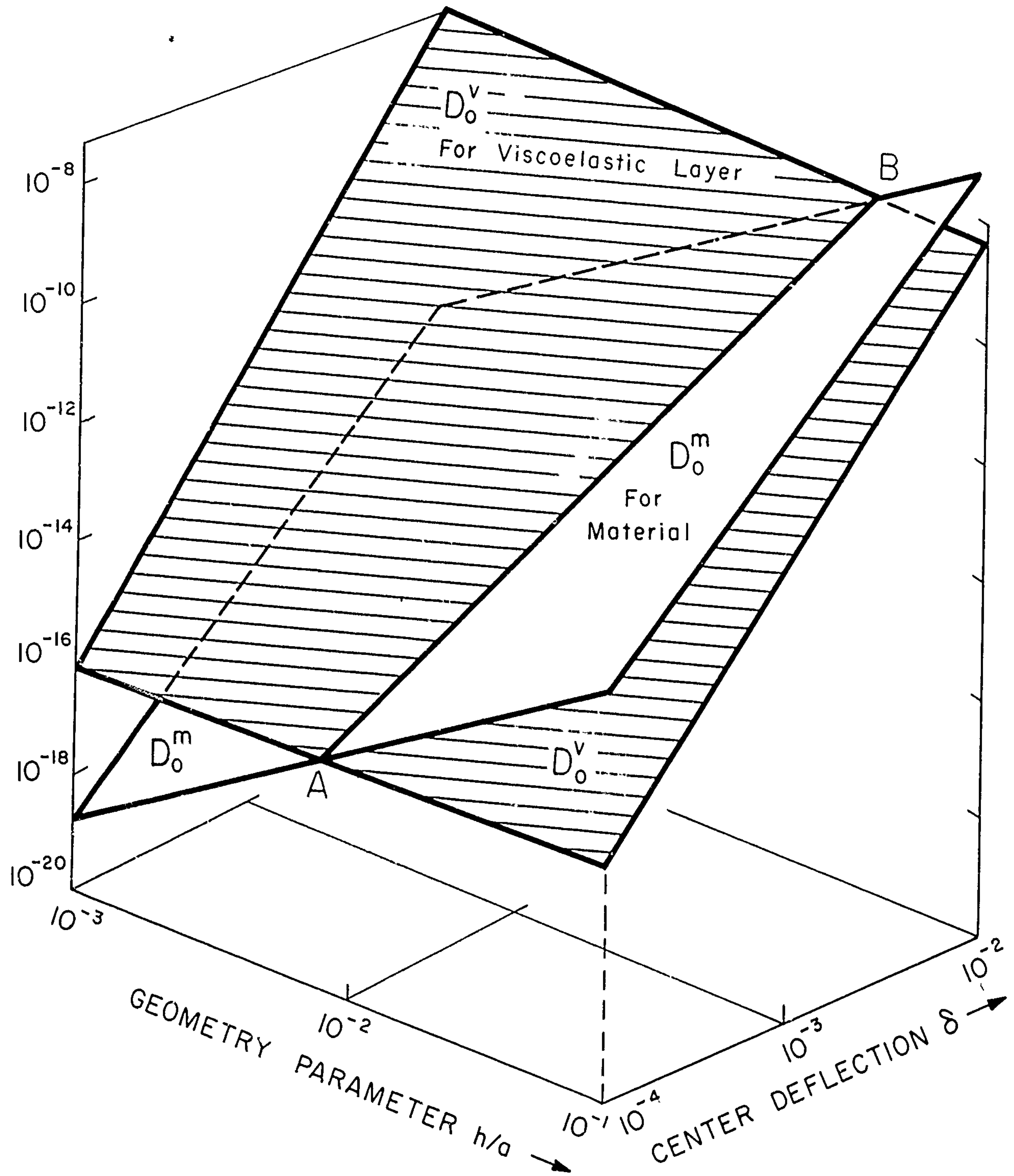


Fig. 9 Comparison of Adhesive Interface Damping and Material Hysteretic Damping for a Circular Plate.

An analytical approach such as outlined above also enables studying the effects of changes in design and material parameters other than those considered above (13). Curves and surfaces similar to those shown in Figures 8 and 9 may be constructed to clarify the significance of many different design parameters.

3. The "Design Optimization" and "Surface Addition" Approaches for Increasing Damping

The approach discussed above might be identified as the "design optimization" approach. Its object is to suggest the proper proportions and materials for a configuration which must have high damping. However, in practice it is usually difficult to change an existing design, yet there is often room to add a surface treatment to increase the damping of a member which otherwise cannot be changed radically. Examples of such a "surface addition" approach are damping coatings, damping tapes, spacing layers, sandwich additions, and other types of layered construction. Various types of surface additions have been tried in the past. Three of the more important types are described below.

One of the earliest approaches for increasing the damping of panels is to add a thick layer of viscoelastic or similar type of material (identified by heavy cross-hatching in Figure 10) to a panel. As the panel is subjected to cyclic bending deflections $\pm d$, cyclic direct strain (tension-compression) is developed in the viscoelastic layer, as shown in Figure 10 by the difference in the lengths of line A' B' (> A B) and A" B" (< A B). Although reasonably satisfactory in some types of panel noise problems, this approach for increasing damping is generally ineffective since the cyclic strain in the viscoelastic coating is usually small and only a small percentage of the damping capacity of the viscoelastic layer is utilized.

A second "conventional" surface addition approach for increasing damping involves the addition of so-called "sound damping tape". This is a configuration shown schematically in Figure 11, consisting of a thin layer "A" of viscoelastic adhesive (thickness exaggerated in figure for clarity) and a backing band or tape "T", usually aluminum or some other metal. The mechanism of energy dissipation in this case is one of shear in the viscoelastic layer as shown schematically by the distortions in the cross-hatch bands. As the beam flexes, the viscoelastic layer, located well above the neutral axis and restrained by the backing tape, receives a cyclic shear, as shown in the figure, and thus dissipates energy. However, as in the case of the viscoelastic coating shown in Figure 10 only a very small percentage of the damping which the viscoelastic layer is capable of dissipating can generally be realized. For most thin beams or panels the distance between the neutral axis of bending and the place of the viscoelastic adhesive is so small that the cyclic shear strain imposed on the adhesive is well below its limits; thus the energy dissipation is also well below the capacity of the adhesive.

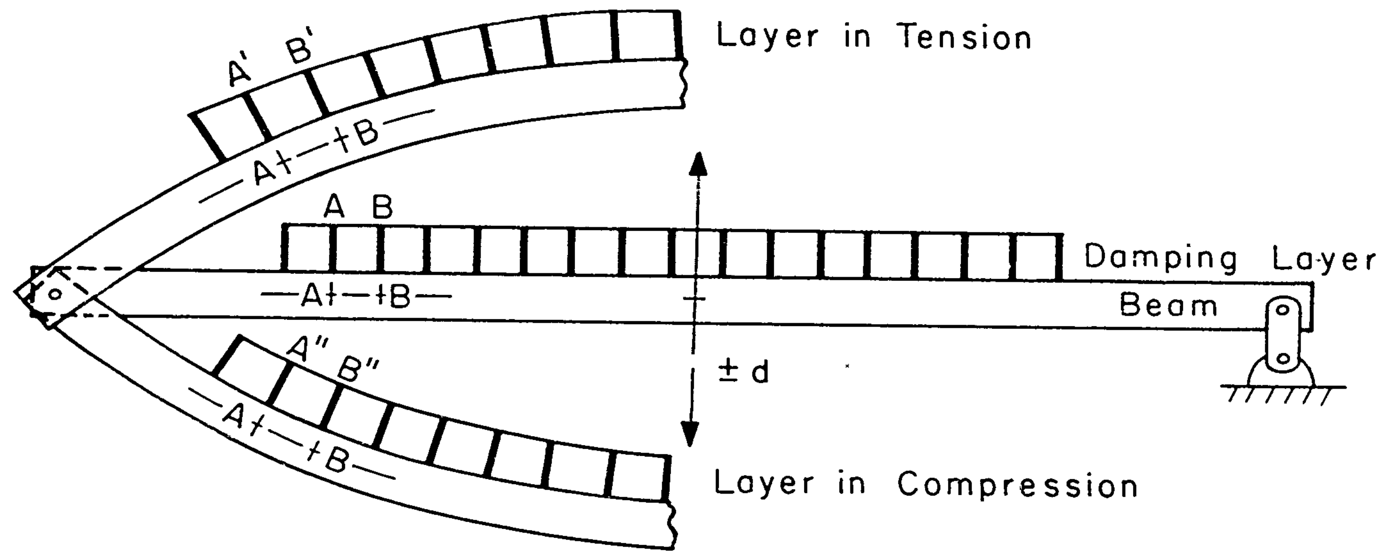


Fig. 10 Viscoelastic Coating Which Dissipates Hysteretic Damping Energy Under Cyclic Direct Stress (Tension - Compression).

An improvement on the damping tape method shown in Figure 11 involves a spacing layer (14) (15) shown in Figure 12(a). This spacer locates the viscoelastic layer further from the neutral axis and thus increases the cyclic shear associated with a given cyclic flexing. Theoretically, the damping energy which can be dissipated by the adhesive shear mechanism can be greatly increased by such a spacing layer. However, there are some practical limitations. In order to be effective, the spacing layer must be reasonably thick. Furthermore, most applications require a low density material to avoid excessive weight and cost. Unfortunately, low density spacing layers, or core materials, have relatively low shear modulus. Thus, as illustrated in Figure 12(b) much of the cyclic shear which would otherwise be transmitted to the viscoelastic adhesive may be lost as shear strain in the core material. For a rigid spacing layer (Figure 12(a)) all the cyclic shear strain γ_o is felt by the adhesive, whereas for a low density spacing layer having relatively low shear modulus, Figure 12(b), the available total cyclic shear γ_o is partially dissipated in the core (part γ_c) leaving only part γ_a for the adhesive layer. Thus, as in the previous cases, the adhesive will be subjected to relatively small cyclic shear and the full potential of the viscoelastic adhesive is again not realized.

A spacing layer approach can also be used to increase the distance between a damping layer, such as shown in Figure 10, and the neutral axis of bending, thus increasing the direct (tension-compression) cyclic strain. However, the same difficulty discussed above is again encountered; low shear modulus of low density spacers will absorb much of the available shear strain.

Other surface addition approaches are currently under study (16). As in the case of the configuration optimization approach it is also important in the surface addition approach to carefully optimize the design for maximum damping. As an illustrative problem, consider the solid beam AB shown in Figure 13 to which lugs have been added as shown. This simulates the spacing layer approach illustrated in Figure 12(a) in that the viscoelastic shear region is moved a greater distance from the neutral axis N.A. of the beam and the cyclic shear strain in the viscoelastic layer is correspondingly increased. Equations which have been developed for this case (15) show that, as in the previous case, it is now possible to optimize any one of several design and material parameters and by so doing to maximize the damping. However, for illustrative purposes we consider just two of the design parameters: (1) the thickness b and (2) the property G_1/G_2 of the viscoelastic layer. For the particular beam considered the total damping energy as a function of these two parameters may be plotted as shown in Figure 13. It is observed from this figure that the optimum thickness for a variety of viscoelastic materials (various values of G_1/G_2) is approximately five mills. Furthermore the lower the ratio G_1/G_2 the higher the damping energy dissipated at the viscoelastic layer. As the thickness decreases below two mills the damping decreases very rapidly but as the thickness increases above five mills the corresponding decrease in damping is

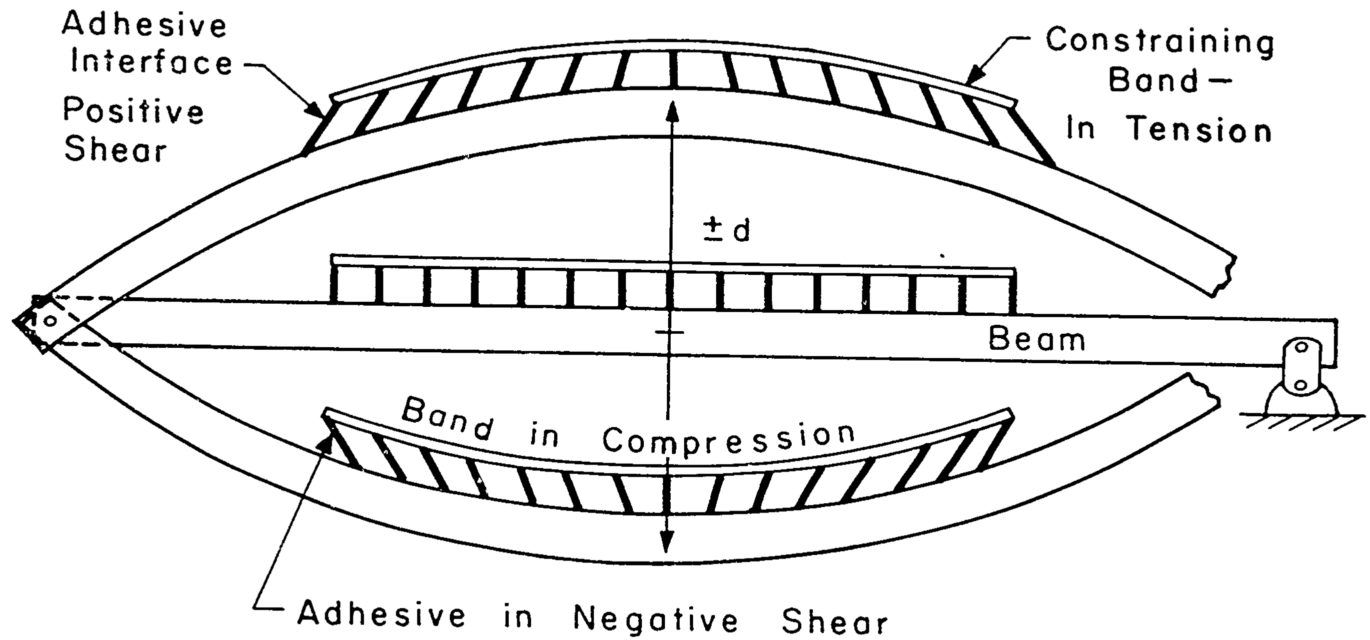


Fig. 11 "Damping Tape' which Dissipates Hysteretic Damping Energy Under Cyclic Shear Strain in Adhesive Between Beam and Constraining Tape.

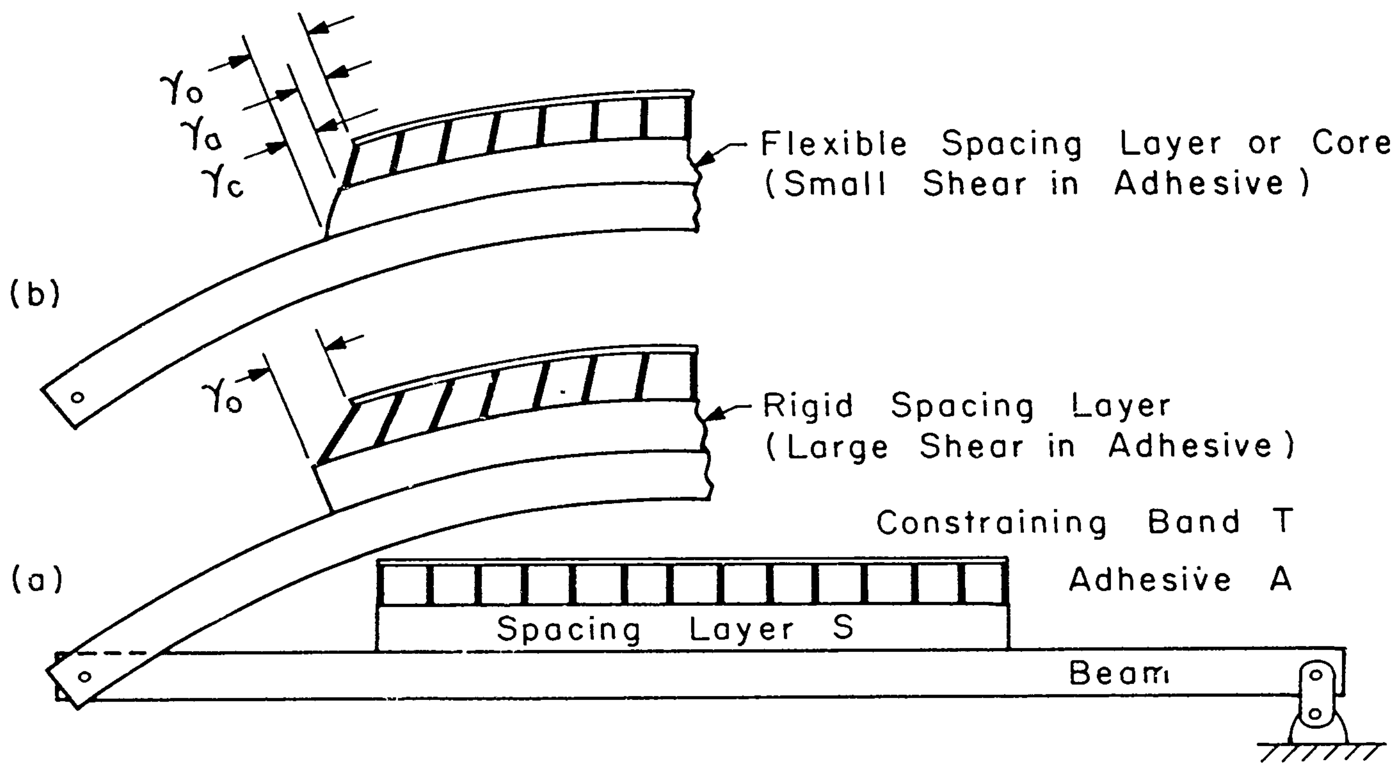


Fig. 12 Damping Tape on Spacing Layer.

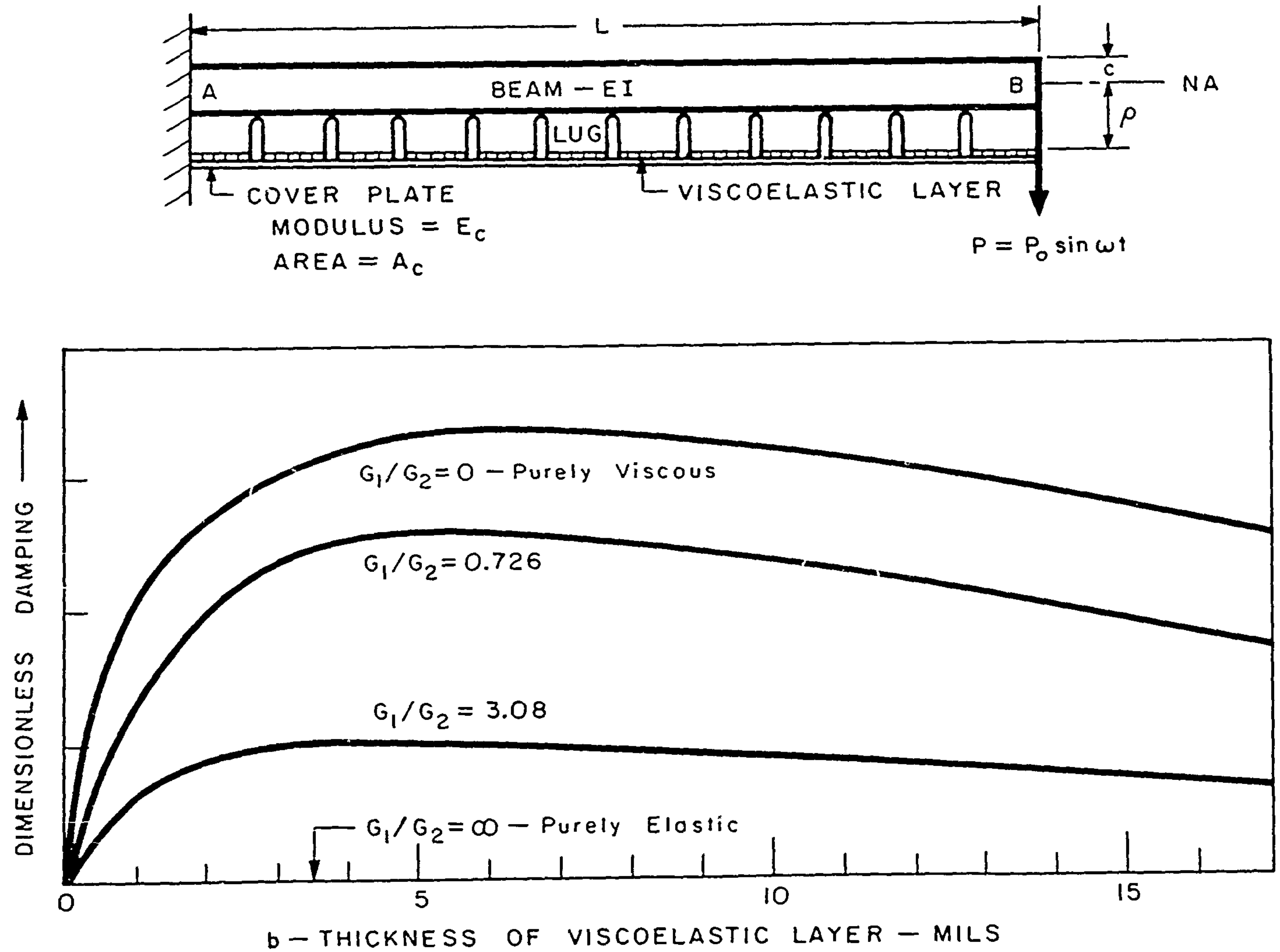


Fig. 13 Effect of Thickness of an Adhesive Layer on Damping Due to a Surface Addition.

relatively small. Observations such as these help indicate how critically the design must be optimized.

As in the previous case, it is desirable to compare the damping associated with the hysteretic effects in the beam material itself with damping associated with the viscoelastic layer. Such a comparison is made in Figure 14. The damping of the viscoelastic layer as a function of two parameters (representing the material and the geometry factors as defined in Figure 13) optimized to produce maximum damping is shown by the curved surface. For comparison purposes the hysteretic damping of the beam material itself stressed to its fatigue limit is shown as a horizontal plane having an elevation of 0.0037. The intersection of this horizontal plane, representing material damping, and the curved surface, representing viscoelastic damping, curve ABC, separates regions in which each mechanism dominates. Observe that for very low values of the material and geometry parameters the material damping may exceed viscoelastic damping and that for intermediate and high values of these two parameters the viscoelastic interface damping is much larger than the material damping. At the best combination of the two parameters shown in the figure the viscoelastic damping is approximately 20 times as large as the material damping.

Other cases illustrating design factors important for high viscoelastic damping are discussed in references (16) and (13).

V. CONCLUSIONS AND CLOSING REMARKS

1. In the past either the natural frequency of a system or its exciting frequency could generally be adjusted to avoid resonant conditions. However, this is no longer possible in many types of aerospace vehicles. In particular, many types of random excitation, especially those of acoustical origin, cover such a wide frequency band that resonant vibrations can no longer be avoided.

2. The design concepts of former years are inadequate to deal with the fatigue problem associated with resonance amplification. Although "beefing-up" a structure for increased strength and stiffness has "fixed" resonant fatigue difficulties in some cases, this approach cannot be considered a long term engineering solution. Not only is the cost and weight penalty large in such an approach but also it is totally inadequate to meet the problems encountered in many of the newer types of aero-space vehicles. Thus, the greater utilization of structural damping as an engineering property to control resonance is a necessity. Looking to the future it will still be necessary, of course, to stiffen panels and other configuration by using sandwich and other similar build-ups. However, if the maximization of damping is also considered to be a design objective, in addition to the stiffness increase, then a much larger gain in resonant fatigue strength can be realized than using either criterion alone to the exclusion of the other.

3. Whereas in the past the damping of a structural assembly was often increased by the addition of separate energy absorption units (dash pots and the like) or energy transfer devices (dynamic absorbers) this approach no longer provides an engineering solution for many of the newer types of configurations and resonance conditions. Looking to the future the design concept for <u>optimizing a configuration for maximum inherent structural damping</u> (damping built into the structure itself) must be emphasized.

4. The experimental and analytical knowledge on the major sources of <u>inherent structural damping</u> (hysteresis in structural materials, interface slip, and interface adhesive shear) have now reached a point where application to structural design is practicable. Interface slip must, however, be used with caution because of the fretting and corrosion it may produce.

5. The relative importance of hysteresis in the structural materials and interface adhesive shear as damping mechanisms depends on the configuration, how it is optimized, and the force regimes considered. In most cases however structural materials are selected and shaped for properties other than large damping whereas the interface adhesive may be especially selected and the layer thickness optimized for large damping. In such an optimized design the contribution of interface adhesive shear to the total damping of a structural system can be very significant.

6. Looking to the future the greater utilization of interface adhesive shear as a damping mechanism, both in an optimized configuration and in surface addition, should be encouraged.

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SONIC FATIGUE RESEARCH GOALS

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The high level of service maintenance for many areas of secondary structure and equipment components of current aircraft is attributed to sonic fatigue. It is established that the energy source for the component loadings is the acoustical energy generated by the high thrust jet engines. The acoustical energy is in the form of pressure pulsations occuring at all frequencies from a low cps to 12-14,000 cps. Generation, by these pressures, of structural resonant response greater than endurance limit stresses results in the problem of sonic fatigue.

A series of environmental and laboratory tests were conducted to develop different types of structure capable of withstanding a current requirement of approximately 165 db. A resume of the results of this program is presented.

Through the use of stress response versus sound level change, random s-n curves and environmental test data, changes in environmental lives are predicted from single frequency horn test results.

The continued demand for high thrust engines necessitates the development and testing of structural arrangements capable of withstanding overall sound levels from 175 to 178 db. Agressive development and completion of this type of research is required to keep pace with our aircraft performance requirements.

SONIC FATIGUE RESEARCH GOAL

This paper presents some of the aspects of a relatively new and most serious problem confronting the aircraft industry.

The magnitude of this problem is measurable in cost and usage. The cost is running into millions of dollars to maintain aircraft in flying condition. The usage is important when key defense vehicles are grounded for repairs.

The problem, known as sonic fatigue, has arisen with the continuous use of aircraft equipped with high thrust jet engines.

The areas associated with the sonic problem that will be discussed are:

1. The definition and nature of the problem.

2. Our present design capabilities as related to the anticipated immediate future sonic environment.

3. Future research goals.

DEFINITION AND NATURE OF THE PROBLEM

In defining the problem it may be stated that "Aircraft Sonic Fatigue" is the type of structural and equipment failures attributed to loadings caused by the air pressure pulsations (noise) generated during high thrust jet engine operation.

To understand the nature of the noise or Sonic loading, it is necessary to be familiar with some fundamental concepts of acoustics and of the response of structures to sound. A sound is essentially a fluctuating pressure with the number of fluctuations per unit time determining its frequency usually given as cycles per second.

The intensity of a sound is determined by the displacement amplitude and frequency of these fluctuations. The commonly used method of expressing the intensity of a sound is by a decibel rating. The decibel rating or level is ten times the logarithmic ratio of the intensity of the sound in question to a reference intensity. For most purposes, the reference intensity has arbitrarily been taken as the threshold of hearing. Expressed mathematically:

$$db = 10 \log_{10} I_1/I_o$$

where I_1 = Intensity of the sound in question (watts/cm^2)

I_o = Reference intensity (threshold of hearing) (watts/cm^2)

The decibel level of a sound may also be determined by its pressure amplitude, that is:

$$db = 20 \log_{10} \frac{P_1}{P_o}$$

where P_1 = Pressure of sound in question

P_o = Pressure at the threshold of hearing

From these logarithmic ratios, it can be shown that an increase of six decibels results in doubling the pressure amplitude.

Where a panel or some composite structure is subjected to sonic loading (air pulsation) it is forced to vibrate. The problem of sonic fatigue arises when the induced vibration amplitudes result in stresses greater than endurance limit stresses.

As previously stated, the modern jet engine is a powerful source for the generation of these sound (pressure) pulsations. Further, the sound generated is completely random in amplitude and is produced at all frequencies. Therefore, with a forcing function available at all frequencies, all components will be loaded at resonance frequencies. An additional characteristic of the modern jet engine is the fairly constant power level over a rather broad frequency band. This is illustrated in Figure 1.

The question of why sonic fatigue has recently become a major problem is illustrated in Figure 2. Here may be seen that the forcing function has grown by a factor of 18 from the J-47 jet engines of 1947 to the J57-P43W engines of 1958. As a result of this growth in the magnitude of the forcing function, component stresses are now being generated which are greater than endurance limit stresses (for conventional construction).

The next question to be answered is what procedures are to be used to solve this complex problem. A pure analytical approach has not been considered feasible due to the complex nature of the forcing function coupled with the completely heterogeneous characteristics of the aircraft components, the type and nature of incurred failures, etc. In fact, the probability of developing a purely analytical analysis method to which a high degree of confidence could be attached for the majority of the problem areas still does not appear too feasible. Therefore, it has been and will continue to be necessary to resort to testing to develop the required component arrangements, types of construction, and design experience, and to establish component service lives for present and future designs.

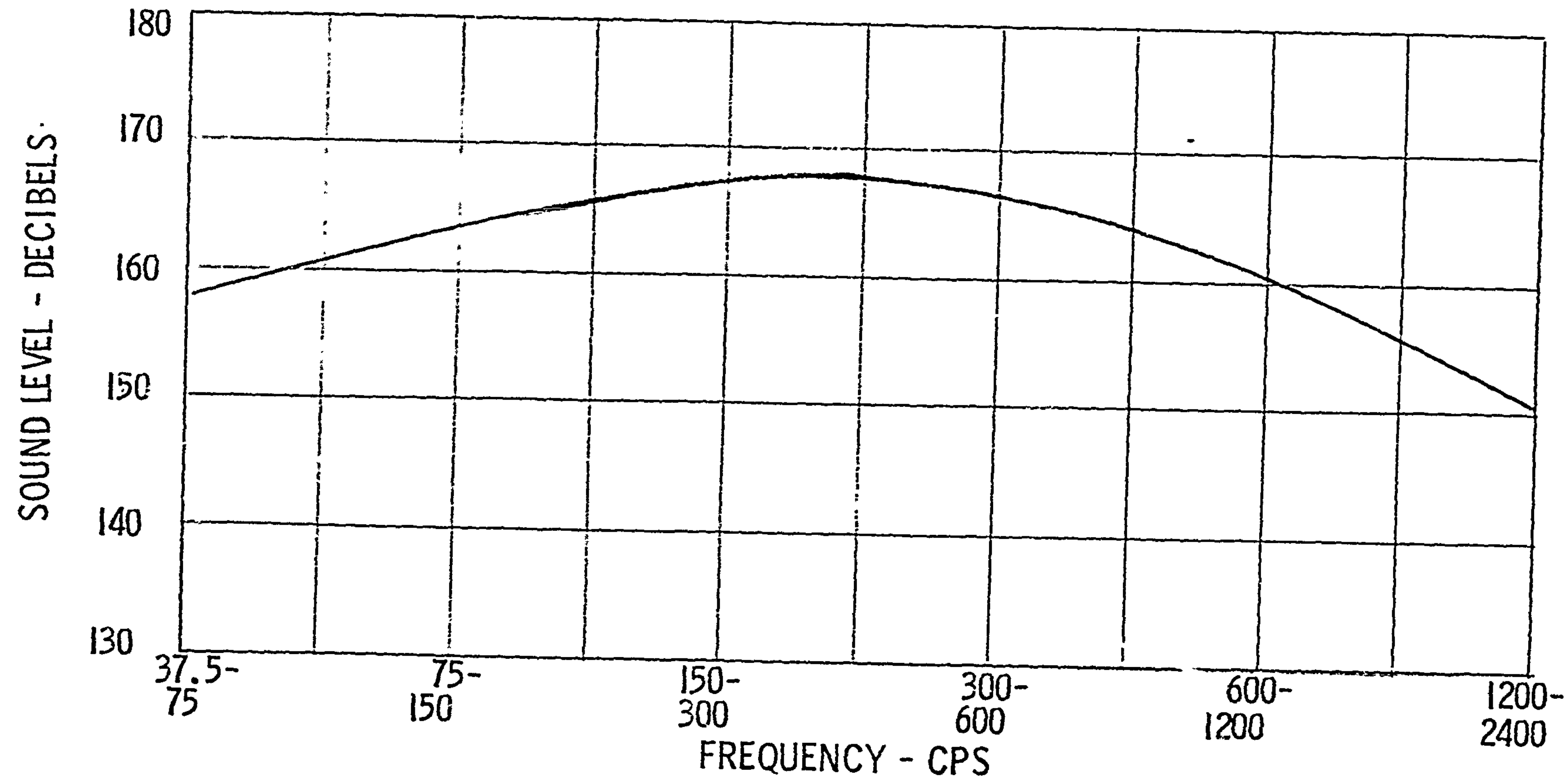
PRESENT DESIGN CAPABILITIES

Before stating where future sonic research efforts should be applied, it appears appropriate to briefly outline present capabilities.

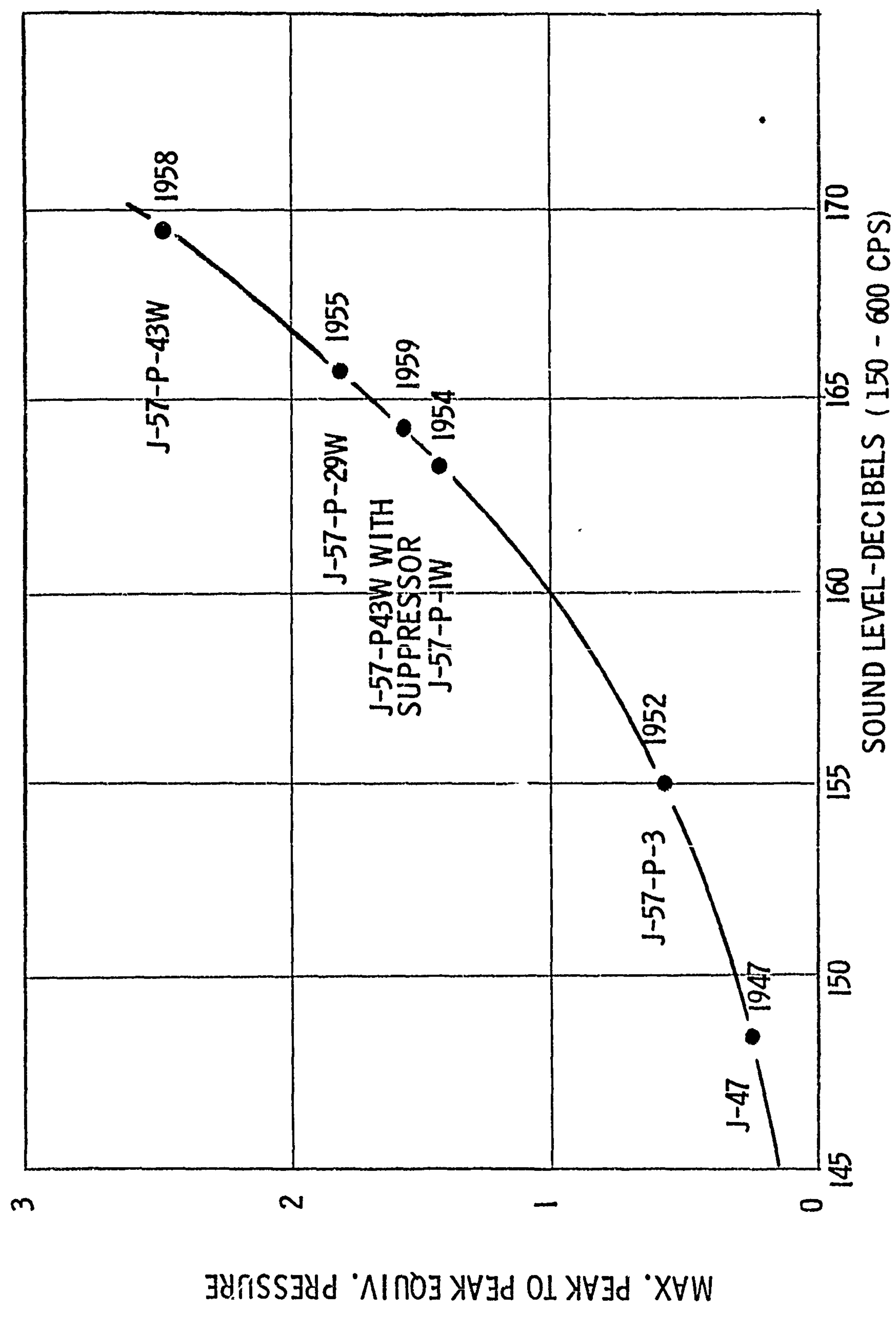
Utilizing the testing philosophy of the preceding section, extensive development work both in the laboratory and in the true aircraft environment has been conducted over the last six years at Boeing. Representative of the areas investigated are the structural arrangements shown in Figure 3.

FIGURE I

TYPICAL NOISE SPECTRUM FOR J-57 JET ENGINE



SOUND PRESSURE VS. SOUND LEVEL - FIGURE 2



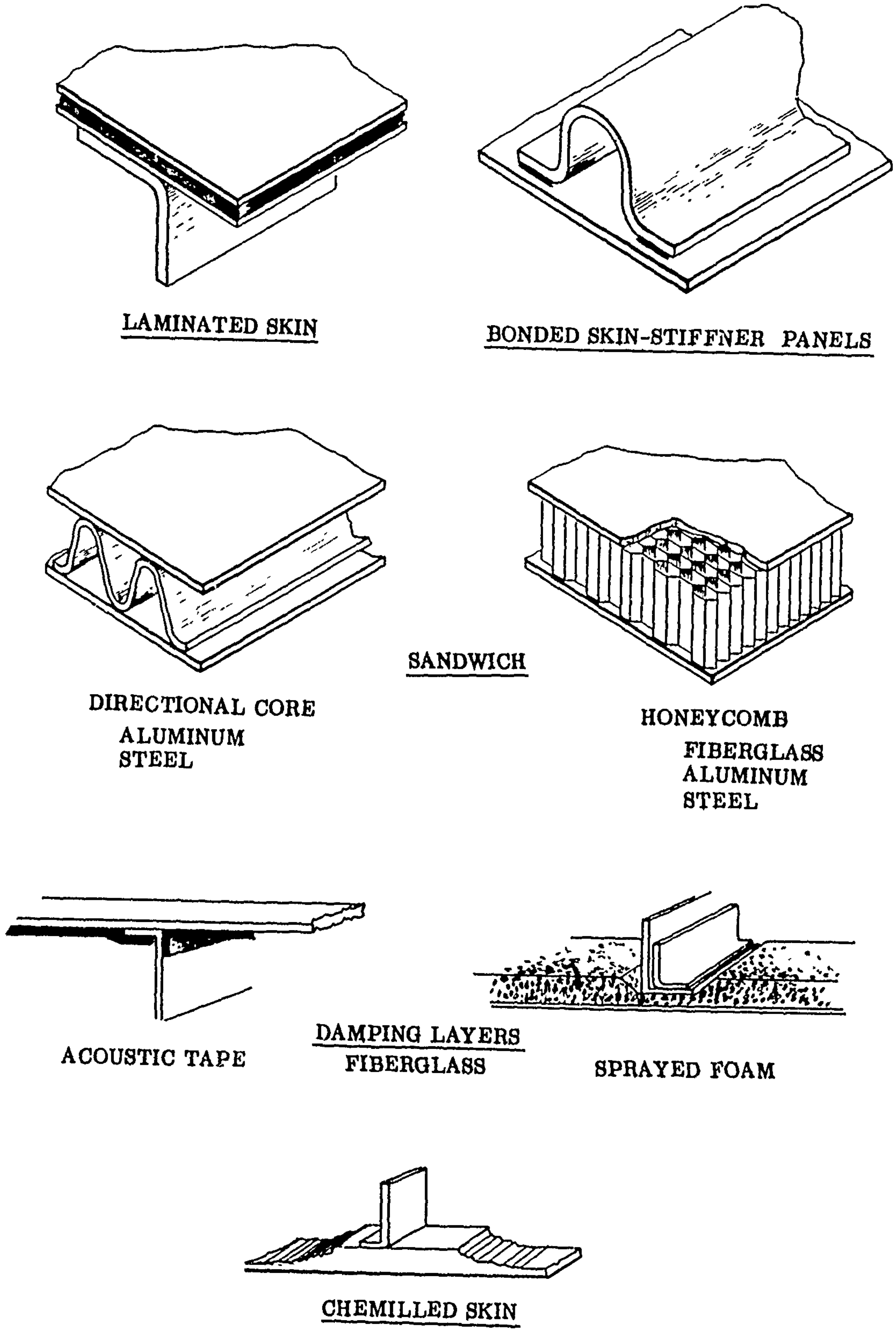


FIGURE 3 STRUCTURAL ARRANGEMENTS SONIC TESTED

Briefly some significant items resulting from the sonic study effort are:

1. Honeycomb structure is a relative lightweight structural arrangement for use in high sonic environments. Of considerable importance here was the development and testing of edge arrangements and attachments as shown in Figure 4.

2. Skin stiffener arrangements were developed with sonic fatigue lives of more than 100 times that of conventional structure. Figure 5 illustrates the arrangements considered in the program and the large increase in life obtained with the bonded doubler back to back angle arrangement.

3. True airplane environmental testing is needed for prediction of component service life. The complexity of the forcing function, structural arrangements and structural response prevents satisfactory overall evaluation with a single frequency noise generator.

4. Documentation was accomplished to record the many types of sonic failures associated with service usage and laboratory testing. Included in this documentation were:

   a. Sound Levels

   b. Detail Description of Failures

   c. Engine Operating Time Before Component Failure

   d. Method of Redesign

5. Test results were correlated to aircraft service life. A theoretical method was developed for correlating life improvements in horn and environmental tests to corresponding improvements in service life and predicting service life from environmental testing. The method employs a random service curve to predict service life based on a test life under corresponding random excitation. The essential steps of the method are outlined in Appendix A.

Through this program there have been developed structural arrangements capable of withstanding environmental sound levels of the order of 165 db in the 150-300 cps octave band. As weapon systems under current development, in general, have acoustical environments less than 167 db, many of the immediate future sonic problems can be solved by the rigorous application of the results of these development programs.

In summarizing our current capabilities, it is the author's opinion that currently available test data provides the basis for information required for many of the immediate future structural design sonic problems. The thin gages, brazing, welding and mechanical attachment problems and material characteristics represent some of the new problem areas associated with the use of steel in design. However, employment of the concepts developed during the aluminum program will give acceptable answers in many of these areas. Further, the complexity of any structural assembly dictates the necessity of environmental testing.

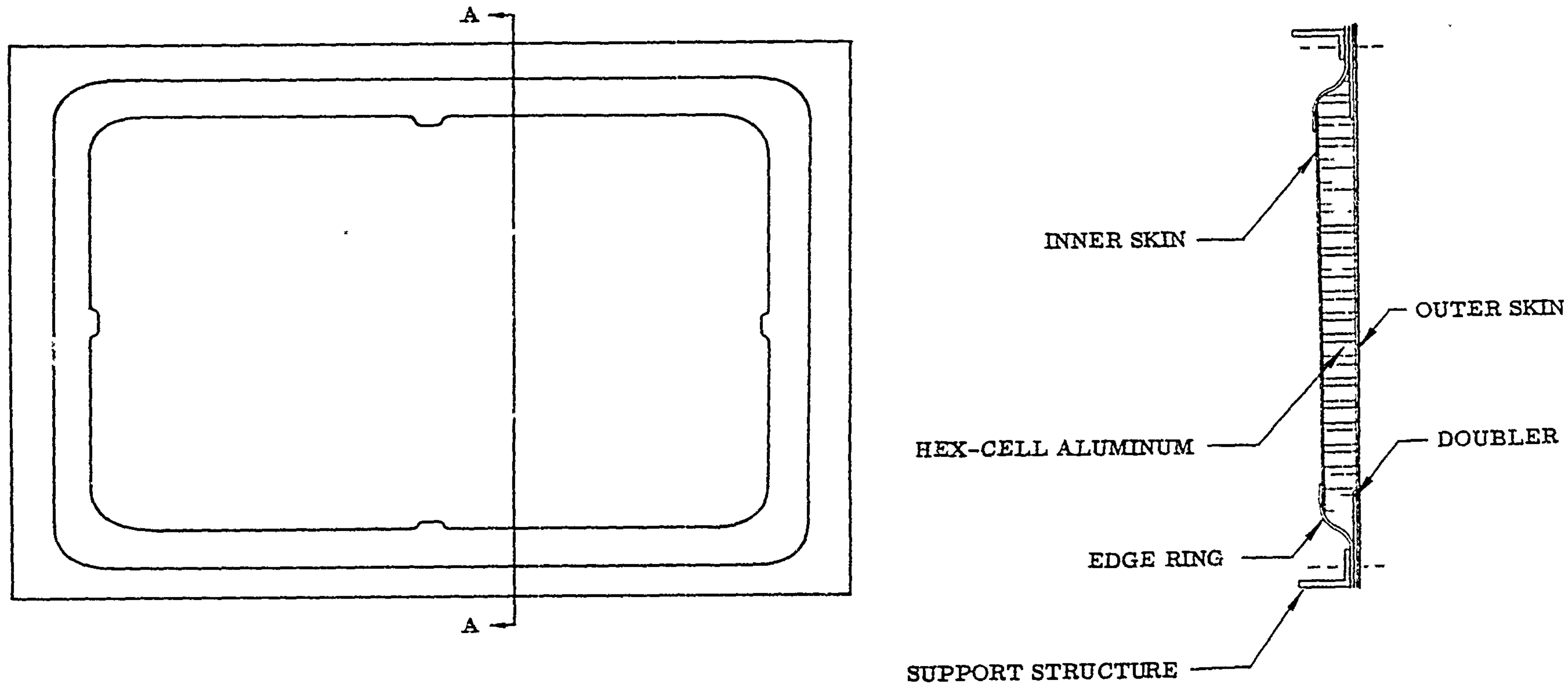
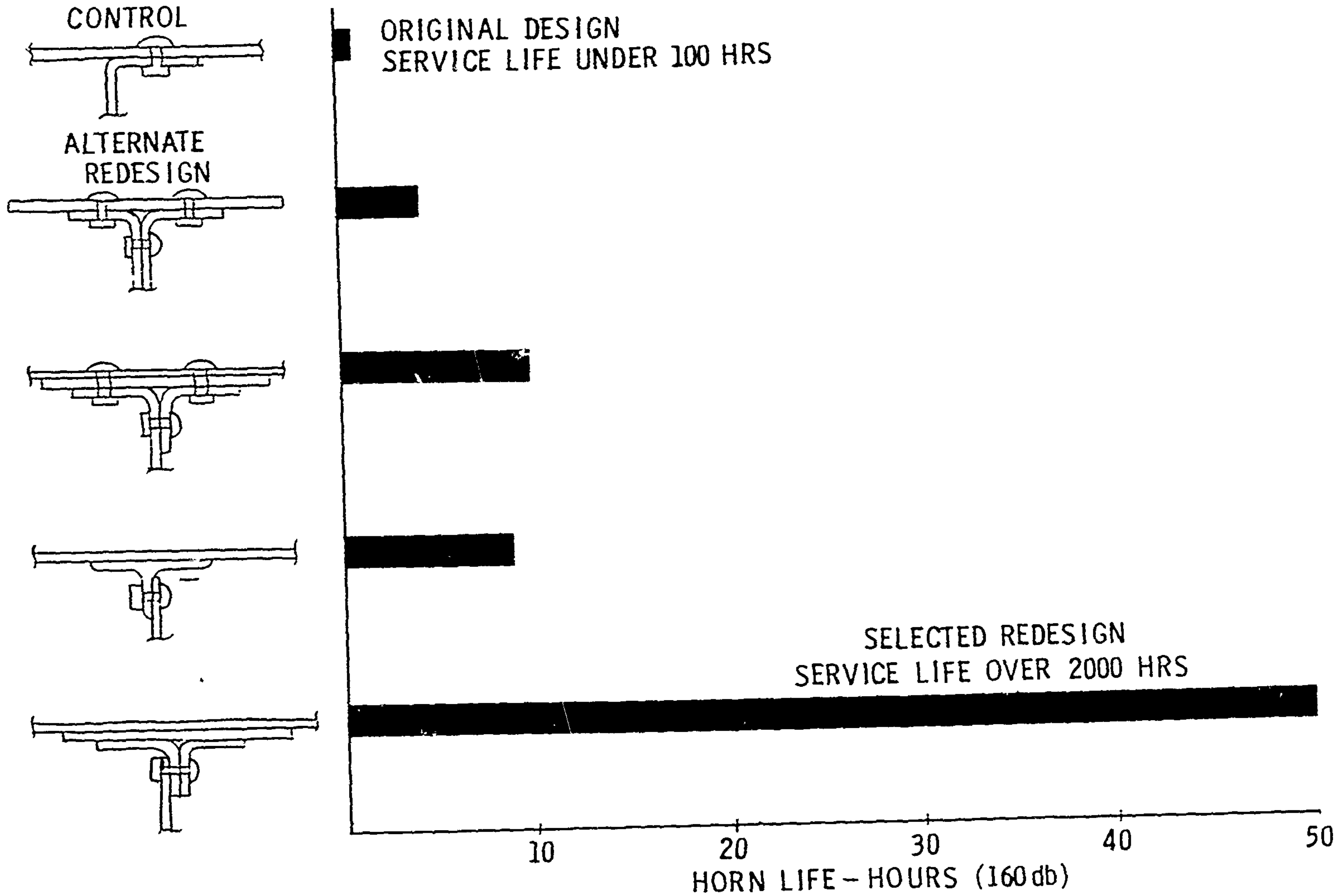


FIGURE 4 TYPICAL ALUMINUM HONEYCOMB PANEL

FIGURE 5

TYPICAL DESIGN EVALUATION BY LAB TESTING



FUTURE RESEARCH GOALS

With the future prospects of designing to an environment of 175 to 178 db in the 150 to 600 cps band (in the 1965 - 1970 period), it is apparent that a considerable amount of work must be accomplished to develop efficient light weight structural arrangements. Service life of from one minute to 3 to 4 hours at these levels must be considered.

The necessity of accomplishing major portions of this development work in as short a period as possible cannot be over emphasized. In the past, an inadequate state of the art existed for solving the sonic problems, therefore, aircraft fabrication has actually preceeded development and testing. Hence, very costly maintenance problems have resulted as the aircraft are put into service without adequate evaluation. It is of necessity that present research efforts eliminate this condition.

The areas requiring research and development efforts are many and varied. Based on past environmental testing and the types of failures experienced during this testing, a few problem areas are apparent that require solutions for accomplishing satisfactory designs:

1. Structural Arrangements - All designs are represented by an array or assembly of structural elements. Also, the response of any element is dependent on the characteristics of the assembly. Therefore, in addition to studying the simple panel problem, structural assemblies (representative of design arrangements) must be studied and tested.

2. Attachments - The problem of attaching structural elements is of major importance. Development work must cover all types of welding, bonding and mechanical attachments including blind attachments and quick disconnect fasteners.

3. Structural Joint Design Techniques - It is in this area that the majority of the sonic failures occur. Also in this area, the stress analysis and stress concentrations become so complex that rigorous theoretical analysis is impractical. However, this is the problem area which requires satisfactory solutions to provide acceptable service aircraft.

4. New Materials - The advent of the super alloys, ceramics and other materials programmed for the future high speed vehicles present many areas requiring research to provide satisfactory sonic service fatigue lives. The low ductibility, stress concentration sensitivity, and attachment problems represent but a few of the problem areas associated with these new materials.

5. Testing - The development of test equipment must keep pace with improvements in design methods and materials. Two items of considerable importance for future sonic fatigue testing are:

a. Development of the Random Noise Generator - Preliminary work at Boeing utilizing a modulated air valve has shown satisfactory duplication of recorded engine sonic environments up to 600 cps. Resulting from this type of testing will be considerable improvement in the correlation of laboratory results to true environment.

b. Necessity of Evaluating Structural Assemblies - The necessity of evaluating the structural assembly rather than a single element under idealized supports can not be over emphasized.

6. Test Correlation - In order to correlate various test data and to define accelerated test requirements, there is a need for the development of random s-n curves based on present and future aircraft materials. Although analytical methods utilizing an assumed force distribution and accumulative damage theories have been developed, adequate experimental verification is required.

7. Acoustical Environment - The complexity of the forcing function constantly results in new problem areas. The desire to test larger components requires knowledge of the time-space correlation of the forcing function. This requires continued research into the character of the sound field.

To date the problem appears too complex to obtain the required degree of reliability from a purely theoretical analysis. However, because of timing and cost of testing, it is of extreme importance that fundamental analytical tools be developed to aid the engineer in designing a reliable structure, of minimum weight.

The solutions to the problems of sonic origin will involve many new and unique concepts. For example, one area of considerable interest and promise for future research is the field of damping devices or techniques. A series of methods have been investigated at Boeing with the most promising being a free mass damper. Figure 6 illustrates one of the arrangements. It should be noted that at resonance a damping of approximately 55 percent of the original amplitude was achieved for a weight penalty of 3 percent of the basic panel weight.

CONCLUSIONS

Present capabilities developed through a series of both laboratory and engine environment testing will answer many of the sonic problems of the immediate future with aluminum designs. The advent of high strength steels and super-alloys and ceramics present new problems. However, with the acoustical environment anticipated for the immediate future many of these problems can be solved using techniques developed during the aluminum program.

During the coming decade, structural arrangements capable of operating in an acoustical environment of 175 to 180 db will be required. Service lives of from one minute to 3 to 4 hours at these levels can be expected.

The past and present maintenance costs that are directly attributed to sonic fatigue dictate the necessity that we not only aggressively pursue the outlined research goals, but insure that all available criteria are considered in all new designs.

APPENDIX "A"

SONIC FATIGUE LIFE PREDICTIONS

The use of accelerated environmental and laboratory testing leaves the service life in doubt. A theoretical method was developed for correlating life improvements in horn and environmental tests to corresponding improvements in service life and predicting service life from environmental testing.

The essential steps of the method are:

1. For the particular component in question, sound levels and number of resonant responses must be estimated for various segments of the aircraft mission.

   The relative stress levels σ/σ_o for the various segments of the mission are then obtained from Figure 7 using as a base the test sound levels.

   σ = root mean square stress for segment under consideration.

   σ_o = root mean square stress for test condition.

2. To relate sinusoidal (constant amplitude) loading fatigue life to that obtained from a random type loading it is necessary to construct random s-n curves. Using a Rayleigh load distribution for the random loading, a random s-n curve is developed and plotted on Figure 8. The associated sinusoidal (constant amplitude) curve is also shown in Figure 8.

3. In order to compare the service life to the test life demonstrated by a random environmental test, a random service life s-n curve is required. This curve relates the number of cycles to failure in service to a non-dimensionalized r.m.s. test failure stress (stress rates: σ/σ_1. To construct this random service life s-n curve, several values for the r.m.s. stress ratio for the environmental test are selected. The stress ratio for the various segments of the service operation are determined by multiplying the selected r.m.s. test stress ratio by the stress ratio for each flight segment. A Rayleigh distribution is then applied to these determined stress ratios and the total damage incurred per year is accumulated by $\sum_0^j \frac{n_i}{N_j}$, where 0 to j includes the Rayleigh distribution of occurrances near and over the ultimate stress of the material. Shown in Figure 8 are two random service s-n curves obtained by applying the above procedure to the example mission profile of Figure 9.

USE OF CURVES

The superposition of these curves on Figure 8 permits the prediction of the increase in environmental random test time from an increase in horn test

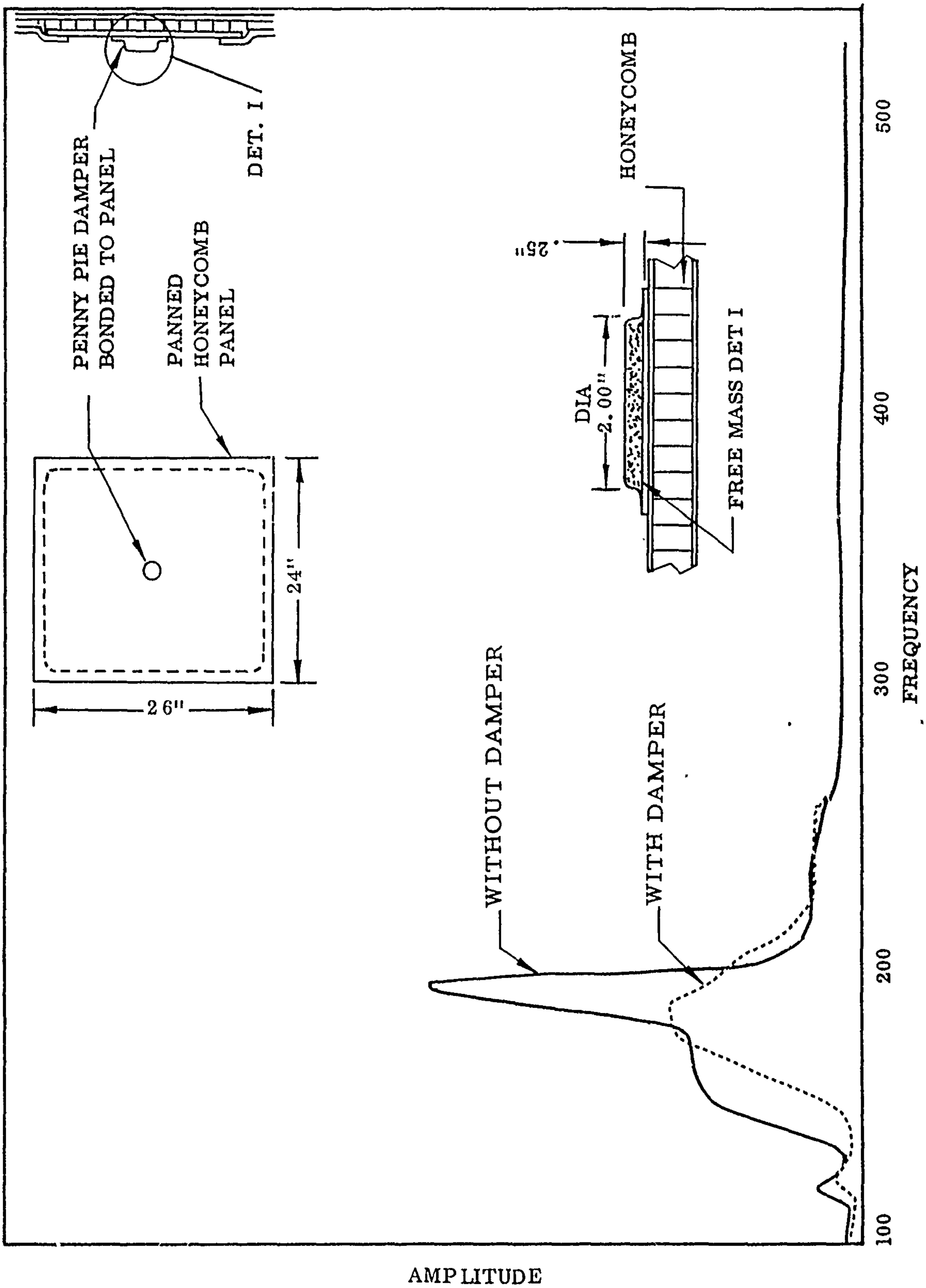
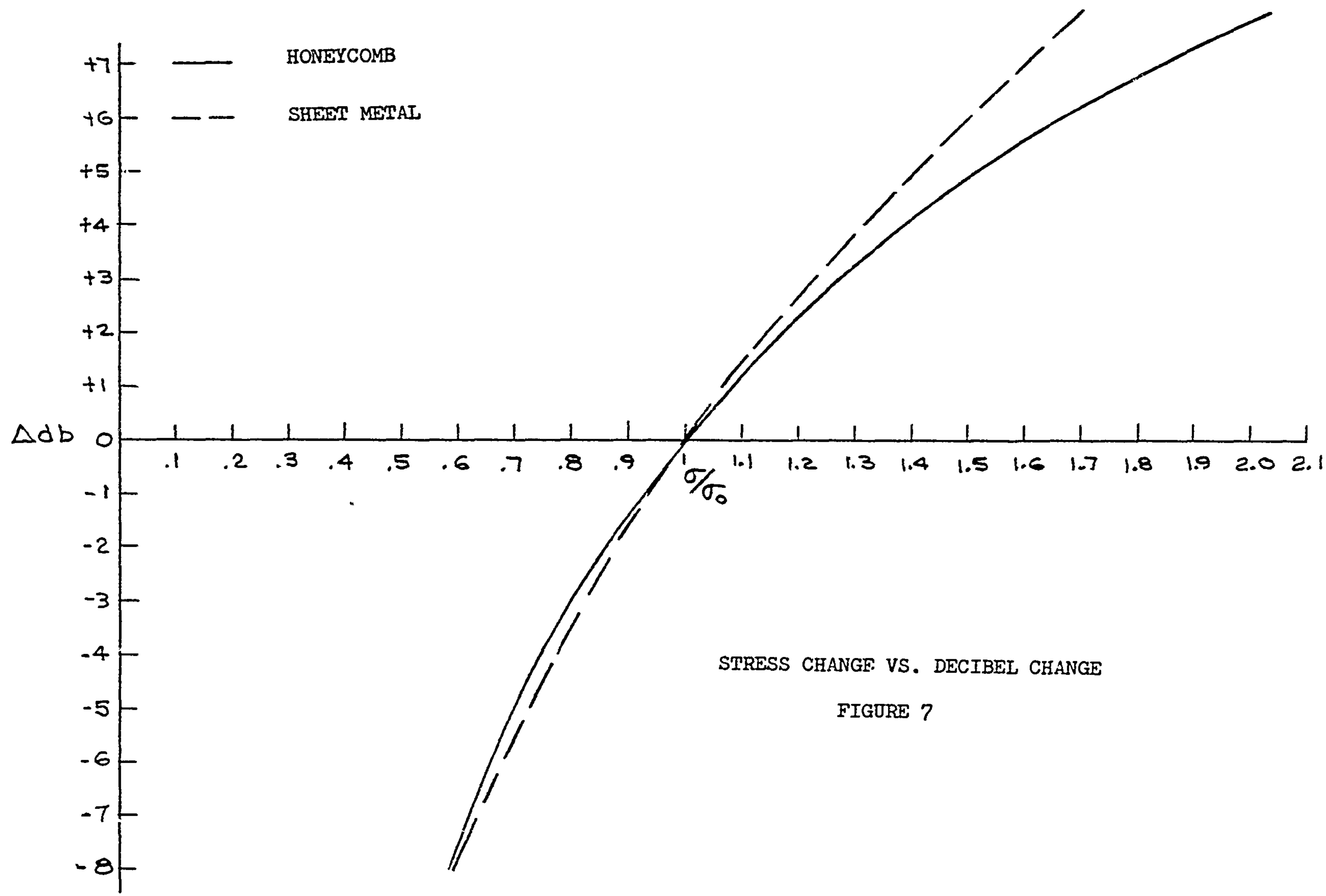


FIGURE 6 A2b 837 PANNED HONEYCOMB DAMPER EVALUATION-PENNY PIE .25 DEPTH



STRESS CHANGE VS. DECIBEL CHANGE

FIGURE 7

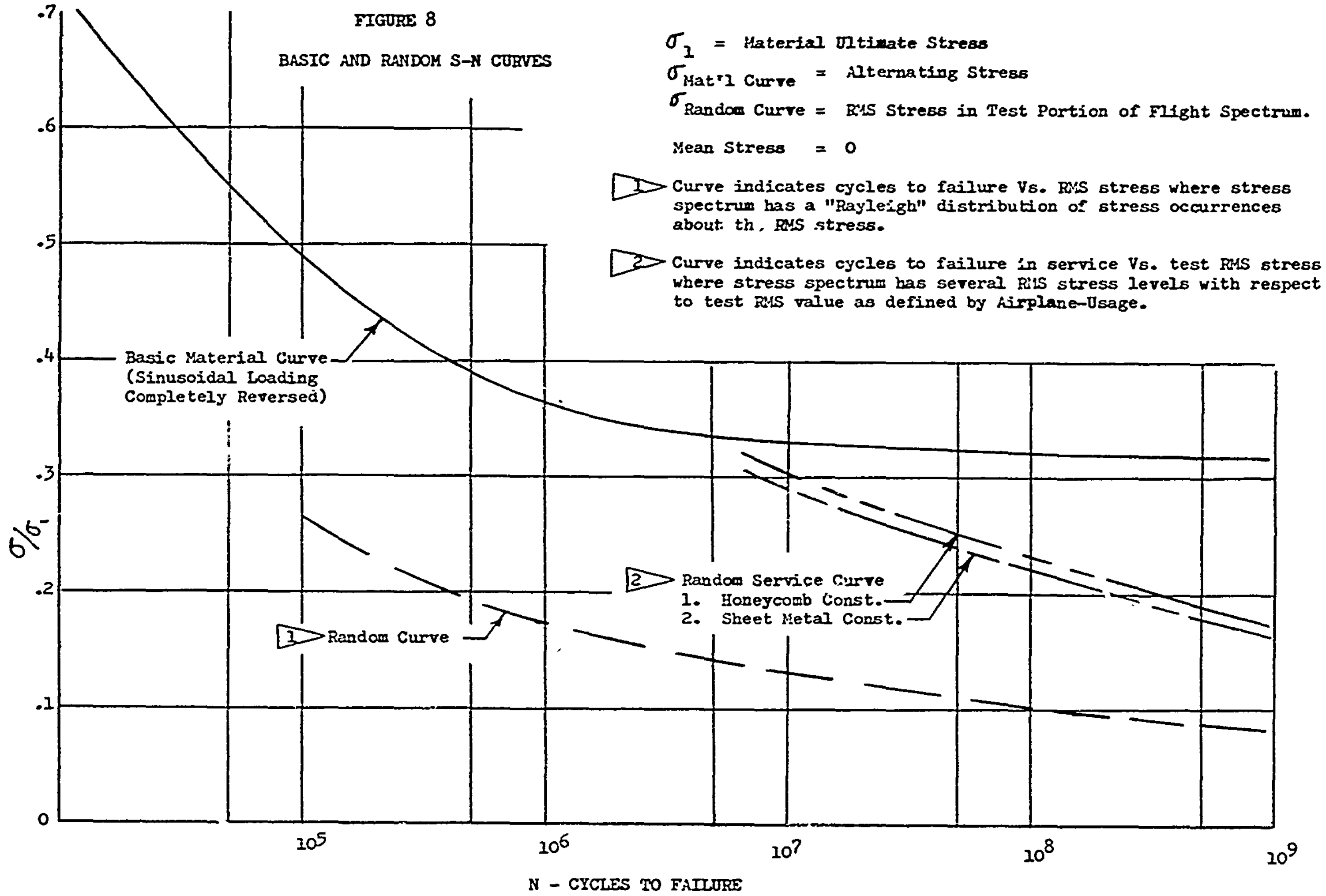


FIGURE 8

BASIC AND RANDOM S-N CURVES

FIGURE 9

ONE YEARS SERVICE OPERATION - EXAMPLE

Mission Segment	1	2	3	4	5	6
db [1]	0	-1	-4	-2.68	-3.68	-6.68
P/P_o [2]	1	.893	.631	.735	.653	.463
Sheet Metal [3]	1	.940	.780	.850	.800	.658
Honeycomb [4]	1	.930	.755	.820	.700	.640
Seconds of Operation/ Year	644	114	232	3985	4250	16630
Cycles of Max. Stress [5]	161000	28500	58000	995000	1066000	4162000

[1] Change in sound level with respect to test levels.

[2] Sound pressure/test sound pressure.

[3] Stress Ratio stress/test stress (Reference Figure 7).

[4] Stress Ratio stress/test stress (Reference Figure 7).

[5] Cycles based on resonant frequency of 250 cps.

time or to predict increases in service life from increases in either horn or random environmental test time.

It is also possible to predict increases or decreases in test time or service life produced by specific changes in decibel level.

Example I:

1. Component Part "A" failed in 1.5 hours of random environmental test time. Resonant frequency of sheet metal part of Component "A" which failed is 225 cps.

2. Determine service life of Component "A".

Number of Cycles to Failure

$$N_R = 225 \times 3600 \times 1.5 = 1.215 \times 10^6 \text{ Cycles}$$

Apparent Random r.m.s. Stress Amplitude

$(\sigma/\sigma_1)_R$ = .170 From Figure 8 (Random Environmental Curve)

Number of Cycles to Failure In-Service

$N_S = 7.0 \times 10^8$ From Figure 8 Opposite $(\sigma/\sigma_1)_R = .170$ on Service Life Curve

Service Life

$$\text{Hrs.} = N_S/225 \times 3600 = 7.0 \times 10^8/225 \times 3600 = 864 \text{ Hrs.}$$

Example II:

1. Part "B" a honeycomb panel fails after 1:15 of random environmental test time with an engine producing 160 db in the 150 to 600 cps band width, or an equivalent of 800 service hours.

2. Part "B" fails after 0:30 of horn test time at 160 db with a resonant frequency of 250 cps.

3. Part "C" an improved part "B" fails after 4:00 of horn test time at 160 db with a resonant frequency of 250 cps.

4. Estimate the service life of part "C" if more powerful engines producing 163 db in the 150 to 600 cps band width are used, as well as the service life if the original 160 db engines were used.

Number of Cycles to Failure ("B") (Environmental Test)

$N_{R_B} = 1.25 \times 250 \times 3600 = 1.125 \times 10^6$ Cycles

Apparent Random r.m.s. Stress Amplitude (Environmental Test)

$(\sigma/\sigma_1)R_B = .172$ From Figure 8 (Random Environmental Curve)

Number of Cycles to Failure Part "B" (Horn Test)

$N_{B_H} = .5 \times 250 \times 3600 = 4.5 \times 10^5$ Cycles

Apparent Stress Amplitude Part "B" (Horn Test)

$(\sigma/\sigma_1)B_H = .385$ From Figure 8 (Basic Material Curve)

Number of Cycles to Failure Part "C" (Horn Test)

$N_{C_H} = 4.0 \times 250 \times 3600 = 3.6 \times 10^6$ Cycles

Apparent Stress Amplitude Part "C" (Horn Test)

$(\sigma/\sigma_1)C_H = .338$ From Figure 8 (Basic Material Curve)

Ratio of Stress Improvement Part "C" to Part "B"

$(\sigma/\sigma_1)C_H \quad (\sigma/\sigma_1)B_H = .338/.385 \doteq .878$

Change in Random r.m.s. Stress from 3 db Increase

$(\sigma/\sigma_1) + 3db/(\sigma/\sigma_1) = 1.275$ Figure 7

Predicted Random Service r.m.s. Stress Part "C" (163 db)

$(\sigma/\sigma_1)R_B \times 1.275 \times (\sigma/\sigma_1)C_H/ \ (\sigma/\sigma_1)B_H = (\sigma/\sigma_1)R_C$

$.172 \quad \times 1.275 \times \ .878 \quad = .1927$

Predicted Number of Cycles to Failure Part "C" in Service (163db)

$N_{C_S} = 3.5 \times 10^8$ From Figure 8 Opposite $(\sigma/\sigma_1)R_C = .1927$ on Service Life Curve

Predicted Service Life (163 db)

$N_{C_S}/250 \times 3600 = 3.5 \times 10^8/250 \times 3600 = 389$ Hours

Predicted Random Service r.m.s Stress Part "C" (160 db)

$(\sigma/\sigma_1)R_B \times (\sigma/\sigma_1)C_H / (\sigma/\sigma_1)B_H = (\sigma/\sigma_1)R_C$

Predicted Number of Cycles to Failure Part "C" in Service (160 db)

$N_{C_S} = 2.2 \times 10^9$ From Figure 8 Opposite $(\sigma/\sigma_1)R_C = .151$

Predicted Service Life (160 db)

$N_{C_S}/250 \times 3600 = 2.2 \times 10^9/250 \times 360 = 2440$ Hours

FORUM - SESSION III

Session Chairman:
Mr. C. E. Reichert, Wright Air Development Center

Panel Members:
Mr. Paul Kuhn, National Aeronautics and Space Adm.
Mr. I. Bouton, Norair
Mr. R. H. Christensen, Douglas Aircraft Co.
Mr. Phil Parmley, Wright Air Development Center
Dr. B. J. Lazan, University of Minnesota
Mr. Andrew K. Hepler, Boeing Airplane Co.

Editorial Note: Attention is directed to the editorial policies presented in the Preface which were followed in editing the discussions of the Forum.

CHAIRMAN, C. E. REICHERT:

I think we will give our first attention to the question cards. I am going to ask Mr. Kuhn if he will start out with one and we will have the rotation in which they appear on the floor.

MR. KUHN:

This question is from Mr. Bennette of Chance-Vought: "What information is available regarding reduction of original static strength versus percentage of available fatigue life use, for structural components?" I think the percent of available fatigue level used is an impossible parameter to use for estimating reduction of strength.

To the next part of the question then: "Is there an appreciable reduction of static strength before onset of initial cracks, or are cracks the only reason for a reduced static strength?" There is not very much information on change of properties before onset of the crack. In some cases there is a slight reduction and in some cases there is a slight increase. In either case, however, the effect is negligible for practical purposes.

I think the third part of the question is the same thing, so I can skip that.

CHAIRMAN:

I'll have Buzz Bouton pick one up.

MR. BOUTON:

I have two questions here that touch on the same thing. On my statement that 63% of the airplanes that have design required life will probably have failures before attaining the average life specified; the question is asked, "Would you recommend the life requirement figures be multiplied by 1/.63?" My statement was based on the fact that if you take a series of specimens that have a certain average life, say 5000 hours, and operate them on the basis of 5000 hours, less than half of them will still be operating when 5000 hours is reached. Looking

at this from another standpoint, sixty-three per cent of them will already have failed by the time 5000 hours is reached. I haven't done a real good analysis of it. I was just extrapolating this kind of thinking to the fact that if our load spectra are based on the average life figure, which as far as I know they all are, for example, the average number of times that an aircraft exceeds five G in 10,000 hours, I think we should consider this statistical variation in our thinking.

I also have an addition to that question. Incidentally, this question was asked by Ed Griffin of BuAer and by Major Griswold of Headquarters, ARDC. Major Griswold went on to ask if I would recommend that the life requirement figures be multiplied by one over point six three. It was not my intent to do that. I think it's quite possible that, if you have a 5000 hour life requirement, while 63% of them may have failed at 5000 hours, the point at which they would begin to fail, would only be 4800 hours, or something like that. They are not quite the same thing.

CHAIRMAN:

Roy, let's have one from you, please.

MR. CHRISTENSEN:

Mr. J. D. Marble from GE: "From your proposal to Professor Gatewood, it seems that now is the time to transfer to strain-analysis rather than stress-analysis." He wants to know if I believe this is correct. I say yes. Maxwell Morris energy methods are now being used in the Air Force. The Air Force is sponsoring, I understand, a NATO conference in September in Aachen, Germany, on just this subject; the successes that have been achieved in this field.

CHAIRMAN:

Can we let Dr. Lazan have one?

DR. LAZAN:

The first one I received is from W. R. Winslow of WADC: "Are any visco-elastic materials, known to have good adhesion and strength properties for use as bonding agents, useful also for dissipating damping energy?" Well, actually, all adhesives, partly tacky, have radiological properties which include the possibility for dissipating damping energy; the tacky ones, generally speaking, dissipate much more energy than can the structural hard adhesives, but both actually can dissipate significant energy. It's just a continuous spectrum, one might say, of damping and strain properties as you go from the hard, strong adhesives to the soft, tacky ones. In this connection, I think as one looks to the future, with greater use of structural adhesives, it might be desirable to think not only in terms of the strength of the structural adhesive, but also, possibly, the damping properties. Instead of selecting the hardest, strongest, or most brittle adhesive, perhaps these properties can be considered along with damping properties in selecting an optimum adhesive for a given job.

CHAIRMAN:

Phil, how about you grabbing one now?

MR. PARMLEY:

I have one from Mr. Galef, Radioplane Division of Northrop; he asks: "We have frequently been instructed (by the Aircraft Laboratory) to design for a discrete gust, one minus cosine shape, with a length of 25 wing chords. A zero margin on yield is considered satisfactory for the design value gust. In light of the present concern over fatigue, may we expect a different criteria for present or new projects?" I don't believe so. We have very seriously considered converting to the power spectral density type gust requirement, but we're not completely convinced that we can eliminate the one minus cosine design gust. I don't think that the aspect of fatigue will change our gust criteria.

CHAIRMAN:

How about your grabbing one, Andy?

MR. HEPLER:

I have two here that are very similar, so I think I can answer them at the same time. One is from Mr. Cohen, AMC Headquarters; one from Mr. Sangster, Curtiss-Wright. I will read Mr. Cohen's: "Since the root of the entire sonic fatigue problem is generated by the propulsion unit, why is not more emphasis placed on solving fatigue problems on the propulsion unit itself?"

Efforts have been made in this direction through the use of sonic suppressors and optimum location of the power plant on the aircraft. One other fact to keep in mind in considering sound suppressors is that, associated with any sound suppressor is, a certain degradation in range, and you must balance this off with what it takes to resolve your acoustical problems with structural changes.

MR. PARMLEY:

Well, I've got a number here, most of them very loaded. Mr. Bouton of Norair asks: "Why is it important to know what the shortest life is? We have always had the situation where the plane might be lost on the first flight. Therefore, it should be a question of whether an early failure is remotely probable rather than can it happen." We don't know which airplane is going to be the one that will fail first, and it is quite likely with some probability that you can describe, that the first airplane to come off the manufacturing line would be the last airplane to leave service. Also, it could very probably be that that first airplane would be the one which would receive, during its lifetime, the worst load experience. Each of these parameters is statistical in nature and probably can be described, but the thing that the Air Force wants to know is, with some definition of some statistical quantity, what is the life probability of the worst airplane. We then would have a definition which we could not only use ourselves but also give to the using command and higher headquarters for purposes of basing procurement schedules on a system which will have a life that is defined. In talking about the worst airplane, let me take one of the others that Buzz has asked me. He says: "Your mention of 5% probability of failure, as an acceptable figure, interests me. Can you expand on it? How was it chosen? Does this really mean that it would be acceptable to have one out of twenty aircraft failing catastrophically?" No, it is not acceptable for one airplane out of twenty to

fail catastrophically, and this is not what I implied. I probably went out on a limb in the first place even mentioning a 5% probability, and maybe that was a poor time to have said it; but, if we have a complete fatigue analysis and the results of structural component and complete airplane tests, we have gained a considerable amount of information about the structure and its characteristics. If, then, the items that we know to be most critical, or most sensitive as far as fatigue is concerned, are inspectable, I think we can design that structure compatible with 5% probability of failure at the design life and carry it by inspection. By monitoring the load experience of an aircraft, either with a recorder or on a statistical basis, and instituting a systematic inspection and maintenance procedure when it reaches a prescribed life, more efficient utilization of the inherent life of an aircraft structure can be made. I don't believe that this kind of selection of a probability design and maintenance procedure implies catastrophic failure.

MR. BOUTON:

There are three questions here that I think will touch on the use of statistics in the design of aircraft. One is from Les Fero of the Martin Company: "Today's design of tomorrow's space vehicles may require a more rational approach to structural design criteria and can be supported by statistics. The increased ignorance factors and flight times of years rather than hours may justify increased safety factors. Would you care to comment?" I feel that something like this question of increased ignorance factors is involved. These are things which you can't put into statistical criteria. You may have to do it on a judgment basis, but you should be able to estimate what your ignorance factors are. For instance, I once predicted the loads on a certain structural component to be a thousand pounds plus or minus a thousand pounds. If that's all the better you can estimate it, that is what you have to work with. I think you could insert a term in your structural computations, not necessarily in your statistical analysis, which would compensate for the ignorance factor. This could also take care of this question of years of life rather than hours of life. If you want to operate for years, and you have a certain probability of failure over these years, this would just have to be accounted for.

Another question is: "How does the eight parameter program take care of the flow separation or buffeting inputs in the frequency range?" Well, basically I would say that the 8 channel data would tell you where you are in the flight spectrum. This, then, broken down on a probability basis and combined with analytical or flight test load data, could determine what loads you would encounter when you were in that particular flight regime.

The last question was from Mr. Pettingall of Douglas Aircraft and Mr. Flomenhoft of North American Aviation, Columbus Division: "Since so many parameters are needed to determine airplane loads in flight, and these don't include dynamic effects from gust loading, would it not now be desirable and practical to obtain statistical measurements of structural loads directly by use of a strain-gauge bridge in some critical location in service airplanes?" I might point out that in connection with our statistical criteria program at WADC, one of the questions that we've been asked to answer is: "Is there some way that you could measure the flight loads of airplanes which are fully instrumented and establish correlation between the 8 channel measurements and the structural loads values - let's say the wing root bending moment, the mid-span torsion, the tail

loads and so forth?" This may be feasible. It is possible that you could establish the relationship that way without having to directly instrument the operational airplanes for measuring loads.

CHAIRMAN:

Mr. Kuhn, do you want to take over now?

MR. KUHN:

This is from Mr. F. R. Stone, WADC: "Will you please outline the significant accomplishments or 'break throughs' on elimination of aircraft fatigue problems which have evolved within the past ten years?" I hope I don't offend too many publicity-minded people by saying that the research man, certainly in his own field, never sees any "break throughs." He sees the gradual accumulation of knowledge in a stream that just goes on and on.

The next part of the question is: "The 'Gooney Bird' (DC-3), which preceded this period, appears to have been a design miracle. Why?" Well, the explanation is generally given in terms of design stress; but, I hate to admit that, in spite of almost thirty years in the business, I still don't know what a design stress is. I think it can be explained in better terms by saying that the design twenty or twenty-five years ago was just more conservative. There were many simplifying assumptions made that resulted in the actual stresses in the structure being much lower than they are now. The designer now utilizes the material much more fully than he did twenty-five years ago.

MR. CHRISTENSEN:

This question is from John Klocksiem, Hughes Tool Company, Aircraft Division: "With reference to fretting corrosion, would you care to make any recommendations or comparisons of the relative efficacy of various methods of combatting fretting. For example, in the case of a bolted joint with steel fittings subject to cyclic loading, what fits, surface coatings or surface treatments (such as shot peening) are good, and how effective are they for fatigue?" I'm sorry I can't remember any data on steel joints or fittings. However, we have made some tests on heavy aluminum plates with thin sheets of foil of gold, platinum, nickel plate and aluminum in the paying surface. These joints were made with extremely high carping stresses to purposely cause this type of failure to occur, that is, due to the wear of the chafeing away from the bolt holes. Some of these treatments gave improvements over the untreated joints. I can't remember just which of them; the gold, platinum, nickel plate, etc. But I think it should be remembered that the fretting or chafeing type of failure - at least in general in the past - has always occurred later than the type of failure that would be caused by a loose joint for instance. A tightly clamped joint lasts longer than one that isn't clamped together so tightly.

Another question from Ron Smith, Radioplane: "Temperature and fatigue - creep may alleviate fatigue in some instances; yet in contrast, creep relaxation of a preloaded bolted joint may induce a lower fatigue life for the joint. Has work been done on this?" This is in line with some of the work we are just starting, and so far, the only data we have is on plain materials. Eventually, though, we are going to have some data on built-up joints.

CHAIRMAN:

Professor Lazan?

DR. LAZAN:

Apropos to chafeing or fretting or whatever you want to call it, I have one here from Klint of GE: "Can fretting corrosion be minimized in slip damping joints by use of lubricants, such as dry lubricants?" We have tried this on several occasions and it does improve the situation; however, slip joints that are optimized for maximum damping are also, as I indicated previously, optimized from maximum fretting, chafeing, and corrosion, and under these circumstances we could not find any lubricants dry or otherwise that would do the job.

Shall I go on to another question?

CHAIRMAN:

Yes, please, sir.

DR. LAZAN:

The next one is from Howard Wright, National Bureau of Standards: "What about 'fatigue', i.e., deterioration of the visco-elastic materials with vibration and time?" The visco-elastic materials will begin to deteriorate at very high strains. I don't know how you define fatigue under these circumstances but there is a tendency for separation and the like. The typical material that we work with could be carried to a shoe strain of five. This may seem rather ridiculous but it is five; that's what it takes. The value used in the slide as a fatigue limit was one. In a sense, the slide that would show visco-elastic damping was conservative by this one factor. In the case of damping tapes, it depends a lot on the configuration. Shear strains in the order of one are generally used; very much lower than the potential. The limitations on the visco-elastic layer may not be fatigue or deterioration at high strain, but may be higher temperature. The very high damping of these materials will result in high internal heat generation unless the visco-elastic layer is quite thin so that the surrounding metal surfaces can dissipate the heat. The internal heating and temperature increase can possibly be a factor.

MR. KUHN:

The next question is from Mr. Harmsworth, WADC: "In your talk, and that of Mr. Peterson, the point was made that the reduction in fatigue strength becomes significantly less severe when the notch radius becomes smaller. Would you care to comment on this observation in the face of evidence that very small inclusions (notches in form) and cavities in some materials adversely affect the fatigue strength of these materials?" The curve which I showed as an example was for aluminum alloys. There the curve of stress concentration against notch radius bends over rather gradually. In materials which are sensitive to inclusion, the curve rises very steeply at the origin and approaches very quickly, the asymptote which is a theoretical value and, in that case, it is very important to note whether the inclusion is just above or below that limit.

CHAIRMAN:

In answering this next question, Buzz would like to use the blackboard. So we've cranked him up and we're ready to go!

MR. BOUTON:

The question that I was asked by Major Griswold of ARDC was: "Did you not use two different (implied) definitions of failure in (1) discussing that all failures are due to 'adventitious' loads and (2) suggesting that criteria eliminate only a majority of fatigue failures?" What I tried to get across here is this: Considering the airplane lifetime, we start out with an airplane that has, what is commonly called, an ultimate strength; and you must remember that this is usually a 99.9% point in determining the ultimate strength. So you have an average value of 50%. If you consider that you have scatter between values, you also have a one in ten probability of getting one load, one in a hundred that you'll get another, one in a thousand that you'll get a third, and so forth. So you have a scatter band of loading to which a specimen may be subjected. That is part of what your factors of safety are trying to take care of, at least incompletely. Assuming no accident during the life of the plane, these probabilities of loads of one in ten, one in a thousand, and so on, remain more or less constant for the lifetime. So you end up having, presumably, an increasing probability of failure with increasing life. If you drew the probability of failure on a log curve - the probability of failure early in aircraft life would be quite low. As time passed, the probability of failure would increase. It is my belief that we are interested in the area under this curve, which gives the probability of the whole airplane failing sometime during its lifetime.

Now just one other subject if I may quickly, I will point out something that Professor English of U.C.L.A. has pointed out to us in connection with our statistical criteria contract with WADC. He hasn't documented it yet; it is just a thought right now. He suggests that possibly some of our peak problems with higher strength materials come not from the fact that they are more brittle but from the fact that they have less scatter in their strength characteristics. In other words, the average strength value of these materials is much closer to the guaranteed strength value. So all of these things have to be included in the statistical approach to the problem.

CHAIRMAN:

Thank you. (To Mr. Hepler) Do you have one more there?

MR. HEPLER:

I have one from Lt. Iwasko, ARDC, Wright-Patterson. It says: "Do you feel that subjecting an aircraft to a high db level within a closed shell is as harmful as the same db level in the open air?" I take it this is related to the new facility to be developed here and my comment in this area is that it is not different. The only thing that you must be sure of is that you are duplicating the acoustical field in the immediate area, and that is right adjacent to the component which you are testing. You have to do this by moving the specimen around in the closed chamber until the force or acoustical field duplicates the desired sonic environment.

CHAIRMAN:

I see we've wound up pretty close in time. I'd like to express my thanks to the people who have been on this panel session. Thank you very much.

INTRODUCTION TO SESSION IV

COLONEL HARVEY P. HUGLIN

WRIGHT AIR DEVELOPMENT CENTER

The subject of this session will be "Simulation of Fatigue Loads in Testing." Your chairman will be Mr. Clarence R. Smith. Mr. Smith majored in physics at Stanford University. He joined Convair in 1941 and now is a design specialist at Convair. He has authored many papers on fatigue testing, life prediction and structural design. He is an active member of the American Society for Testing Materials where he is serving on the Committee of Fatigue in Aircraft Structures. He is also a member of the Society for Experimental Stress Analysis. Mr. Smith is currently serving at the University of California as a staff Member on Strain Gauge Techniques.

COMPLETE VEHICLE - FULL SCALE - TESTING TECHNIQUES

By

R. E. Watson
and
L. L. Gore

Boeing Airplane Company
Wichita, Kansas

This presentation will cover fatigue testing of complete airplanes generally, with specific references to the B-47 and B-52 test programs.

The significance of fatigue analysis and laboratory testing in determining critical areas and test loads will be discussed. The diminishing significance of fatigue analysis and laboratory testing as the test spectrum approaches the service spectrum will be noted. Various types of load spectra are compared and reference will be made to the factors which influence their selection.

The presentation will include mention of the systems used for supporting the airplane during testing and the means used to apply loads. Reference will be made to the need for adequate strain gaging to provide correlation with component tests.

Experiences of the difficulties encountered in detecting small cracks because of high propagation rates in heavy structures will be discussed.

Information will be presented on the value of "tear down" inspection of the specimen after test completion, in determining true damage.

The concluding statements will cover the manner in which the test data can be used in establishing life expectancy and directing operational requirements.

INTRODUCTION

The increased flying hours and low attrition rates attendant to the military readiness program of the United States Air Force, coupled with the more severe operational usage, have re-emphasized the matter of structural fatigue prevention. As a result, structural life evaluation is now, and will continue to be, an important consideration in military vehicle design. It is, therefore, most appropriate that the many facets of structural life evaluation be considered at this time.

Structural life evaluation programs are currently in progress, at the Boeing Airplane Company, under contract with the Air Force, on three military airplanes, the B-47, B-52 and KC-135. In addition, Boeing has an extensive life evaluation program on the commercial 707 Jet Transport. The various phases of a typical military structural life evaluation program are illustrated in Figure 1. As is shown one phase of the program is "complete vehicle"[(a)] testing. Boeing has recently completed such a test on the model B-47 and is currently engaged in full scale tests of the B-52F and of the B-52G. This presentation will cover the subject of complete airplane testing from the point of view developed in these three specific programs.

PURPOSE OF COMPLETE VEHICLE TESTS

In formulating a life evaluation program it is important to objectively consider the need for a complete airplane test. Such a test is inherently quite expensive and time consuming. On any operational model it will usually be prohibitive to test more than one vehicle. Considering symmetry, this provides essentially two test specimens. This is still too few for much insight into scatter effects. These limitations, not withstanding, let us consider some of the advantages and purposes of such a test.

First a complete vehicle test represents the true mechanical environment, i.e., it allows for the presence and interaction of all structural components, not just selected ones. Secondly, it allows observation of all the structure and brings to light any unsuspected critical areas not adequately covered by analysis or specimen test. Finally, a complete vehicle test gives a realistic basis for development of service inspection and repair requirements.

(a) The terminology "complete vehicle", as used here, refers to two or more major components which are structurally combined; e.g., a wing and fuselage section.

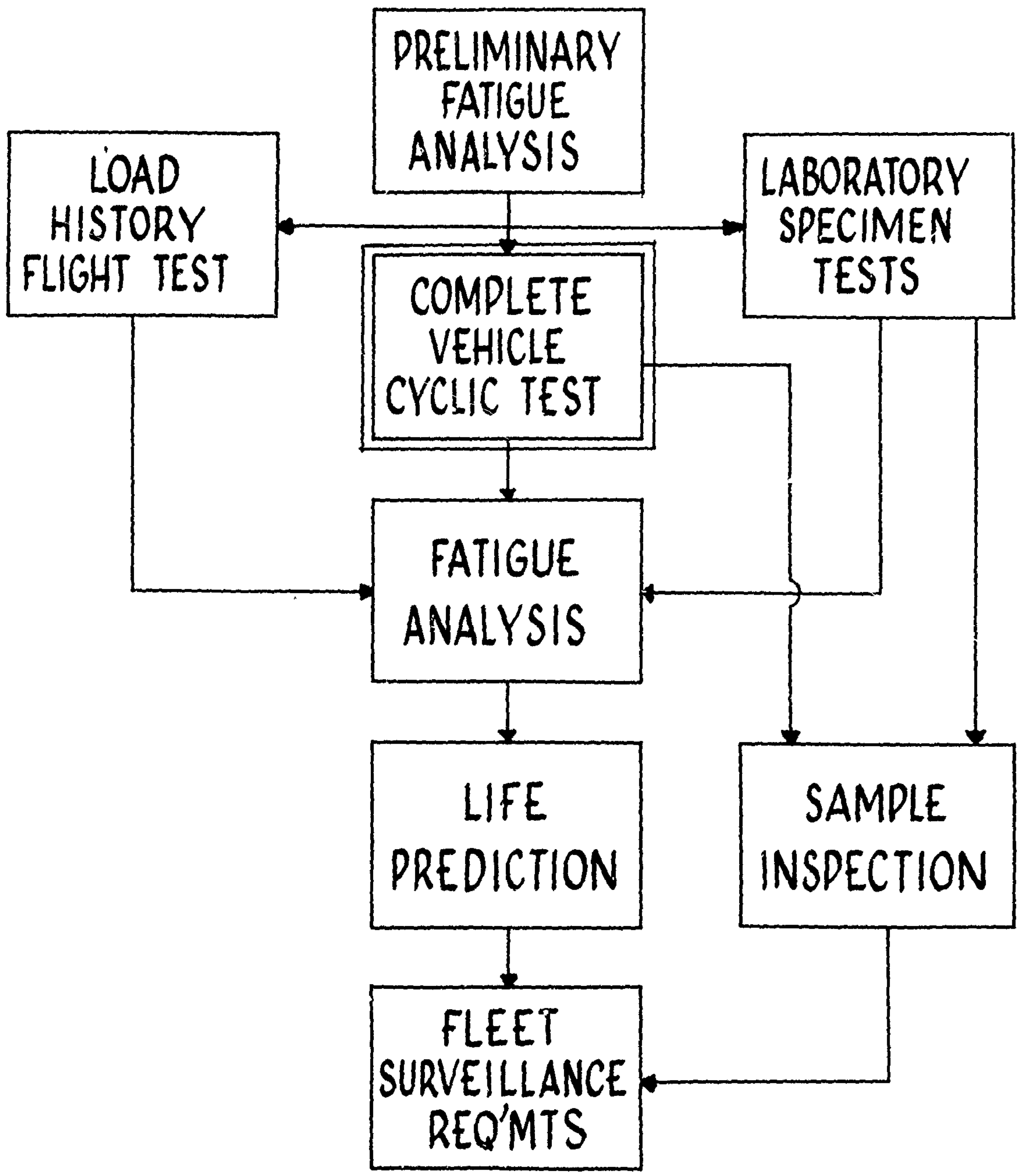


FIGURE 1

Typical Life Evaluation Program

SCOPE OF THIS PRESENTATION

This presentation will be limited to structures which are critical for loads requiring analytical simulation. This excludes such items as pressure sections, where the phenomena producing loads are well known and can be accurately reproduced. In such cases the structure may be tested to the service load spectrum until adequate life is demonstrated and no fatigue analysis is needed.

For most components, however, the service loads are not precisely known. In such cases, analytical methods are required to determine the test loads to be applied to the structure and to correlate damage rates between service and test.

For the B-47 and B-52 programs previously mentioned, the principal areas under evaluation were the wing, fuselage and the empennage primary structure. The method employed was to pre-select certain locations in the structure which were thoroughly analyzed for service life. Test loads were then derived to simulate the service effects.

SELECTION OF CRITICAL AREAS

In the selection of potentially critical areas, consideration was given to the relative ability of the structure to withstand fatigue, to the frequency and magnitude of stress encountered in service, and to the consequences of failure should it occur.

For the B-47, the most critical areas were known from service failures. The areas of concern were the lower wing skin splices at the body side, BL 45, and at the inboard nacelle, WS 354.

Tests were conducted on three airplanes at the Boeing Wichita facility, the National Aeronautics and Space Administration facility at Langley Field, and the Douglas Aircraft Company facility at Tulsa.

Based on service knowledge of the critical areas, the B-47 test was designed to apply calculated damage at BL 45 and WS 354 on the wing lower surface. Compatible vertical bending moments were applied to the wing splice at WS 642 and to the fuselage splices at BS 515 and 861. A separate test was run to evaluate the effects of repeated lateral tail loads on the fin and the aft body.

On the B-52 series, inspections of high-time aircraft have revealed no indications of fatigue in primary structure and the test programs must rely completely on analytic determination of critical areas. This of course, will be the situation on all new models.

The first consideration in selection of critical areas, i.e., the relative ability of the structure to withstand fatigue will, in most cases, be adequate to determine the points for analysis. Usually, joints and cutouts will be more critical than "basic" structure. Since a number of control points is required to determine the distribution of test loads, analysis of other significant transitional areas will usually suffice.

On the model B-52F, a total of twenty-two splices (ten wing, eight fuselage, two fin, and two stabilizer) were analyzed. In this cyclic test, loads are applied to produce calculated damage on the wing, fuselage, and empennage at these twenty-two locations.

The B-52G is structurally similar to the prior models in the fuselage and empennage. The wing, however, is of completely different structural design. Through the use of 80 foot skin panels (upper surface panels are integral extrusions) all skin and stiffener splices are eliminated between the BL 55 and WS 1025 production breaks. In addition to the upper and lower surface splices at these two locations, analysis of service damage was made at other locations such as access holes, boost-pump cutouts, and fuel drain holes. In the B-52G cyclic test, loads are applied to produce calculated damage on the wing only (mounted on a partial fuselage) at six locations. Compatible bending moments are applied to the mid-fuselage to provide realistic wing to body attachment loads. The remainder of the fuselage and empennage, although analyzed, are not included in the airplane test because of similarity to the B-52F.

SERVICE DAMAGE ANALYSIS

The linear theory of cumulative damage[b] is used in all three airplane test programs. The procedure for determining service damage rates is as follows. Information as to the type and number of missions to be flown is obtained from the Air Force. These data are resolved into mission profiles showing the timewise variation of speed, altitude, and gross weight as well as a description of the mission.

Aeroelastic loads data are then calculated over a wide range of speeds, altitudes, weights, and steady state load factors. From these data, curves are derived showing "1g" stress and $\Delta f/\Delta n$ variation with V, h, and GW for all stations chosen for analysis, as shown in Figure 2.

(b) Miner, M.A.: Cumulative Damage in Fatigue, "Journal of Applied Mechanics", Vol. 13, No. 3, pp A159-A164, September 1945

B-52G WS 282
SEA LEVEL & CONSTANT C.G.

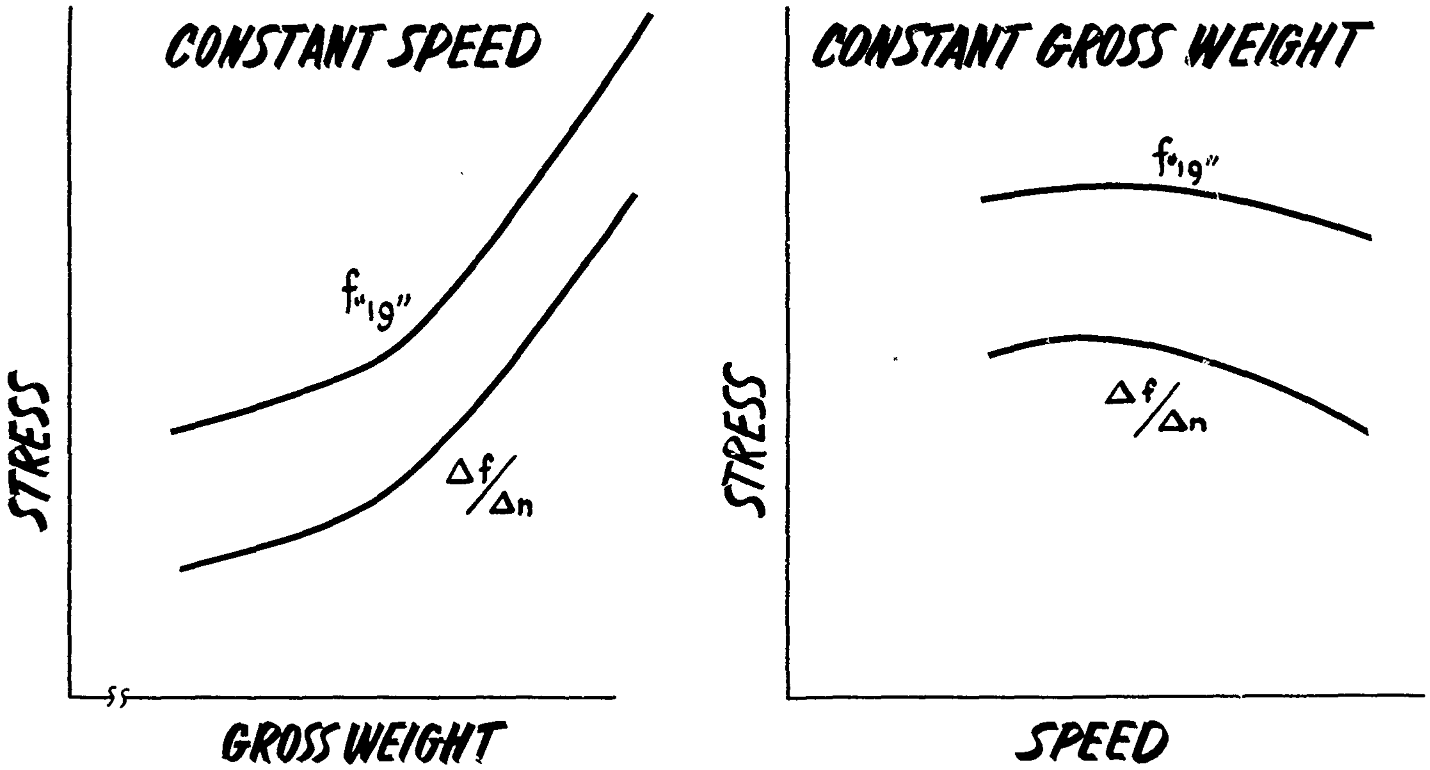


FIGURE 2
Variation in $f_{"1g"}$ and $\Delta f/\Delta n$

Within these profiles, consideration is then given to ground-air-ground (GAG), gust, maneuver, touch-and-go, landing, runway roughness, braking, and engine runup. The load factor amplitude and frequency of occurrence is derived from a variety of sources. Gust effects, for example, were based on 70 million miles of airline and military experience data and were resolved as derived gust velocity exceedances. Flight test programs are currently underway to obtain additional acceleration and load data, especially at low altitude. Allowances are made for dynamic magnification in the response of the airplane to gust, landing impact, and taxi. By summation of all effective load inputs, a spectrum of stresses is obtained for each analysis station.

In the fatigue damage calculations, some knowledge of the S-N characteristics is required. In the initial stages, assumed curves based on tests of similar structure are widely used. For all potentially critical joints, however, laboratory specimen tests are conducted to establish accurate S-N data. Typically, in the Boeing tests, three specimens were tested at each of three stress levels (a total of nine) at each joint. As an example, for the B-47 WS 354 splice, nine specimens were tested to initial crack. Once cracked, the panels were reworked to the fleet repair and retested to initial crack. In this way, two sets of data were obtained from one set of basic specimens.

Most of the laboratory tests were made in a 500,000 pound fatigue machine. Because of the sweep angle, the wing splices at the side of the body were tested in spedially designed jigs. Alternating loads as high as 670,000 pounds were applied in the swept jig. The static capacity of the setup is 2,000,000 pounds.

Having established the S-N data, service damage calculations were made for each mission phase at each critical location. It is interesting to compare some of the results and note the causes of damage at various locations (Figure 3). Of particular significance is the relative effect of the GAG cycle on the B-52 wing as compared to the B-47 wing. This difference is primarily due to the fact that fuel is carried in the B-52 wing and not in the B-47 wing.

TEST SPECTRUM COMPOSITION

Resolution of the service load spectrum into an "equivalent" test spectrum can be accomplished in several ways. While the linear damage theory signifies that spectra are equivalent if their cycle-ratio summations are the same, test experience indicates that the results may differ widely and that reasonable effort should be expended to establish a test spectrum which simulates the service spectrum. Obviously, as the test spectrum approaches the service spectrum the results become identical and no theory whatsoever is required for correlation. As mentioned previously, knowledge of the "exact" service spectrum is severely limited and, when coupled with environmental, time, and economic factors, it is not practicable to attempt exact spectrum simulation.

LOCATION	PERCENT OF TOTAL DAMAGE		
	GAG	GUST	MANEUVER
B-47			
BL 45 LOWER	11	44	45
WS 354 LOWER	12	72	16
B-52G			
BL 55 UPPER	100	0	0
BL 55 LOWER	80	13	7
WS 1025 LOWER	35	48	17

FIGURE 3

Damage Analysis Results

In each of the three Boeing programs, the test spectrum is distinctly different. However, certain fundamental criteria were observed. First, a minimum of three alternating stress levels was used and the levels were selected to correspond with those at which the bulk of service damage occurs. Secondly, the "spectrum" was made sufficiently short so as to allow several applications (a minimum of twenty) before failure. Finally, periodic applications of high load (90% of limit) were made to represent infrequent severe load occurrences.

Two basic types of spectra have been used at Boeing. "Block" loading, in which the cumulative damage from several missions is reproduced by separate blocks of constant alternating stress cycles, was used on the B-47 (Figure 4). The order of application of the blocks was varied continuously. In this case the test spectrum can no longer be identified with the individual flight. This approach is highly desirable from the standpoint of time and economy and was considered satisfactory in this case because the majority of damage was done by random gust cycles.

The B-52, however, as shown previously, is highly sensitive to the ground-air-ground effect which is a characteristic of each flight. Hence, "flight" loading, in which damage for each mission is reproduced individually with the ground-air-ground effect as an integral cycle, is used in the B-52 tests. The B-52F spectrum reproduces the elements of the idealized load history, although the great majority of calculated damage is done in the GAG and gust cycles. (Figure 5). The B-52G spectrum utilizes two levels of alternating stress to simulate all flight cycles (Figure 5).

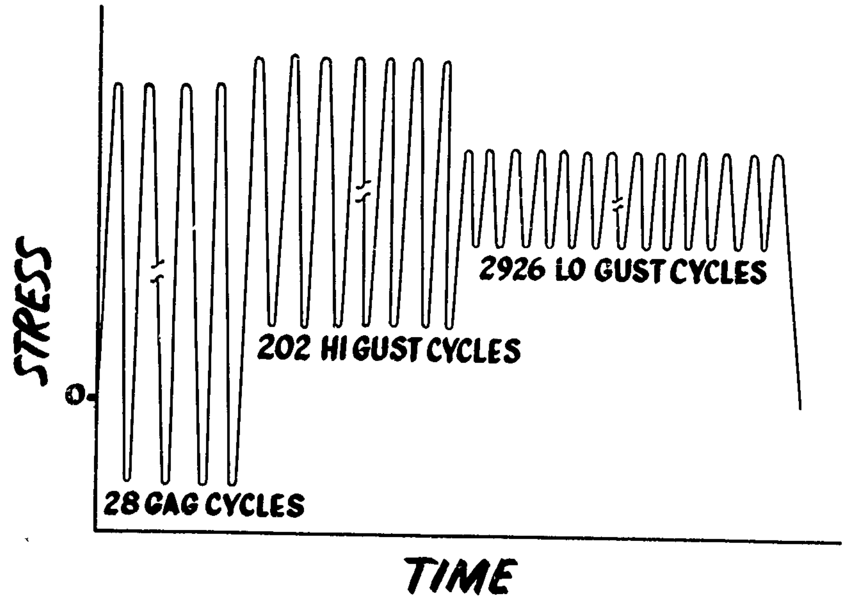
INSTRUMENTATION

Extensive strain gage instrumentation is used on the three Boeing tests. Axial and rosette gages in both single and back-to-back applications are used. While some gages are located close to discontinuities to measure stress concentration effects, most gages were located on "basic" structure to read gross stress values.

The principal purpose of the gages is to obtain stress distribution data for correlation with analysis and laboratory tests. The readings are also useful, as a secondary check on applied loads and, when displayed by oscillograph, they are used as a check against dynamic overstress and secondary cycles.

TEST SET UP

While each of the three test setups has distinctive features, basic principles are similar and only one, the B-52G, will be covered here. In this case the test specimen consists of a structurally complete wing and the fuselage forward of the aft gear bulkhead (Figure 6).



SPECTRUM	SEQUENCE
1	G, H, L
2	H, L, G
3	L, G, H
4	G, L, H
5	L, H, G
6	H, G, L

— ETC. —

FIGURE 4

B-47 Block Spectrum

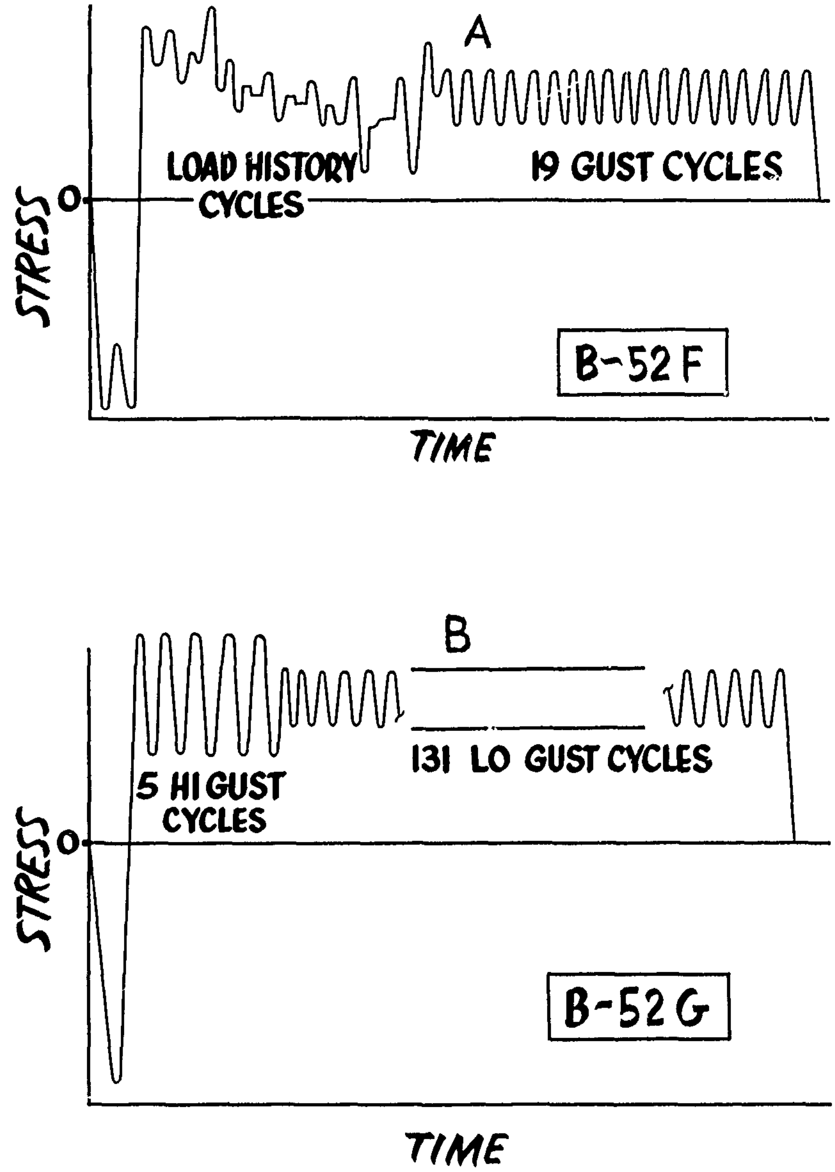


FIGURE 5

B-52 Flight Spectra

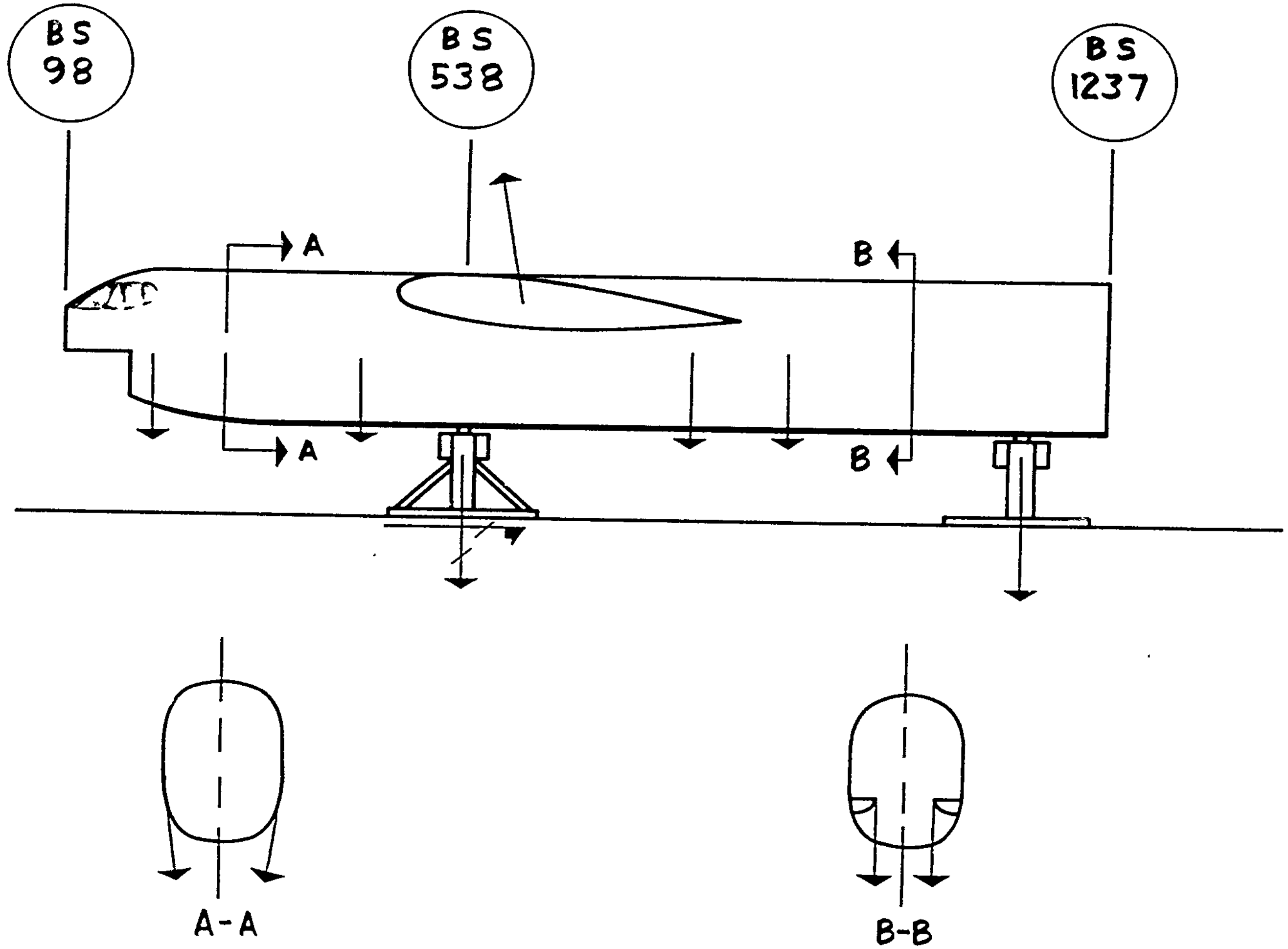


FIGURE 6
B-52G Cyclic Test Set-up

The specimen is jig-mounted on the main gear structure in such a manner that vertical, drag, and side loads are reacted at the forward gear with only vertical and side loads reacted at the rear gear. Down loads only are applied to the fuselage through four evener systems attaching to straps mounted on the fuselage skin and fuel tank deck structure. The up loads are applied to the wing through twelve evener systems attached to load blocks mounted on the front and rear spars (Figure 7). Fourteen evener systems are used to apply down loads. Down loads only are applied to the nacelles through direct ram systems.

Hydraulic rams are used to apply all loads. Double-ended rams operating through cable and sheave systems apply both up and down loads to the wing. The fuselage, like the nacelles, is loaded directly by single acting rams.

A closed loop servo system is used to control the loads. Ten rotating drum programmers provide the signal to operate the twenty-two rams. This system is fully automatic for cyclic loads and assures coordinated loading of all points to within 1% accuracy. Fail safe circuitry is provided to "freeze" the system instantly if a malfunction or structural failure occurs.

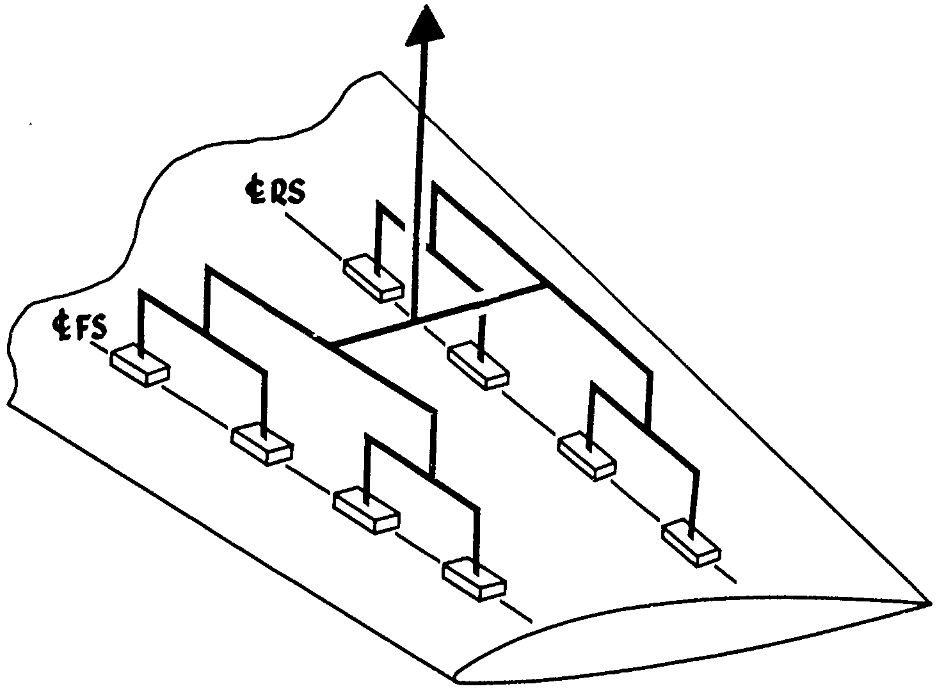
The principal difference between the B-52F test set-up and the B-52G test set-up just described is that flap and engine thrust loads are separately applied to those components in the B-52F test.

Loading rates are generally established by trial as they are dependent upon electrical-hydraulic capability, and must preclude binding or chatter. Average rates are given in Figure 8. (excluding inspection and set-up time.) It may readily be seen that the total cycles per simulated flight hour have considerable effect on the ratio of flight hours to test hours.

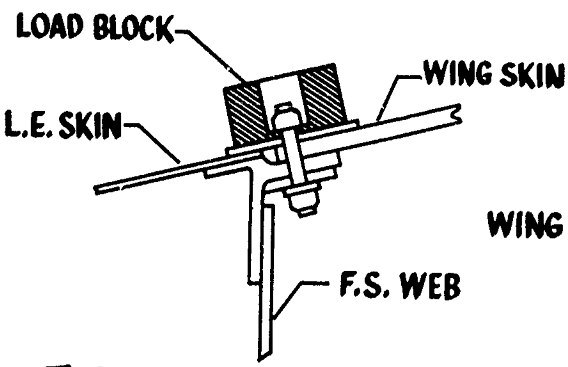
INSPECTION PROCEDURE

The value of complete vehicle testing is greatly enhanced if fatigue damage is detected early. In addition to providing a better basis for correlation with laboratory tests, early crack detection affords an opportunity to observe propagation and make repairs. For these reasons, inspection periods take as much or more time than does cycling in the Boeing tests.

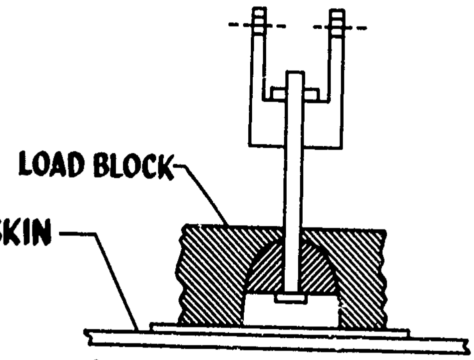
Several methods of detection have been used by Boeing with varying degrees of success. It should be noted that on the B-47 and B-52 the structural materials used at the previously described locations are 7075 and 7178 alloy in skin thicknesses of 1/10 to 5/8 inch, and extruded longerons of large cross sectional area. The crack propagation rates of this type and thickness of material are very high and inspection procedures used here would not necessarily apply in cases where thin gage structures are used.



TYPICAL WING LOADING SYSTEM



TYPICAL LOAD BLOCK ATTACHMENT



TYPICAL PULL-OFF DETAIL

FIGURE 7

B-52G Wing Evener System Details

AIRPLANE	CYCLES/TEST HR.	CYCLES/FLIGHT HR.	FLIGHT HRS./TEST HR.
B-47	160	19	8.5
B-52F	60	2.7	22
B-52G	90	9.0	10

FIGURE 8

Average Test Rates

Considerable experience with crack detection techniques was gained on the B-47. Approximately 1100 bolt holes were inspected on each airplane of the fleet. It was found that visual inspection with a lighted glass of four to ten power magnification was most reliable for inspecting a large number of bolt holes. The areas to be inspected were cleansed with a mild solution, applied with a cloth or bristle brush, prior to inspection.

During the B-47 cyclic tests, 244 bolt holes were given lighted magnified visual inspection following each second spectrum. Close visual inspections were also made of large areas of the external structure. The personnel making these inspections had previous experience with the component test inspections. Although many fatigue indications were detected in the lighter gage secondary structural areas, unexpected failures occurred on all three airplanes in heavy gage wing structure. One failure initiated in a bolt hole which was regularly inspected 13 times without damage being detected. It was demonstrated repeatedly that a critical location which was undetected prior to failure on one specimen, once known, could be found in the initial crack stage on other specimens.

In addition to strictly visual detection techniques, florescent and colored dye penetrant methods, and eddy current apparatus were sometimes used for verification. Most inspectors have definite preferences and skills according to the method used, so the choice of method was left to the discretion of the responsible personnel as much as possible.

Crack detection wires were used around suspected critical locations as a precautionary measure during the test. While they prevented at least one major structural failure, they were usually not in the right locations to detect cracks. Wide use of these systems makes visual inspection more difficult because the wire and cement obscure the area to be inspected. Also, the systems are easily damaged in gaining position for inspection, further limiting their use.

Consideration of all these factors must be made in deciding detection techniques. For the B-52F program, essentially the same procedures are used as on the B-47. For the B-52G, however, a different philosophy is applied. Because all fasteners in the major structural splices are of swaged collar type, and because the critical areas of these splices are not well known, no fasteners are pulled unless other indications warrant their removal. Instead, crack detection wires are widely used supported by close visual surface inspection.

RESULTS

The B-52 tests are still in operation and results cannot be discussed at this time. All test phases of the B-47 program are completed, however, and will be presented within the limits which security permits.

An excellent example of the importance of complete vehicle testing in disclosing unsuspected critical areas was encountered in the B-47 test at Boeing-Wichita. At the start of the 14th spectrum of block loading, while applying the 90% limit load condition, a failure of the fuselage occurred at the rear spar bulkhead, BS 515. The failure was the result of fatigue in the upper longerons at a flange cut-off. (Figure 9).

Of interest is the fact that the bolt holes adjacent to a flange through which the failure occurred had just been inspected with a boroscope at the end of the 13th spectrum and disclosed no damage.

As a result of this failure, crack detection wires were placed on the Douglas-Tulsa test airplane and a similar failure was prevented when crack damage was indicated by wire failure. Inspections of in-service aircraft disclosed no damage in the flange cutoff. However, cracks were found in the adjacent bolt holes on a few high-time airplanes. Repairs to this area were designed and incorporated on the test airplanes and on high-time fleet aircraft. Subsequent testing proved this repair satisfactory and all in-service airplanes were scheduled for rework. It is highly probable in this example that the loss of service aircraft was prevented by a complete vehicle test.

With the exception of the fuselage failure, the B-47 wing-body test went essentially as expected. Minor cracking occurred in light gage structure such as body skin, wing leading edges, and wing ribs. Good correlation was obtained between the three airplanes and failures on one often aided in detection of similar damage on another.

The first significant damage to primary wing structure occurred in the BL 45 splice on all three airplanes. Confirmed cracks were first detected at 23, 26, and 35 spectra for the Boeing, NASA, and Douglas airplanes respectively. Two major repairs were made to the Boeing wing after it sustained 8 inch and 28 inch cracks in the heavy skin (Figure 10). Testing was discontinued on the Boeing airplane after 37 spectra, at which time adequate life had been demonstrated on the wing and fuselage. Both the NASA and Douglas airplanes sustained 21 inch cracks in the wing skin (Figure 10) after 23 and 52 spectra respectively; and having reached previously established goals, the testing was discontinued.

At the completion of testing, all three airplanes were given thorough "tear-down" inspections in which the structure was dismantled and closely examined. The purpose of this inspection was to uncover any damage which might have been overlooked during test inspections and to classify the cause of damage found. These inspections were highly successful in disclosing cracks undetected during the test program. In addition to several cracks in secondary structure, cracks were also found in the WS 642 and WS 354 lower surface splices and in the upper longeron at BS 515.

It is significant to note that the cracks found at WS 354 were predominantly in unloaded splice plate holes at the splice centerline. Failures in the laboratory specimens of this area were predominantly in the skin and stiffeners.

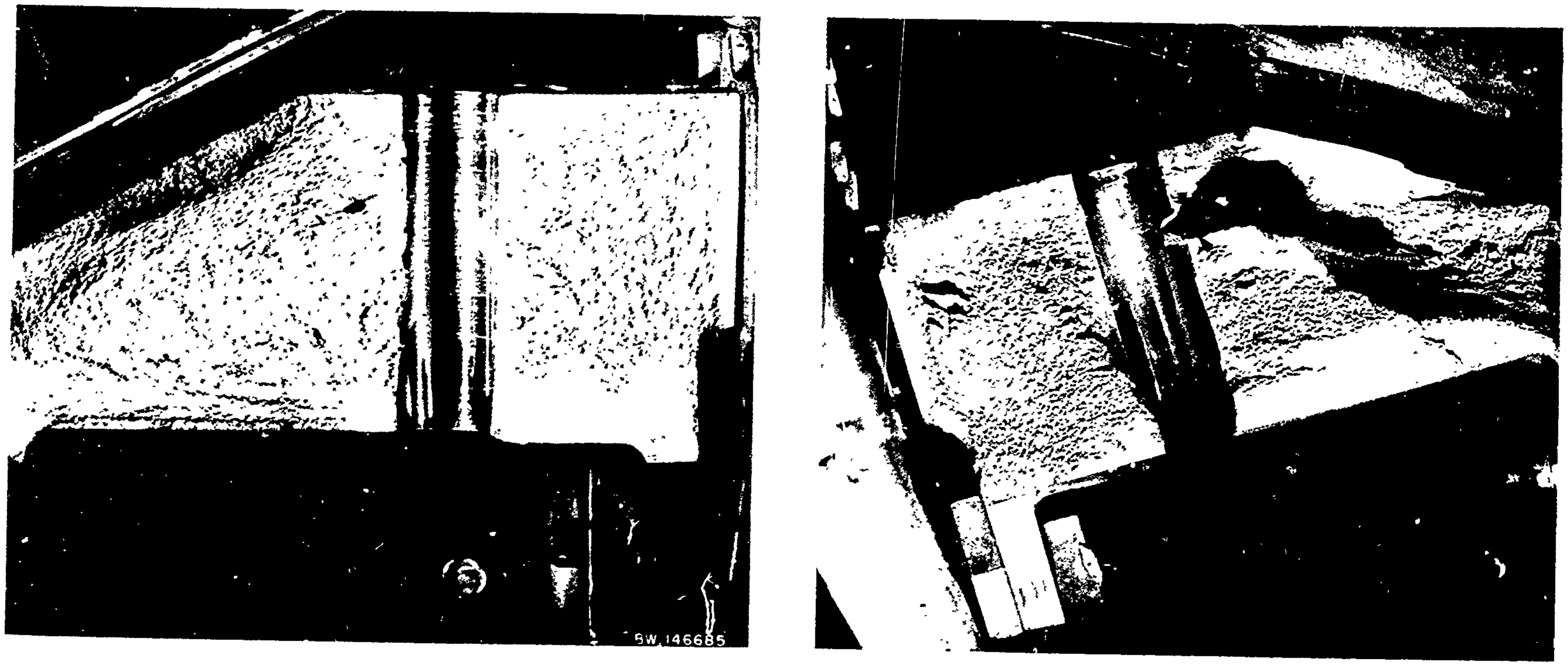


FIGURE 9
Longerons at Flange Cut-off

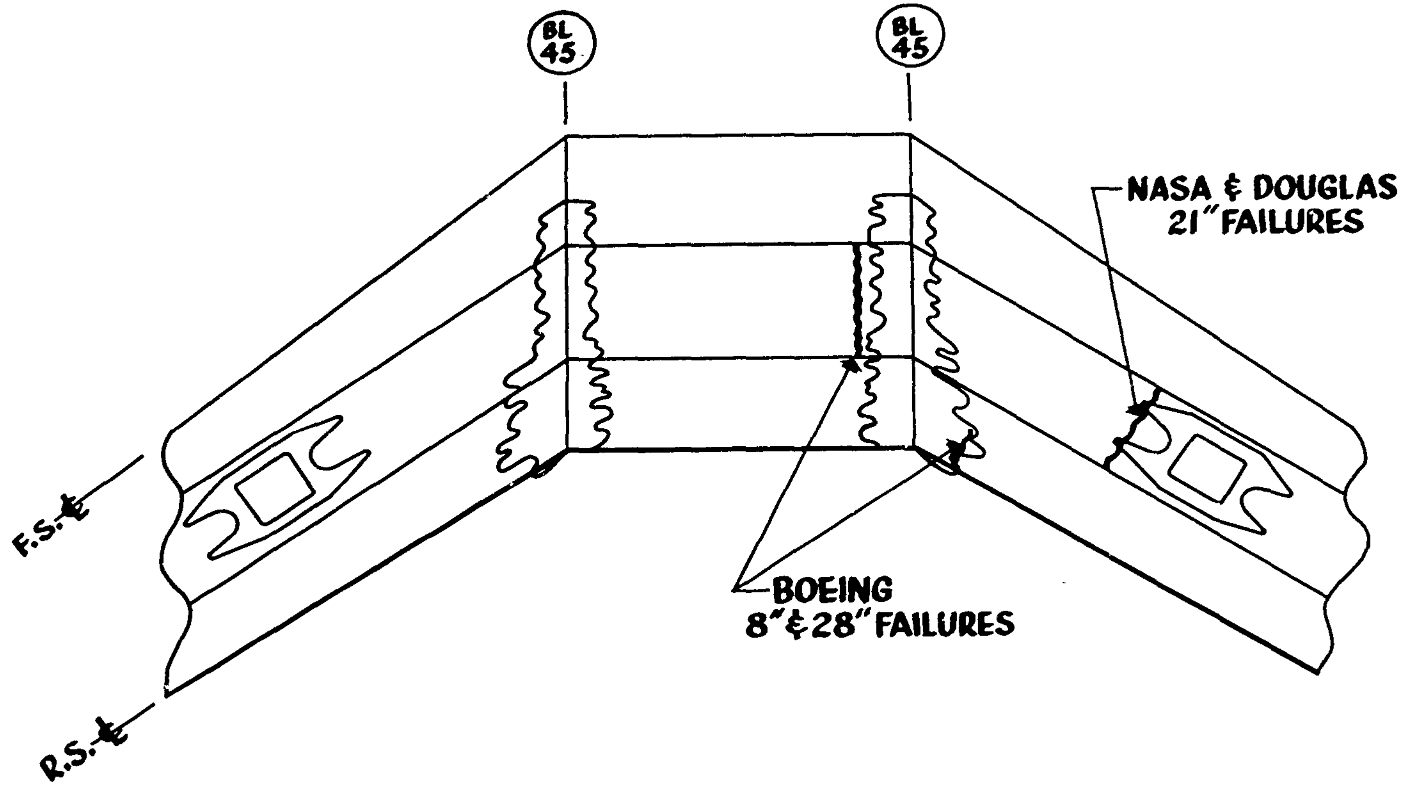


FIGURE 10

B-47 Major Test Failures

INTERPRETATION OF RESULTS

Interpreting the results of the B-47 tests, two answers were needed: First, an indication of the life expectancy of the airplane, and secondly, a realistic basis for fleet inspection and repair.

With regard to life expectancy, the log average "test life" was 1.6 times the "theoretical life" by the linear damage theory (i.e., $n/N = 1.6$). It is not surprising that the ratio is greater than unity as such is frequently the case (Figure 11).

It is interesting to note the relative insignificance of the S-N curve in determining the test life as compared to the theoretical life. Defining these terms as follows:

$$\text{Test life} = (D_T/D_S)(C)$$

$$\text{Theoretical life} = 1/D_S$$

Where,

$$D_S = \text{damage per service hour} = \sum\left(\frac{n}{N}\right)_S$$

$$D_T = \text{damage per test spectrum} = \sum\left(\frac{n}{N}\right)_T$$

$$C = \text{number of spectra to first crack}$$

It is seen that the service and test damage rates D_S and D_T both contain the S-N characteristic in the denominator and the effect tends to cancel in the calculation of test life. S-N curves which vary by a constant factor on cycles will obviously yield the same test life and S-N curves of widely varying shapes will yield nearly the same test life. The theoretical life, however, will vary directly as the S-N curve.

As an illustration, the B-47 service and test spectra are shown in Figure 12, neglecting mean stress variations for simplicity. Also shown are five S-N curves of arbitrary shape and cycle magnitude. Using curve 1 for reference, the relative test and theoretical life were calculated for each of the five curves and the results shown in Figure 13.

As shown, the test life varies by a small factor, whereas, the theoretical life, and consequently the ratio of test to theoretical life, varies by several factors. This illustration is reassurance for, and justification of, the use of an assumed S-N curve in developing the test spectra.

It is concluded that the test life is a less sensitive, and therefore a more reliable prediction of true life expectancy than is theoretical life. It is also apparent that cycle ratio summations other than unity, as typified in Figure 11, may be partially due to small errors in the S-N curve and should be the rule rather than the exception.

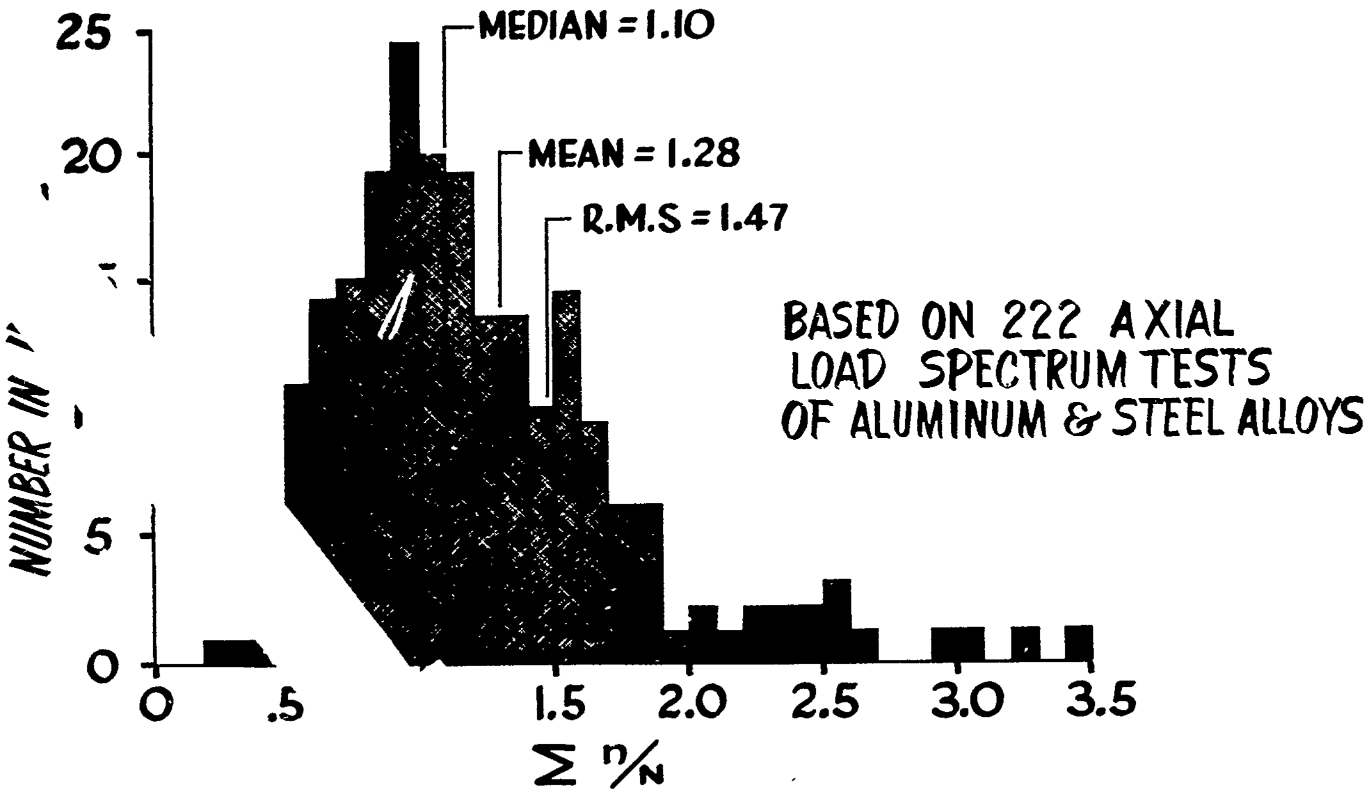
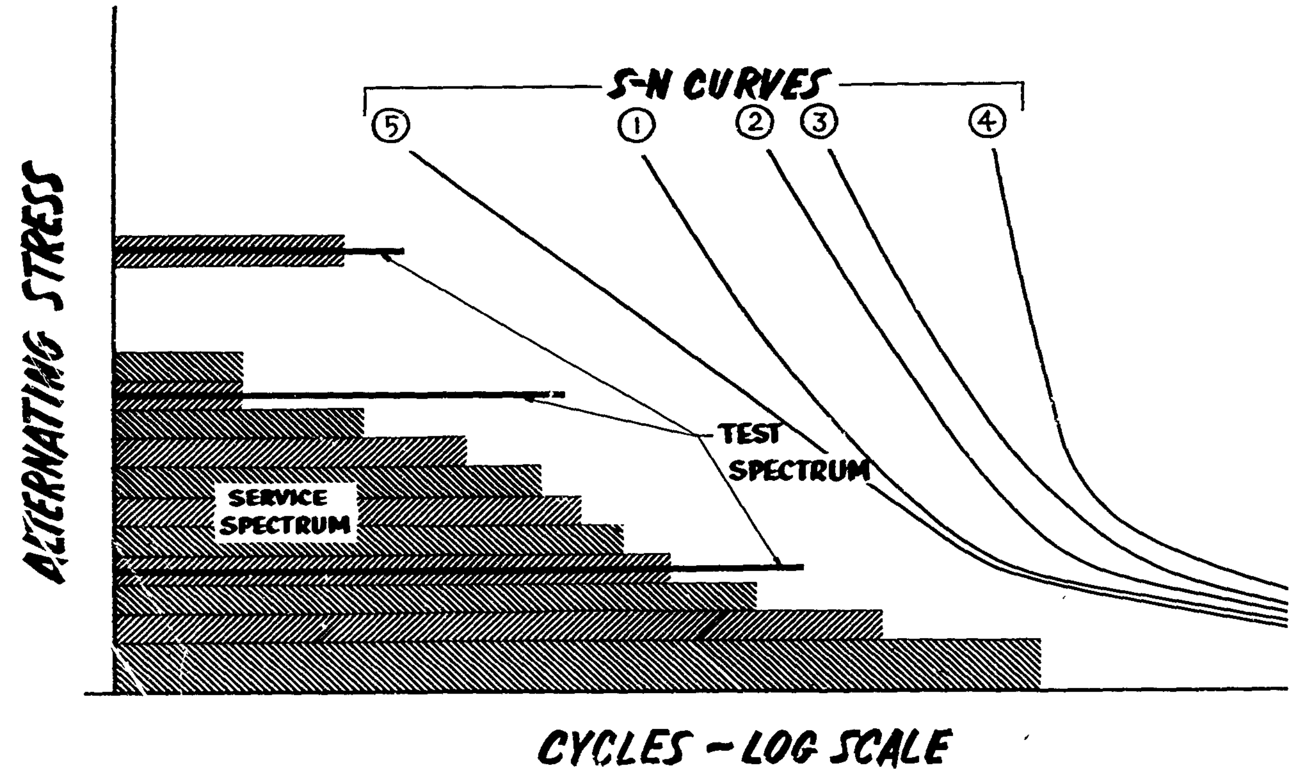


FIGURE 11

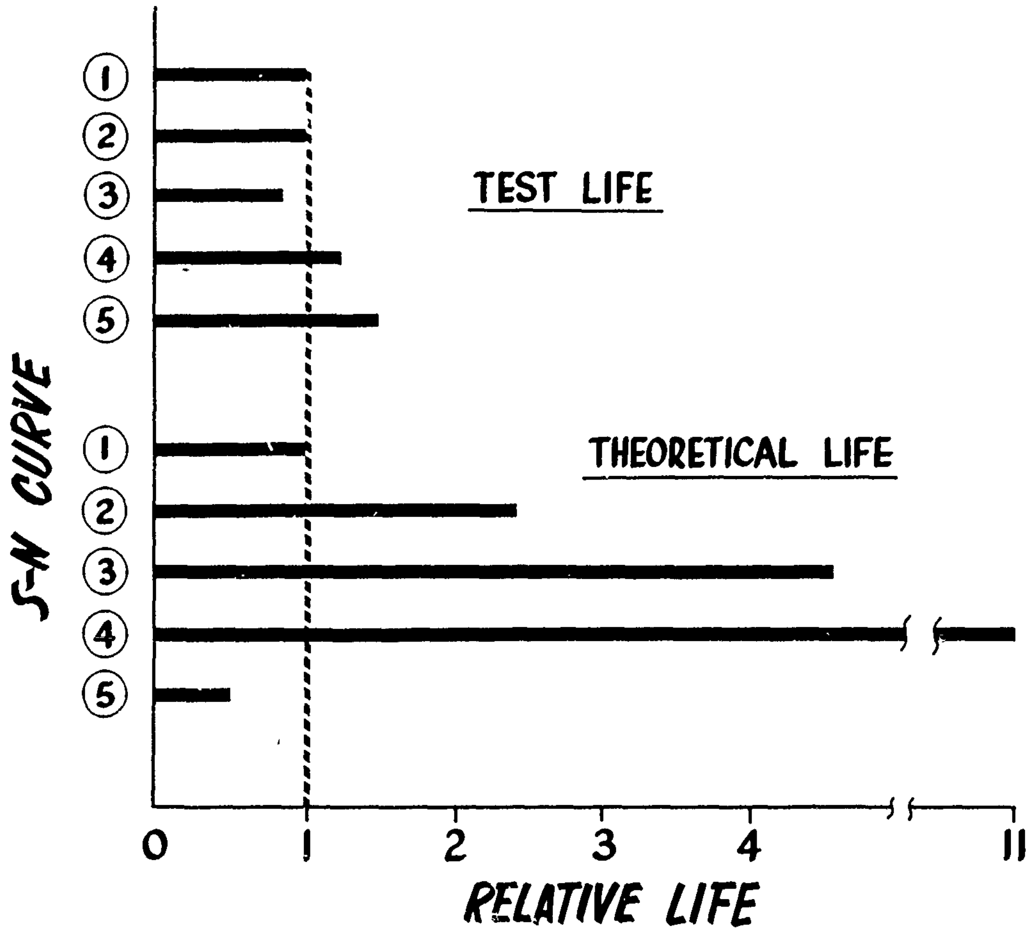
Correlation of Test & Theoretical Life



D SERVICE/HR $= \Sigma\left(\frac{n}{N}\right)_s$/HR D TEST/SPECTRUM $= \Sigma\left(\frac{n}{N}\right)_T$/SPECTRUM

FIGURE 12

Simplified B-47 Test & Servicc Spectra



$$\text{TEST LIFE} = \left[\frac{D_T/\text{SPECTRUM}}{D_s/\text{HR}}\right] (\text{NO. OF TEST SPECTRA})$$

$$\text{THEORETICAL LIFE} = \frac{1}{D_s / \text{HR}}$$

FIGURE 13

Effect of S-N on Test & Theoretical Life

The results of the test and tear-down inspections are invaluable in formulating handbook inspections for in-service aircraft. The real controversy lies in when to inspect. Figure 14, shows a statistical approach utilizing the results of the three airplane tests in the total fleet population. This figure shows the crack probability as a function of test spectra or equivalent airplane hours for 95% confidence on the mean. As shown in the figure, to obtain a probability not to exceed 5% that a crack exists in the fleet, it is necessary to apply a factor of 3.6 to the sample mean. This factor is increased by consideration of environmental effects not simulated in the test. Boeing is still working in this area in hopes of reaching a solution which will be acceptable to the statistical, the economical, and the operational demands. In addition to regular periodic inspections of all aircraft for the more obvious signs of damage, thorough bolt pulling inspections are performed on a sampling basis, and high-time airplanes are given destructive "tear-down" inspections on a less frequent basis.

CONCLUSIONS

Boeing is utilizing complete airplane fatigue tests as a major step in life evaluation of the B-47 and B-52 models. The results of these tests provide a most realistic basis for critical judgement of the theoretical life analysis and component specimen tests. Special attention is required for each model as to selection of critical areas and formulation of the test spectrum. Good inspection procedures and highly skilled personnel are most important in obtaining useful results. The tear-down inspections of high-time fleet aircraft are a valuable supplement to the test inspections. The analysis of scatter effects is a major obstacle to rational interpretation of life expectancy.

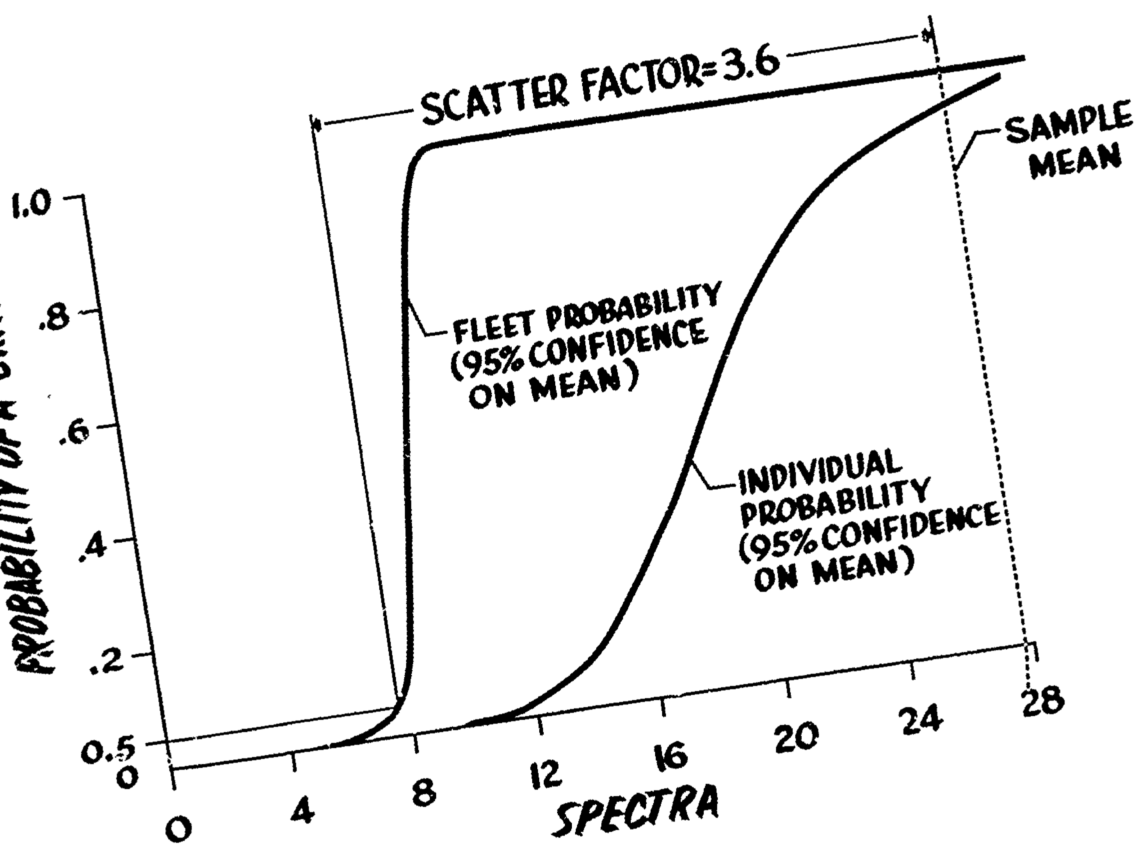


FIGURE 14
Statistical Interpretation of B-47 Test Results

EVALUATION OF FATIGUE SENSITIVE AREAS AND CUMULATIVE DAMAGE IN FULL-SCALE FATIGUE TESTING

By

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ABSTRACT

The data derived from the study of fatigue cracks which occurred during full-scale fatigue tests of C-46 transport airplane wings are summarized. The fatigue tests were of two types, constant-level and randomized-step. The randomized-step tests included two load-spectrum shapes. One load spectrum represented gust loadings on the airplane and the other spectrum represented a more severe loading history based on maneuver load statistics. Comparisons are made between the fatigue sensitive areas occurring in the constant-level tests and the randomized-step tests. The implications of these comparisons on the design of full-scale fatigue tests are discussed. Load-lifetime (S-N) curves are presented for initiation of the first crack, initiation of the critical crack, and final failure. The randomized-step tests are also examined from the standpoint of linear cumulative damage theory based on the constant-level test results.

SUMMARY

A study of the fatigue sensitive areas located during full-scale fatigue tests on C-46 transport airplane wings shows consistent differences in the indications of the constant-level tests as compared to randomized-step tests. No one level in the constant-level tests revealed as many fatigue sensitive areas as compared to the randomized-step tests, and no one level in the constant-level tests consistently gave a good indication of the area which proved critical in the randomized-step tests. The fatigue life to initiation of the first crack was in general agreement with the indications of linear cumulative damage, but final failure was delayed beyond what would be expected on the basis of linear cumulative damage. A qualitative explanation of this result is proposed.

INTRODUCTION

The current interest in full-scale fatigue tests reflects the need for two kinds of information, both bearing on the useful life of the airplane. Early identification of fatigue sensitive areas is important, both for the possibility of correction, and as a guide to the scop[illegible] inspection and repair programs. In addition, an assessment of se[illegible]e life is wanted but such assessment depends heavily on cumulative [illegible]age calculations, especially for the usual case, where the effects of different service missions on life must be evaluated on the basis of one full-scale test.

Various types of load schedules have been proposed for full-scale fatigue tests. They range in complexity from a simple constant-level test, through program loading and randomized steps, to the tests where service life is duplicated mission by mission and flight by flight. Despite these many choices, little information is available by which to judge the relevance of the data obtained by these various loading schedules, or indeed from full-scale testing itself. It is the purpose of the present paper to summarize the data derived from a study of the

fatigue cracks which occurred during the full-scale fatigue study by the NASA on C-46 transport airplane wings. In this study, both constant-level and randomized-step tests were employed (refs. 1 to 3). The loading schedules of the randomized-step tests were of two types, one based on gust statistics as applied to the C-46 airplane, the other a special research schedule based on maneuver statistics for fighters, as scaled to the design loads of the C-46. The data are considered from the standpoint of both crack location information and cumulative damage calculations. The paper represents an expansion of a study originally given as reference 4.

FATIGUE TEST EQUIPMENT

Airplane

The wing panels used for the present tests were obtained from surplus C-46 aircraft. The aircraft logs showed some flight time. Twelve wings had between 200 and 1,031 hours of flight, four (used in the randomized-step test) had 5,796 to 6,979 hours. All had been stored for several years in an open depot.

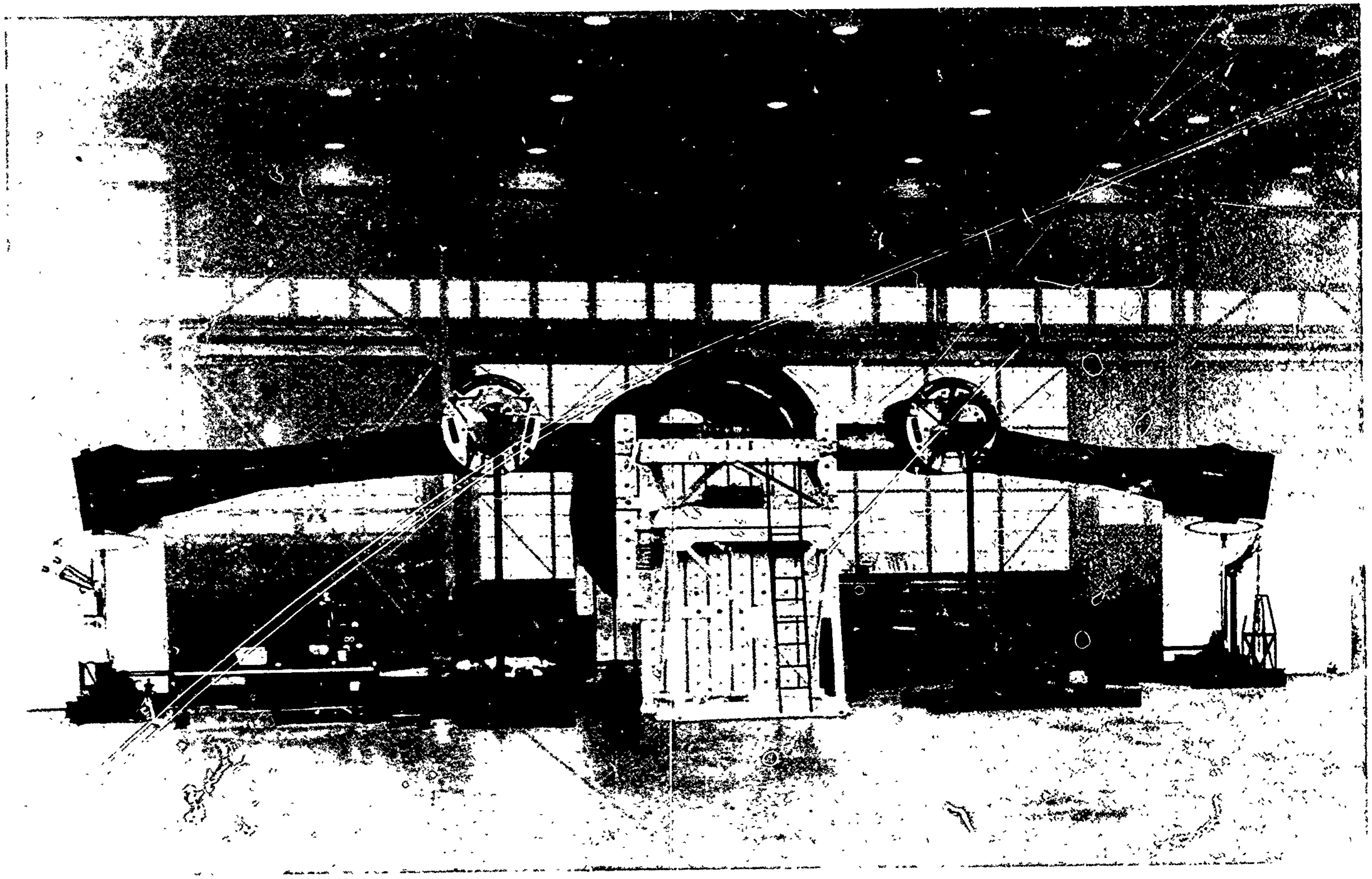
Some geometric characteristics of the wings and pertinent data for the C-46 airplane are given in the following table:

Probable operating gross weight, lb	41,000
Probable level-flight airspeed, fps	281
Wing area, sq ft	1,360
Wing span, ft	108
Mean aerodynamic chord, ft	13.688
Aspect ratio	8.58
Thickness at wing center section, percent chord	17
Slope of the lift curve, per radian	4.88

The airplane wings were of all-metal, riveted, stressed skin construction, made almost entirely of 2024 clad material. For test purposes, the fuselage was cut off normal to the axis, in front of and behind the wing. The wing tips and engines were removed and the fatigue specimen, consisting of a center section and two outboard panels, was inverted and mounted in either the constant-amplitude fatigue machine or the variable-amplitude fatigue machine shown in figure 1.

Instrumentation

In order to obtain stress-distribution information, the wings were instrumented with a number of wire resistance strain gages. Indicating



(a) Constant-amplitude fatigue machine.

Figure 1.- C-46 fatigue equipment.

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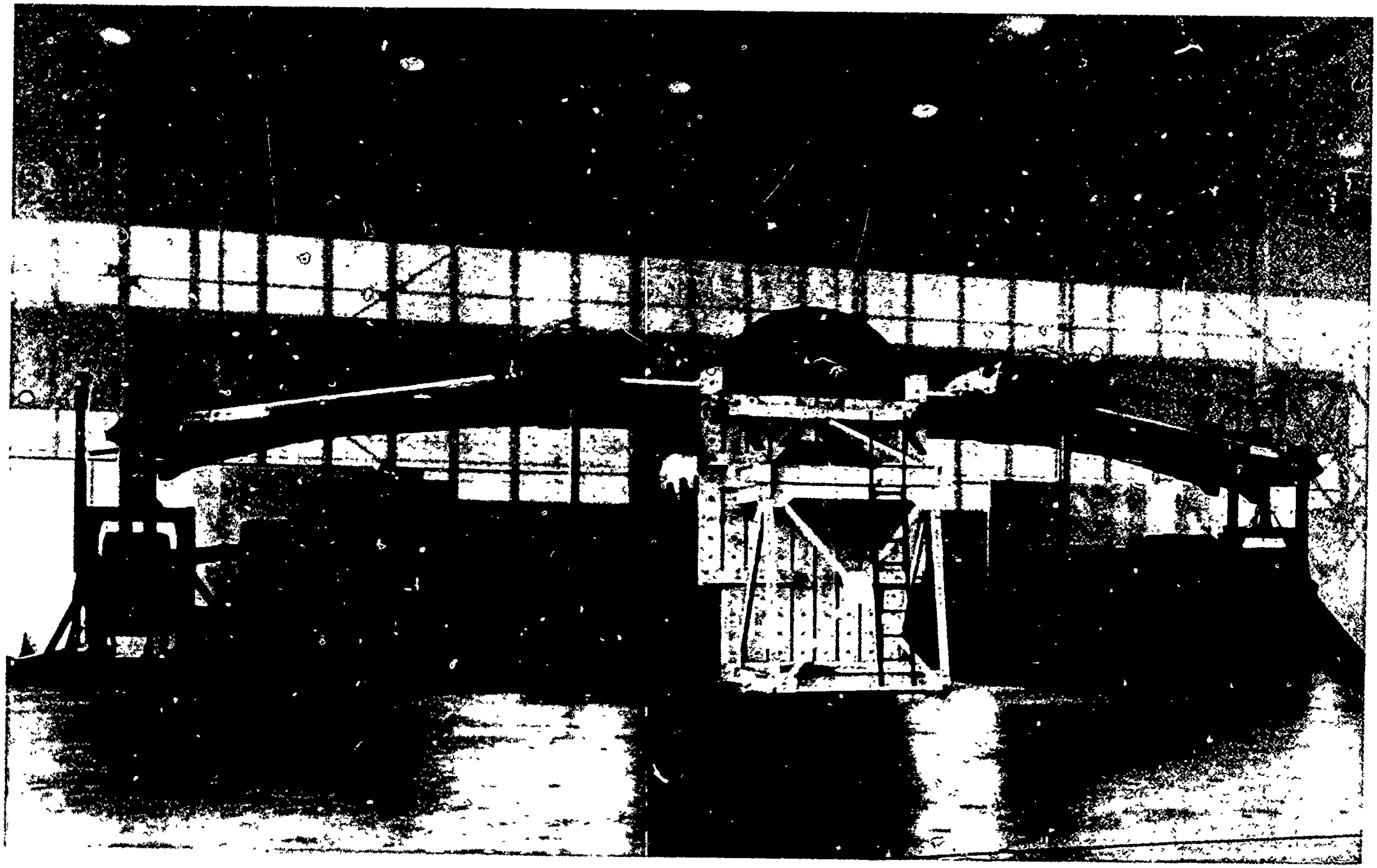

(b) Variable-amplitude fatigue machine.

Figure 1.- Concluded.

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equipment suitable for both static and dynamic measurements was employed. Locations and results of the strain-gage measurements are given in references 1 and 3.

To increase the probability of early detection of fatigue cracks and to reduce the time spent in visual inspection an electrical warning circuit was used. Copper wires of 0.002-inch diameter were cemented to the wing in the vicinity of stress raisers. When a crack occurred and passed under a wire, the wire broke and an alarm system was actuated. Twenty-three such crack detection circuits were used on each wing panel. Frequent and thorough visual inspections were made, supplemented by X-ray examination of hidden parts.

Fatigue Machines

Constant-level tests were accomplished by the resonant-frequency method, the randomized-step tests by a specially built machine of the forced-vibration type. Details of the machines and operations are described in references 1 and 3. In both machines concentrated masses were used to reproduce the 1 g stresses. Because of the method of support and loading, it is not possible in the fatigue machine to reproduce the flight bending-moment distribution exactly. Structural analysis indicated, however, that at span station 214, which is just outboard of the panel attachment joint, the bending-moment margins of safety were lowest. In addition, a distributed load static test in which a brittle strain-indicating lacquer was used indicated that the highest strains were to be found in the vicinity of this span station. The concentrated masses for attachment to the wing were, therefore, proportioned and located in such a way that the 1 g bending moment, shear, and torque at span station 214 for the level-flight, low-angle-of-attack condition were reproduced. A comparison of the 1g design bending moment and the bending moment applied by the concentrated masses is shown in figure 2.

LOADING SCHEDULE

The load levels used in the present study are shown in figure 3(a). Six different levels of load factor increment were selected for the constant-level tests, ranging from ±0.250 to ±1.725. For the randomized-step tests, two loading schedules were used, one based on gust statistics, the other on maneuver load statistics. The number of loads for both schedules at each level is shown by the length of the bar, the spectrum compositions are given in detail in tables 1 and 2, the probability distributions on which these load schedules were based are shown in figure 3(b).

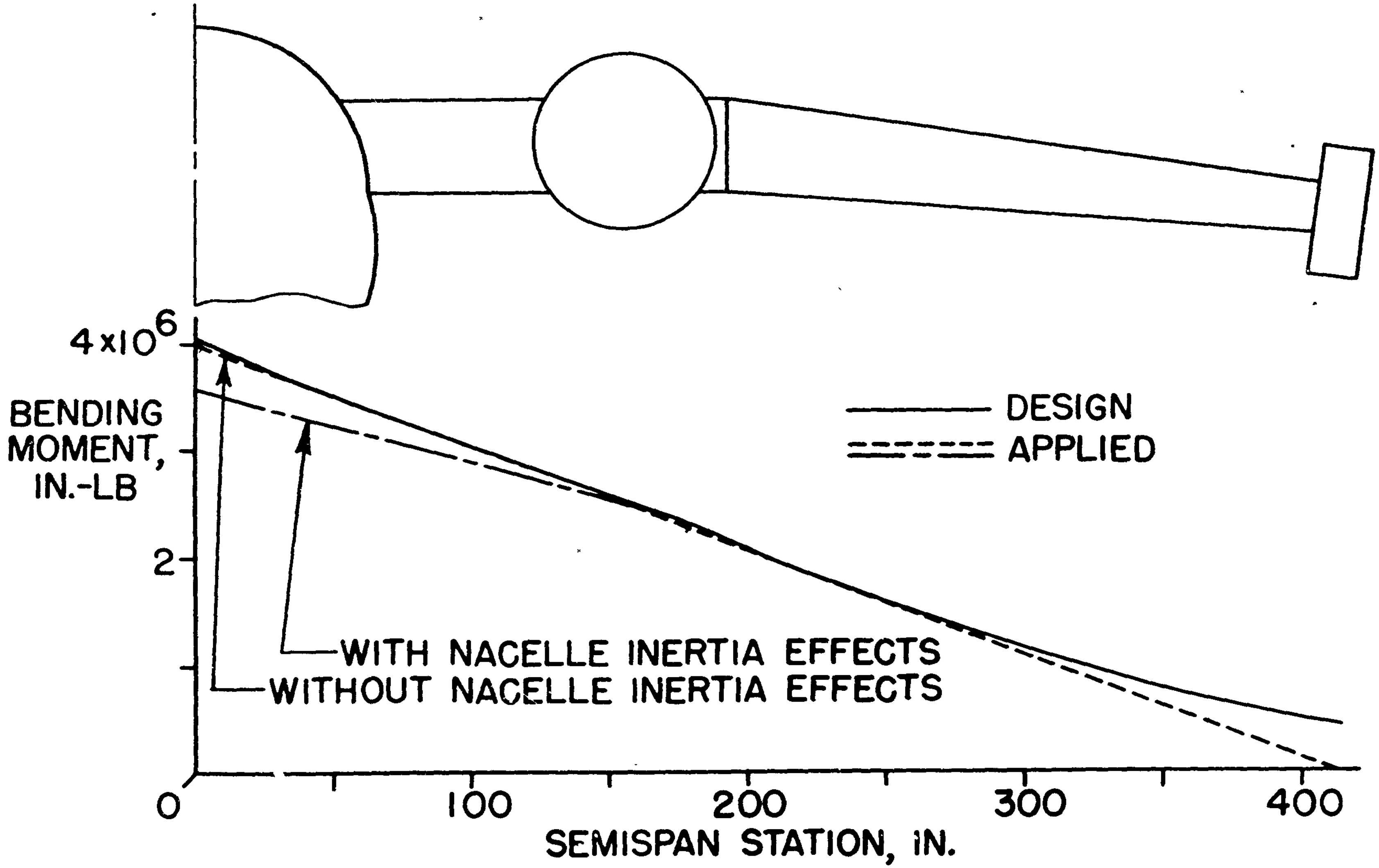

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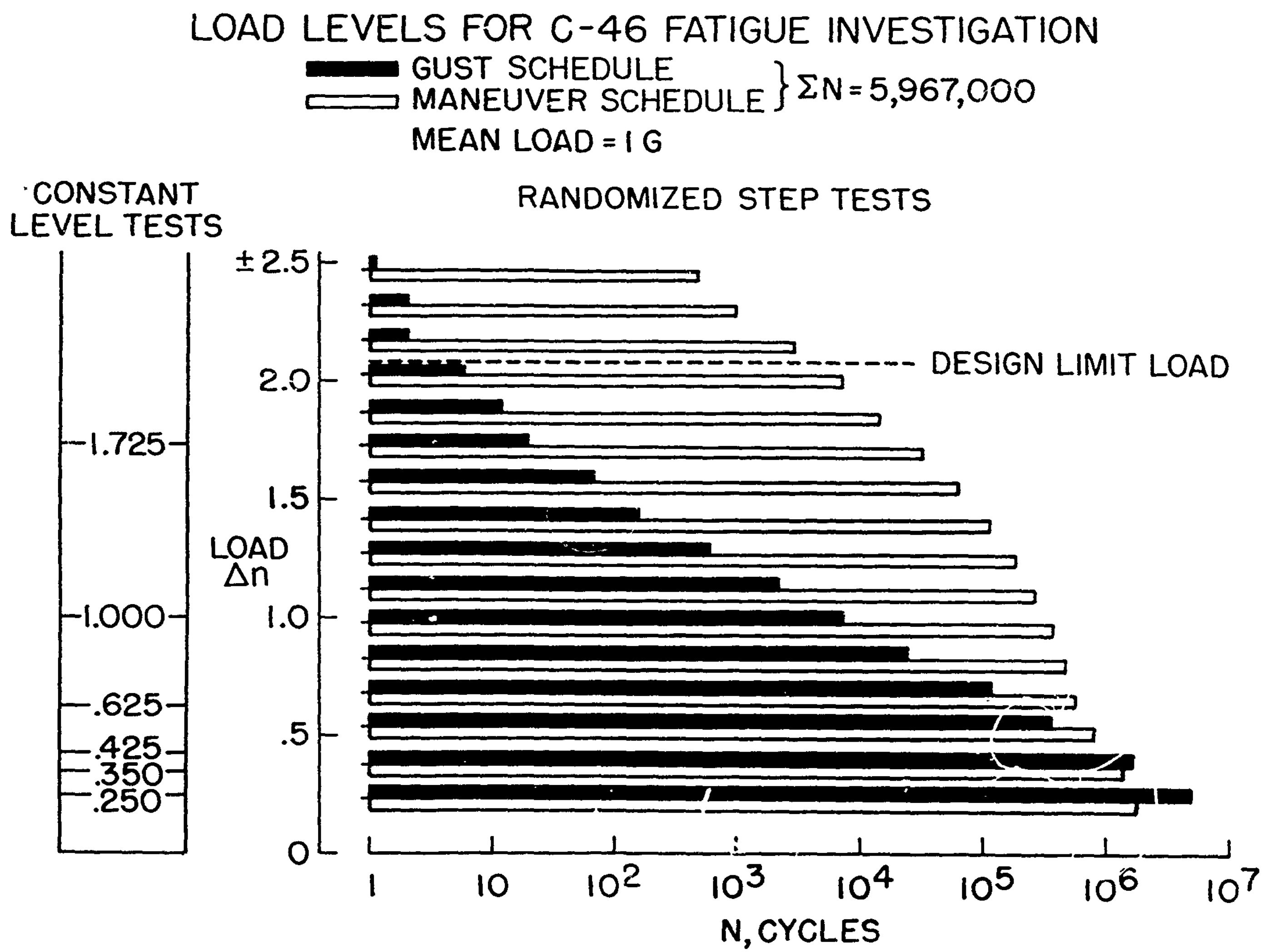
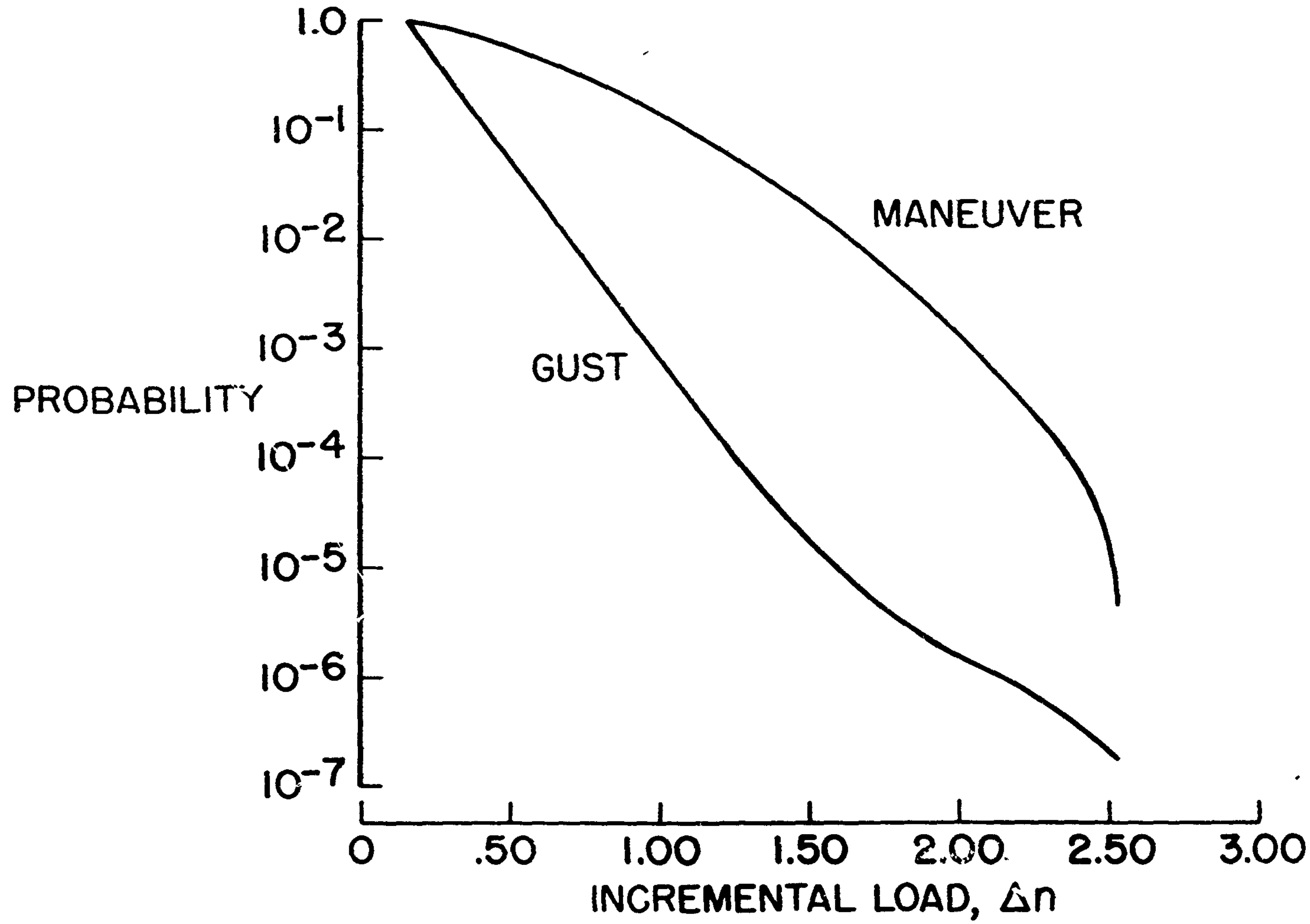
Figure 2.- Comparison of design and applied bending moments for the 1g level-flight condition.



(a) Composition of constant-level and randomized-step loadings.

Figure 3.- Load levels for C-46 fatigue investigation. NASA



(b) Probability distributions for design of gust and maneuver loadings.

Figure 3.- Concluded.

NASA

For the initial randomized-step tests every effort was made to develop a load history which would correspond as closely as possible to the flight operating experience of the airplane. Since calculations indicated that loads due to such effects as landing, taxiing, and maneuvers did not contribute significantly to the load history as compared to the contribution from rough air, the loading schedule was based on the most severe limits of the gust data of reference 5, and the average value given in reference 5 for the path ratio, that is, ratio of flight distance in turbulence to total flight distance. Values of the parameters for the calculation of the gust-load schedule, based on an assumed 50,000 hours of flight, were as follows:

Total flight distance, miles	10^7
Path ratio	0.1
Class interval, Δn	0.15
Lowest threshold value, Δn	0.15

These parameters in conjunction with the data of reference 5 indicate, as shown in table 1, that the threshold of $\Delta n = \pm 0.15$ would be exceeded 5,967,000 times in 10^7 miles, while, for example, 3,931,200 load factor peaks would lie between ±0.15 and ±0.30. In operation of the test, the loads in any class interval were applied at the mean of the interval. To prevent the application of all the cycles at any one load level consecutively, the total number of loads in each class was divided by 100 to produce 100 sequences or blocks. The order in which the load levels were applied in each block was established by reference to a table of random numbers. No two of these blocks were alike. Since loads at ±1.575 and above occur less than 100 times, they could not be represented in each block; these loads were distributed at random among the 100 blocks.

Another load history used in the randomized-step tests and shown in figure 3 is that labeled "maneuver loading." As compared to the gust-load schedule, the maneuver load schedule, table 2, contains a higher proportion of large loads and a smaller proportion of small loads. The shape of the distribution was based on some statistical studies of the peaks of positive normal load factor increment encountered in fighter aircraft during operational training. For test purposes these loads were scaled to the transport on the basis of the design limit load factors of the fighter and transport categories. This loading was devised primarily to explore the effects of spectrum shape on such factors as crack propagation and lifetime and is not to be considered as realistic for C-46 airplanes. Because of the high proportion of high loads and the higher damage rate expected, the maneuver load schedule, for operating purposes, was divided into 1,000 blocks.

TABLE 1.- GUST LOAD SPECTRUM

Threshold, Δn	Summation of cycles	Load level, Δn	Cycles in class interval
0.15	5,967,000		
		0.225	3,931,200
.30	2,035,800		
		.375	1,544,400
.45	491,400		
		.525	351,000
.60	140,400		
		.675	106,700
.75	33,696		
		.825	23,510
.90	10,179		
		.975	7,301
1.05	2,878		
		1.125	2,036
1.20	842		
		1.275	586
1.35	260		
		1.425	148
1.50	112		
		1.575	69
1.65	43		
		1.725	19
1.80	24		
		1.875	12
1.95	12		
		2.025	6
2.10	6		
		2.175	2
2.25	4		
		2.325	2
2.40	2		
		2.475	1
2.55	1		
		2.625	1
2.70	0		

First block		Second block	
Load level, Δn	Cycles in class interval	Load level, Δn	Cycles in class interval
0.525	3,510	1.725	0
2.425	0	2.175	0
1.875	0	.825	236
2.325	0	1.425	1
.675	1,067	1.275	5
2.025	0	1.575	1
.975	73	1.125	20
1.575	1	2.475	0
1.725	0	.375	15,444
1.275	6	.225	39,312
2.175	0	.675	1,067
1.125	20	.525	3,510
1.425	1	.975	73
.825	235	2.025	0
.225	39,312	2.325	0
.375	15,444	1.875	1

First block		Second block	
Load level, Δn	Total cycles	Load level, Δn	Total cycles
0.225	39,312	0.225	78,624
.375	15,444	.375	30,888
.525	3,510	.525	7,020
.675	1,067	.685	2,134
.825	235	.825	471
.975	73	.975	146
1.125	20	1.125	40
1.275	6	1.275	11
1.425	1	1.425	2
1.575	1	1.575	2
1.725	0	1.725	0
1.875	0	1.875	1
2.025	0	2.025	0
2.175	0	2.175	0
2.325	0	2.325	0
2.475	0	2.475	0
Summation of cycles	59,669	Summation of cycles	119,339

TABLE 2.- MANEUVER SPECTRUM

Threshold, Δn	Summation of cycles
0.15	5,967,000
.30	4,184,500
.45	2,852,500
.60	2,062,500
.75	1,495,500
.90	1,031,500
1.05	667,000
1.20	403,000
1.35	228,000
1.50	120,500
1.65	59,500
1.80	27,500
1.95	11,500
2.10	4,500
2.25	1,500
2.40	500
2.55	0

Load level, Δn	Cycles in class interval
0.225	1,782,500
.375	1,332,000
.525	790,000
.675	567,000
.825	464,000
.975	364,500
1.125	264,000
1.275	175,000
1.425	107,500
1.575	61,000
1.725	32,000
1.875	16,000
2.025	7,000
2.175	3,000
2.325	1,000
2.475	500

First block		Second block	
Load level, Δn	Cycles in class interval	Load level, Δn	Cycles in class interval
2.175	3	2.325	1
.375	1,332	1.275	175
1.275	175	.675	567
.675	567	1.425	108
1.425	107	.375	1,332
2.025	7	1.125	264
2.475	1	1.575	61
2.325	1	2.475	0
1.575	61	1.875	16
.525	790	.825	464
.225	1,783	2.025	7
.825	464	2.175	3
1.725	32	.973	365
.975	364	1.725	32
1.875	16	.225	1,782
1.125	264	.525	790

First block		Second block	
Load level, Δn	Total cycles	Load level, Δn	Total cycles
0.225	1,783	0.225	3,565
.375	1,332	.375	2,664
.525	790	.525	1,580
.675	567	.675	1,134
.825	464	.825	928
.975	364	.975	729
1.125	264	1.125	528
1.275	175	1.275	350
1.425	107	1.925	215
1.575	61	1.575	122
1.725	32	1.725	64
1.875	16	1.875	32
2.025	7	2.025	14
2.175	3	2.175	6
2.325	1	2.325	2
2.475	1	2.475	1
Summation of cycles	5,967	Summation of cycles	11,934

RESULTS AND DISCUSSION

Fatigue Sensitive Areas

Fatigue cracks ordinarily were discovered either by visual inspection (usually during cyclic loading) or by the crack detector system before they exceeded 1/4 inch in length. Most cracks occurred in stress raisers that would be present in similar form in almost any metal aircraft structure. The crack locations and a brief description of the nature of the crack are summarized in figure 4, which shows a plan view of the wing lower surface outboard of the wing attach angle at wing station 192. Also shown in figure 4 is a cross section of the lower tension surface in the region of critical strain, showing the construction.

Cracks originated in the skin at the corners of cutouts B, F, G, and H and in the internal doublers of cutouts C, D, and E. These cutouts permit access to the wing for final assembly and for various equipment items; all were provided with removable doors, B and F with simple inspection doors, C, D, E, G, and H had stressed doors. A crack also appeared in the external doubler, at the point where the doubler, attach angle, and 30-percent-chord spar meet, denoted as area III.

Total number of fatigue areas.- The total number of fatigue areas found in the constant-level tests and the randomized-step tests are shown in figure 5 together with the number of panels tested in each phase of the program. At each of the two lower levels in the constant-level tests, a crack developed in area B. At ±0.250 this was the only crack which developed, one in each panel tested. At ±0.350 cracks developed in area E as well as in B while at ±0.425 cracks developed in areas B, D, and H. In general, as the load level increased up to ±1.000, the number of areas in which cracks initiated at some stage during loading increased, until at ±1.000, six out of the eight areas were involved. Results from the two panels tested at ±1.725 show that cracks also initiated in six of the eight areas, a possible indication that the trend for more fatigue areas to develop as the load level increases, may not continue at levels beyond the range of the tests.

Ten panels were tested with the gust loading in the randomized-step tests, four with the maneuver loading. As compared with the constant-level results, most panels developed cracks in all, or nearly all, of the areas where cracks had appeared in one or more of the constant-level tests and there were no new areas. Every panel developed a crack in area B; cracks in areas F, III, and D were nearly always present; in C, E, and H, usually. Even allowing for differences in sample size, no one level in the constant-level tests showed such a wide distribution of fatigue sensitive areas. It is tempting to speculate that two or more of the

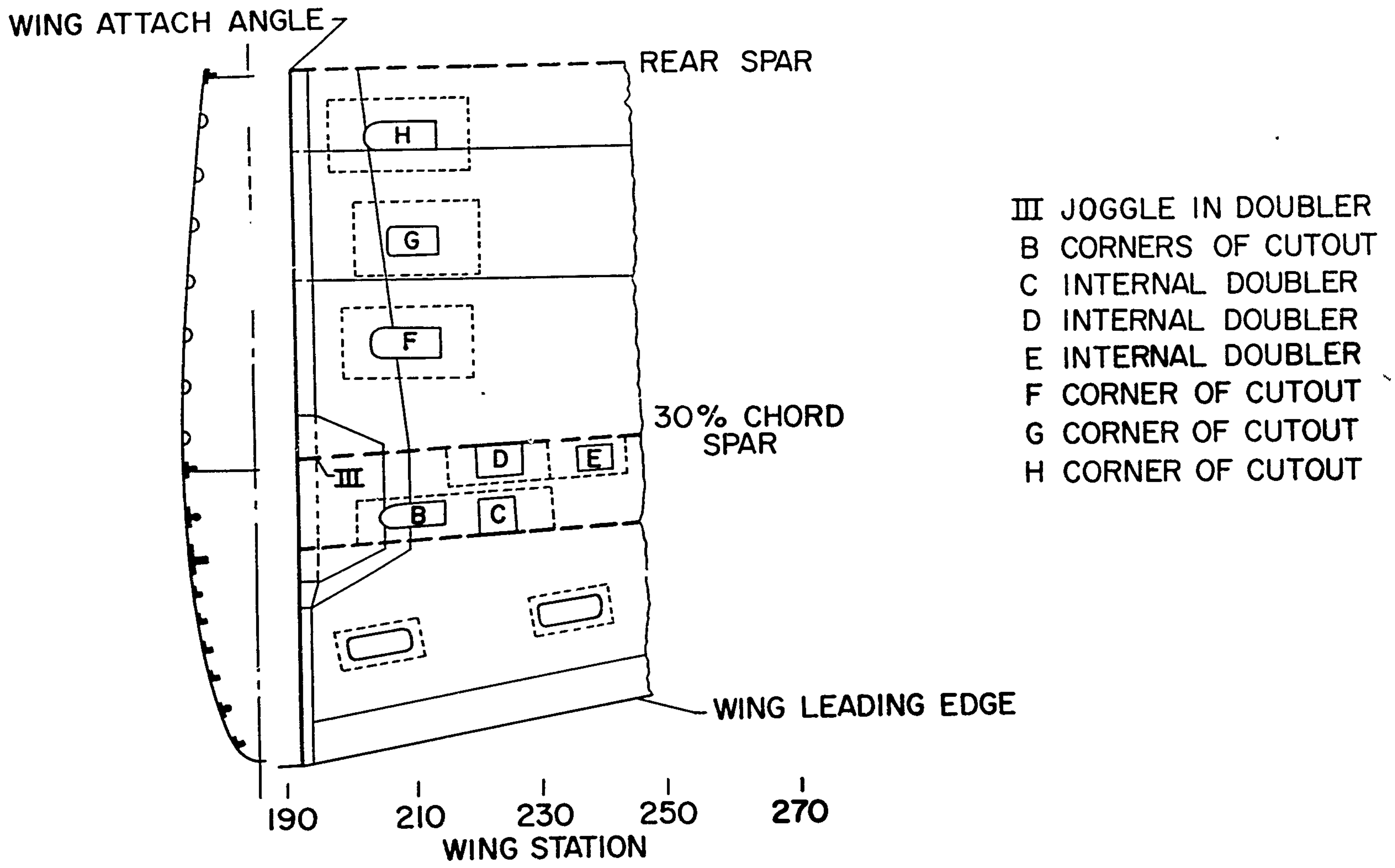


Figure 4.- Fatigue areas, C-46 tests.

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Loading		Number of panels tested	Number of panels with crack in area-							
			B	F	III	C	D	E	G	H
CONSTANT LEVEL	±1.725	2	2	1	0	2	2	2	0	2
	±1.000	4	2	3	1	0	4	2	0	3
	± .625	5	3	1	2	0	0	0	1	3
	± .425	3	3	0	0	0	2	0	0	1
	± .350	2	2	0	0	0	0	2	0	0
	± .250	2	2	0	0	0	0	0	0	0
TOTAL		18	14	5	3	2	8	6	1	9
RANDOMIZED STEP	GUST	10	10	9	9	7	9	9	0	7
	MAN'V'R	4	4	4	4	2	4	3	0	3
TOTAL		14	14	13	13	9	13	12	0	10

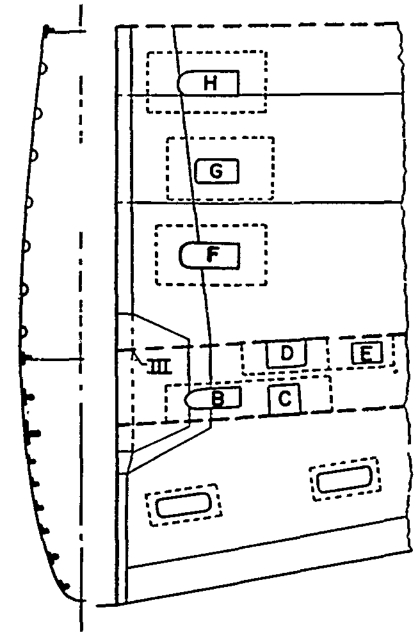

Figure 5.- Total number of fatigue areas.

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constant levels in combination might have given a reasonably close picture of crack areas shown by the randomized-step test but whether these results can be superimposed in this way could be a topic for other research.

First crack.- The appearance of the first fatigue crack in an airplane could be a valuable indication to the aircraft operator, perhaps a guide for the scheduling or revision of inspection programs. It is of some interest, therefore, to compare the locations of the first crack in each panel tested in the C-46 program. The data are shown in figure 6. At the two lowest values of the constant-level tests, the first crack appeared consistently in area B. As the load level increased, the location of the first crack moved aft to area F, and to D and E, and there is more scatter in the point of its appearance; at 0.625 where five panels were tested, four different locations are involved. At the highest level, there is a return to consistency in the location of the first crack.

In the randomized-step tests, area F and area D were most frequently the location of the first crack. There may be some systematic difference in the results for the gust and maneuver loadings in that, for the maneuver loading with its greater proportion of high loads the first cracks did not initiate in area B. Similarly the first crack did not start in area B in the two highest levels of the constant-level tests. But even allowing for differences in sample size, no one level in the constant-level tests gave a close representation of the locations of the first crack in either of the randomized-step loadings.

Except for two cases, one in the gust, the other in the maneuver loadings, the first cracks in the randomized-step tests were nuisance cracks which involved no threat to the structure. After initiation they grew slowly, if at all, and even without repair they seldom exceeded 1 inch in length at the completion of the test.

With the gust loading, on the average, first cracks appeared at a point in the program which corresponds to 13,000 hours of flight. From flight operations not many instances of fatigue damage in C-46 airplane wings are known but only a few of these aircraft have thus far exceeded this number of flight hours. The NASA has examined eight airplanes with 10,000 to 20,000 hours of flight. In four of these with 14,000 hours or less, no cracks were found. Four with 19,000 to 21,000 hours had one or more cracks each. The data are shown in figure 7 along with the first crack data from the gust loading in the randomized-step tests. The two sets of data are not exactly comparable, since one of the eight wing panels had more than one crack, and it was not possible to tell which had appeared first. All of the cracks were, however, in the section of the wing in the vicinity of wing station 214. All were small. Taking them as representative of early cracks in flight, out of eight locations

LOADING		NUMBER OF PANELS TESTED	NUMBER OF PANELS WITH CRACK IN AREA-							
			B	F	III	C	D	E	G	H
CONSTANT LEVEL	±1.725	2	0	0	0	0	0	2	0	0
	±1.000	4	0	1	0	0	1	2	0	0
	±.625	5	1	1	2	0	0	1	0	0
	±.425	3	1	0	0	0	2	0	0	0
	±.350	2	2	0	0	0	0	0	0	0
	±.250	2	2	0	0	0	0	0	0	0
TOTAL		18	6	2	2	0	3	5	0	0
RANDOMIZED STEP	GUST	10	2	3	0	0	2	2	0	1
	MAN'V'R	4	0	2	1	0	1	0	0	0
TOTAL		14	2	5	1	0	3	2	0	1

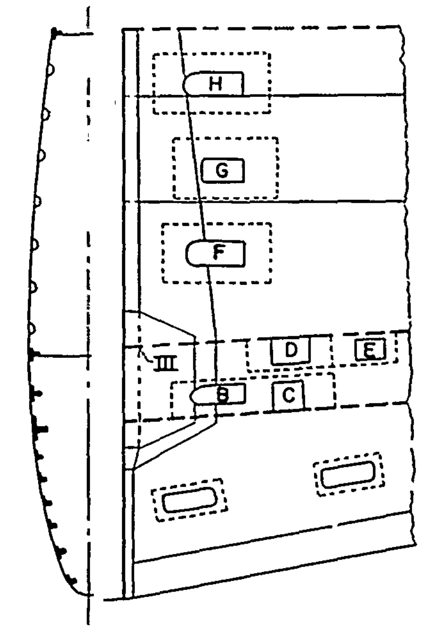

Figure 6.- Location of first crack.

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LOADING		NUMBER OF PANELS	NUMBER OF CRACKS FOUND IN AREA-							
			B	F	III	C	D	E	G	H
FLIGHT										
AIRPLANE 1		2	0	0	0	0	1	0	0	0
AIRPLANE 2		2	1	0	0	1	1	1	0	0
AIRPLANE 3		2	0	0	0	0	0	1	0	0
AIRPLANE 4		2	0	0	0	0	0	2	0	0
TOTAL		8	1	0	0	1	2	4	0	0
RANDOMIZED STEP	GUST	10	2	3	0	0	2	2	0	1

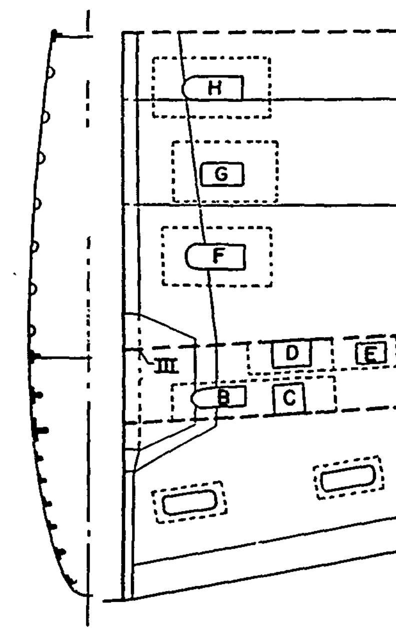

Figure 7.- Fatigue areas found in flight operations compared with first crack in randomized-step tests.

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there are five points of agreement in either presence or absence of cracks. A further point of agreement lies in the character of these early cracks in flight which, with one exception, were regarded by personnel who made the inspection as nuisance cracks. These were the same personnel who had conducted the laboratory study. The one exception is the crack in area B which was regarded as potentially serious. In this respect, it may be only coincidence but in these randomized-step tests, all first cracks were nuisance cracks except for one first crack in B which grew appreciably after its appearance and propagated to final failure very much later in the program. In view of the general agreement in crack character and location in these two sets of data, there is room for some confidence in the belief that the randomized-step gust loading was a reasonable equivalent to the random loading of the flight operation.

Critical crack.- Of the total of 132 fatigue cracks cataloged in the study of the 32 C-46 wing panels, one crack in each panel propagated to such an extent that structural integrity was threatened. The areas in which these critical cracks originated are shown in figure 8. In general, the C-46 tests were not continued to the point of structural failure, although this did occur on two occasions. Careful records of crack propagation were made, however, and testing continued to the point of rapid propagation and imminent complete failure, generally to the loss of 20 to 35 percent of the total tension area.

In the constant-level tests, as shown in figure 8, the critical crack initiated in area B with remarkable consistency. Aside from the two panels at the lowest level where the first crack was the only crack to develop, there were six other panels where the crack which became the critical crack was also the first one to develop and in four of these area B was involved. In a total of 14 out of the 18 panels studied, the critical crack started in area B and grew aft into area F. In three cases, all at the higher levels, the critical crack started in area F and grew forward into B. In only one instance was area III involved.

With randomized-step loadings, area III was critical in 10 out of 14 panels. The two types of loadings are in agreement that areas C, D, E, G, and H, which occurred frequently, are not critical but no one level in the constant-level tests has consistently given an adequate clue to the location which proved critical in the randomized-step tests.

Cumulative Damage

Although the mechanism and mathematics for the accumulation of fatigue damage in a specimen are presently under study on a broad front, the linear cumulative damage concept as expressed by Palmgren and by Miner, perhaps because of its simplicity, is apparently the most widely

LOADING		NUMBER OF PANELS TESTED	NUMBER OF PANELS WHERE CRACK INITIATED IN AREA-							
			B	F	III	C	D	E	G	H
CONSTANT LEVEL	±1.725	2	2	0	0	0	0	0	0	0
	±1.000	4	2	2	0	0	0	0	0	0
	±.625	5	3	1	1	0	0	0	0	0
	±.425	3	3	0	0	0	0	0	0	0
	±.350	2	2	0	0	0	0	0	0	0
	±.250	2	2	0	0	0	0	0	0	0
TOTAL		18	14	3	1	0	0	0	0	0
RANDOMIZED STEP	GUST	10	2	0	8	0	0	0	0	0
	MAN'V'R	4	0	2	2	0	0	0	0	0
TOTAL		14	2	2	10	0	0	0	0	0

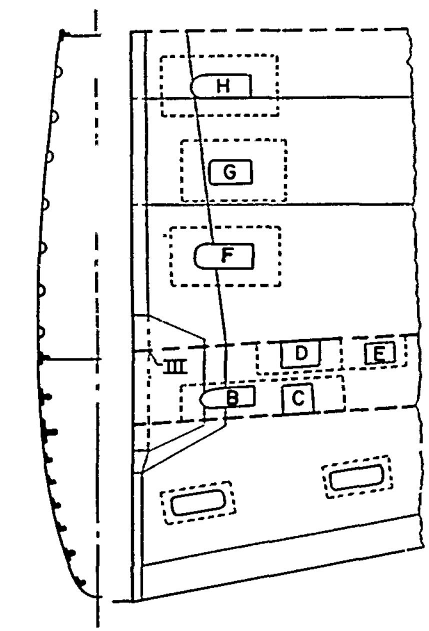

Figure 8.- Location of critical crack.

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used. The randomized-step tests on the C-46, embodying two such diverse loading schedules, together with the load-lifetime information of the constant-level tests appear to be well adapted to a study of the applicability of the various cumulative damage hypotheses. Some preliminary results are available for the linear hypothesis. These results are based on the load-lifetime curves given in figure 9. Three curves are shown, based on the results of the C-46 constant-level tests. These curves give the number of cycles to initiation of the first crack, to initiation of the critical crack, and to final failure; they are denoted in figure 9 as curves FC, CC, and FF, respectively. The curves were established by fairing a smooth curve through the mean of the pertinent lifetimes at each of the six load levels of the constant-level tests. These average load-lifetime curves were used as the basis of the subsequent cumulative damage calculations. Miner's familiar linear cumulative damage concept, as presented in reference 6, states that damage can be expressed in terms of the number of cycles applied at a given stress level divided by the number of cycles to produce failure at that stress level. When the summation of these "increments of damage" at several stress levels becomes unity, failure will occur. The inception of a crack, rather than complete failure of the specimen, was considered by Miner to constitute failure. Therefore, in any treatment of cumulative damage based upon Miner's hypothesis it is only fair to hope for values of 1.0 for $\Sigma \frac{n}{N}$ when constant-level and variable-amplitude load-lifetime data for crack initiation are used. It is also necessary to consider the effects of scatter upon the $\Sigma \frac{n}{N}$ results. Miner's own supporting test data contained scatter about the average value of $\Sigma \frac{n}{N}$ of 1.015 for crack initiation. The scatter limits for Miner's $\Sigma \frac{n}{N}$ values for 22 specimens ranged from 0.61 as a minimum to 1.49 as a maximum.

Initiation of first crack.- The results of the cumulative damage calculations for the C-46 tests are shown in figure 10. Summations of the cycle ratios, $\Sigma \frac{n}{N}$, are shown along the horizontal scale. For comparison purposes, Miner's data from reference 6 are shown at the top of the figure. The C-46 summations are for several events and for several kinds of data. Consider, for example, the initiation of the first crack. The gust loading data, shown by the open bars, averaged 1.46 with scatter between 0.33 and 2.48. The four maneuver loading panels averaged 1.03 varying between 0.85 and 1.30. In order to evaluate this scatter the results for the 18 constant-level tests are shown as the dark bars. The constant-level test can be considered as a special case of a spectrum with only one load level. These values scatter between 0.4 and 1.75, with the average at 1.00. Thus the average values of $\Sigma \frac{n}{N}$ for both the gust and maneuver loading data fall close to 1.00. The scatter for the 10 gust values is slightly greater than that of the 18 constant-level test values and the scatter for the four maneuver spectrum values of $\Sigma \frac{n}{N}$ falls well

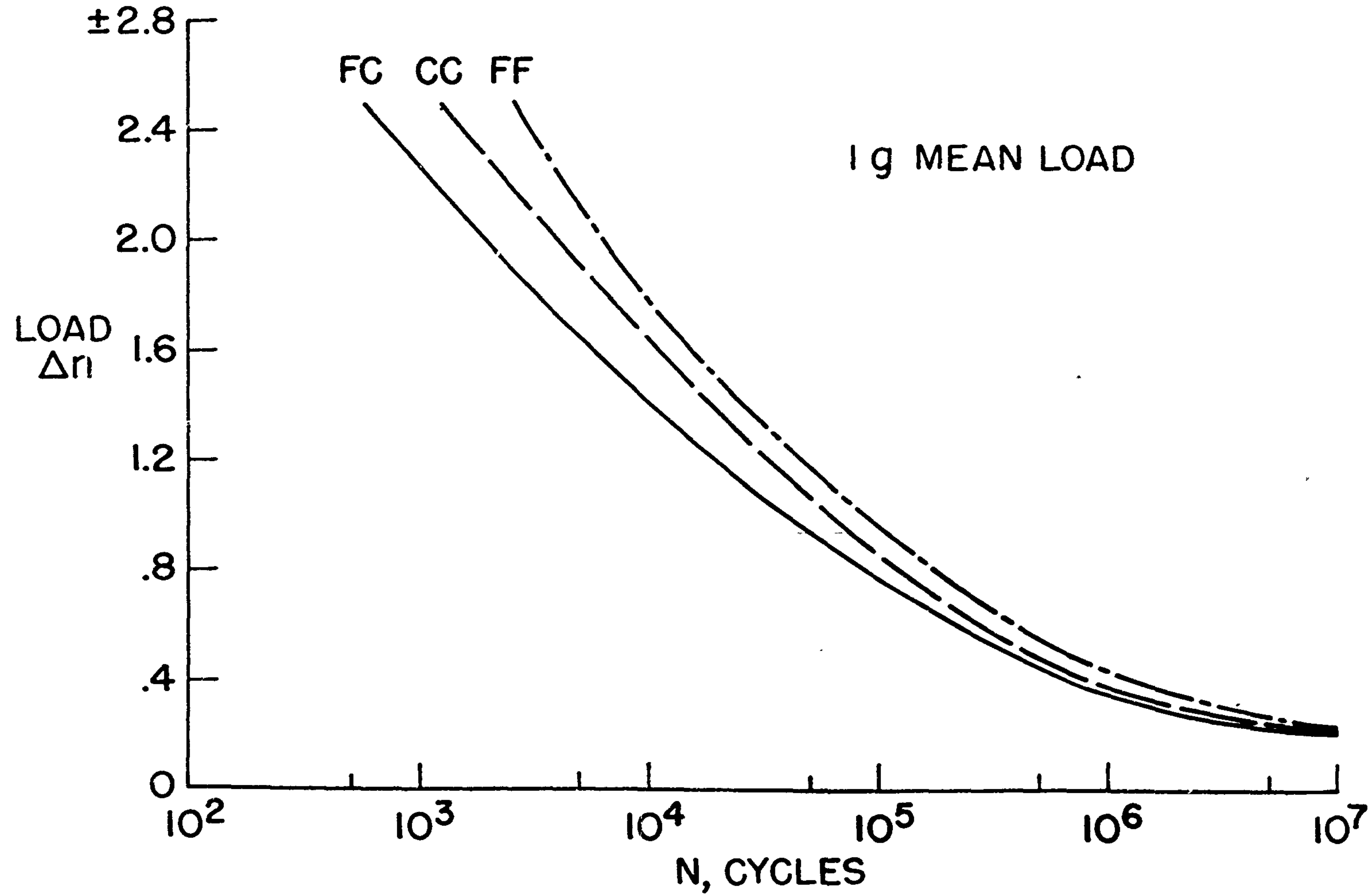

Figure 9.- Load-lifetime curves, C-46 constant-level tests.

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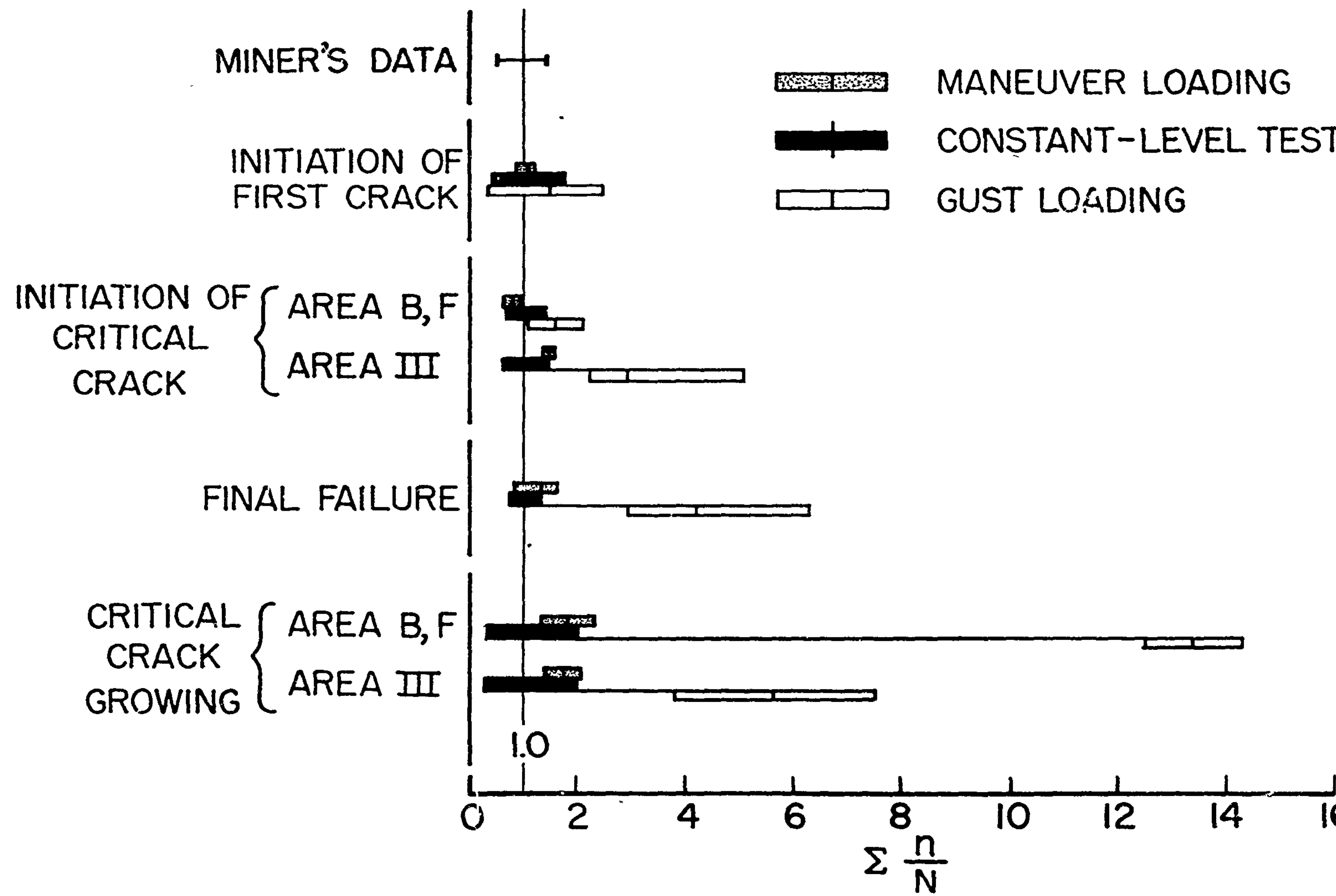

Figure 10.- Summary of cumulative damage data.

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within the scatter band of the constant-level results for initiation of first crack. In general, Miner's cumulative damage concept seems to have provided a reasonably adequate representation of the lifetime to initiation of the first crack.

Initiation of critical crack.- Just as values of $\Sigma \frac{n}{N}$ for initiation of first crack in the variable-amplitude tests were in good agreement with constant-amplitude results, the values of $\Sigma \frac{n}{N}$ for critical crack initiation also show reasonable agreement for areas B and F. The apparent disagreement of area III in the gust loading is felt to be caused by the fact that the constant-level lifetime curve for area III critical crack initiation is not well defined at the lower load levels. A crack occurred in area III only three times in the 18 constant-level tests. Therefore, the values of N used for calculating $\Sigma \frac{n}{N}$ to critical crack initiation are the result of averaging the shorter lifetimes for areas B and F at the low constant-load levels. These lower values of N will result in high values of $\Sigma \frac{n}{N}$ for area III critical crack initiation. In any event, if a correction could have been made based on the true lifetime values for area III, the $\Sigma \frac{n}{N}$ for the gust and maneuver loadings would have been reduced approaching more nearly the value of 1.0.

Final failure.- For final failure the average value of $\Sigma \frac{n}{N}$ is 1.23 for the four maneuver load specimens. The average value of $\Sigma \frac{n}{N}$ is 4.19 for the 10 gust load specimens, very much higher indeed than would be indicated by a linear cumulative damage concept.

In the description of crack locations, it was pointed out that, although cracks existed in areas B and F in both constant-level and randomized-step tests, they were seldom critical in the randomized-step tests, final failure being associated with a crack which initiated in area III. There are two very interesting points involved with these randomized-step tests.

One point is the failure of the area B - area F system to propagate under random loading; some mechanism seems to have been at work which slowed the rate of crack propagation greatly as compared to the constant-level loadings. In fact, the area B - area F cracks were slowed to the extent that the late developing area III cracks were critical. This effect is common to both the maneuver and gust loadings. The other point is the relatively much greater delay in final failure with the gust loading as compared to the maneuver loading. Since the crack system is the same in both cases, this difference must be associated with the differences in shape of the two spectrums.

The general agreement in critical crack initiation and marked delay in final failure indicates the discrepancy in final failure is brought on by the behavior of the propagating crack under variable loading. The delaying effect on the propagating crack is very apparent in the gust spectrum results but not in the maneuver spectrum tests. Using $\Sigma \frac{n}{N}$ values based upon a constant-level critical crack growth lifetime curve, the growth of the two area B critical cracks in the gust loading tests took 13 times longer than the same crack growth during the constant-level tests. The eight area III critical cracks took approximately six times longer to propagate to final failure than in the constant-level tests. The $\Sigma \frac{n}{N}$ values for critical crack growth during the maneuver loading tests averaged about 1.75 for both areas III and F.

McEvily and Illg, of the Langley Research Center, have pointed out a possible explanation for the mechanism of nonpropagating fatigue cracks. A mechanism of this type could account for the observed results in the C-46 tests. In explanation of nonpropagating cracks, it is suggested that once a fatigue crack has started, the application of a high load cycle could cause plastic deformation in the material at the head of the crack. This plastic deformation would result in the introduction of residual compressive stresses around the end of the crack during the unloading portion of the cycle. This introduction of compressive stresses would reduce the effectiveness of subsequent cycles at lower load levels and crack growth would thus be delayed.

These considerations, as applied to the C-46 program, can explain the observed results. The first cracks to initiate in area B or area F having started, their subsequent propagation could be delayed by the occasional high loads, to such an extent that a new crack starts in area III. In area III the nominal 1 g stress is 2,000 psi lower than in area B or area F (ref. 1). A joggle such as that in area III is also a very different piece of structure than the skin cutouts at B and F and the stress concentrations would undoubtedly differ. It is difficult to assess the relative effects of lower stress level and differing structure. It is believed, however, the start of a crack in area III after area B or F is associated with the lower nominal stress and also that the number of excursions into the plastic range during crack propagation would be relatively fewer for area III. The propagation delay would thus be somewhat less, as observed. Thus, area III which was not critical in the constant-level tests becomes the starting point for the crack which leads to final failure in the variable-amplitude tests.

Granted the existence of this growth inhibiting mechanism, it is easy to see why $\Sigma \frac{n}{N}$ at final failure is higher for the gust loadings than for the maneuver loadings. In the gust loadings there are relatively more cycles at the very low load levels. Under a linear cumulative damage

hypothesis, they contribute to $\Sigma \frac{n}{N}$, but it is precisely these low-level cycles which, because of stress relief at the head of the crack, would be less effective in producing fatigue damage. The net effect of stress relief, coupled with differences in relative proportions of cycles at high and at low stress levels, is to give a life under gust-type loading which is greater than would be expected on the basis of linear cumulative damage and this increase is greater, the greater the proportion of the load cycles at low stress levels.

In general, then, it appears that crack initiation in the C-46 tests is in reasonable agreement with the indications of the linear cumulative damage hypothesis. The tests also indicate that once started the subsequent growth of cracks and development of final failure show systematic departures from the indications of the linear damage hypothesis.

These departures are in line with the qualitative indications of a mechanism of delayed crack propagation proposed by McEvily and Illg. Quantitative treatment of this mechanism would require considerable modification to the linear cumulative damage hypothesis. If real, however, the existence of this mechanism appears to necessitate the inclusion of a wide range of load levels in a fatigue test.

CONCLUDING REMARKS

Comparison of the fatigue sensitive areas on C-46 wings as indicated by constant-level tests and by randomized-step tests shows consistent differences in the indications of the two types of loading. No one level in the constant-level tests revealed as many fatigue sensitive areas as did the randomized-step tests. The first crack in the constant-level tests was often the critical crack, whereas the first crack in the randomized-step test was usually a nuisance crack which did not propagate and did not compromise the structural integrity. No one level in the constant-level tests consistently gave a good indication of the area which proved critical in the randomized-step tests. A limited comparison of fatigue sensitive areas in C-46 aircraft as revealed in flight operations with the results of the laboratory tests indicates better agreement between flight and randomized-step tests than between flight and constant-level tests.

The fatigue life of the C-46 fell naturally into two distinct phases. The first phase, life to crack initiation, was in general agreement with the indications of linear cumulative damage. The second phase, propagation of the critical crack, showed systematic departures from the indications of linear cumulative damage. A qualitative explanation of these departures is offered based on a discussion of nonpropagating fatigue

cracks by McEvily and Illg. In explanation of the C-46 results, it is proposed that crack propagation at low stress would be slowed by occasional cycles of stress which are sufficiently high that plastic deformation could occur at the head of the crack.

Based on the indications of both fatigue sensitive areas and fatigue life, the results of the C-46 study indicate that important information would be missed by a constant-level test. If the mechanism of delayed crack propagation is real, its existence would appear to necessitate the inclusion of a wide range of load levels in a fatigue test.

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SONIC FATIGUE SIMULATION*

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I. INTRODUCTION

In the century or so since Jones and Galton experimented with cyclic stresses in cast iron, fatigue has taken an increasingly important part in design technology. In the modern aircraft industry in particular, the desire for increased performance has led to ever-increasing working-stress levels. In addition, the pursuit of new materials has led to greater improvements in static-stress handling abilities than in fatigue strength. Coupled with the increasing levels of alternating stresses to which a modern aircraft is subjected, these factors have made fatigue failure loom as the greatest single bottleneck in the future advance of the industry.

We will define sonic fatigue as that which either:

a) is induced by coupling a structure to an airborne sound field, or

b) occurs in a frequency range where the stress distribution in the structure can most easily be handled by acoustical analysis techniques.

Along with the growth of fatigue problems in general, supersonic flight, the increased thrust and noise from modern power plants and the increased bandwidth of the resulting excitation has forced sonic fatigue to loom as a major future contributor to our problems.

As the name implies, sonic fatigue is composed of one part sonics and one part fatigue. Indeed, the research problems which must be solved before we can consider ourselves capable of handling the sonic fatigue problem can largely be resolved into sonic problems and fatigue problems. This paper will attempt to outline some major problems in these two areas and the potentialities and limitations which we feel sonic simulation offers in attempting their solution.

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II. FATIGUE PROBLEMS

A. Background

While we cannot, of course, present a history of fatigue in a paper such as this, it may be well to mention briefly some portions of past experience in order to sharpen the contrast with some of our new problems.

The alternating stresses produced by rotating machinery, bridge traffic, railroad wheels, etc., is, to a large extent, sinusoidal in nature. Similarly, the alternating stresses which one can produce comfortably in the laboratory are also sinusoidal. In such a sinusoidal stress pattern, the relationships among peak stress, root-mean-square stress and average stress are completely defined. Even the instantaneous stress to which a sample is subjected is a unique quantity.

Laboratory and field experience with this type of stress pattern have led to the formulation of some empirical relationships which have permitted significant progress in the design of structures for fatigue resistance. The most important of these relationships is undoubtedly the observation that, in general, the number of cycles of alternating stress to which a sample can be subjected before failure is inversely proportional to some first order measure of the amplitude of the alternating stress. This first order measure is generally expressed either as the root-mean-square value of the alternating stress, or the peak value of the alternating stress.

Because of its relative independence of sample configuration, the above relationship provides the engineer with a powerful tool to be used in fatigue design. The most important contribution is to provide the engineer with a scaling law without which laboratory investigation in the fatigue field becomes unwieldy, time-consuming and often essentially impossible.

The engineer, looking for a fatigue life of 10,000 hours and seeking to test a design, may, because of the existence of the above hypothesis, perform his evaluation tests at a value of stress many times higher than the design level, thereby decreasing the time necessary for a laboratory test. In addition, it appears that this hypothesis is valid over a wide range of frequencies where the stress-wavelength in the material is very large compared to the sample size and where the cumulative heating by hysteresis under the influence of the alternating stress pattern is unimportant. This additional flexibility permits the engineer not only to run his test at a value of stress which is higher than that anticipated in the field, but at a frequency of application of stress which is higher than that anticipated. The combination of these two, of course, greatly facilitates laboratory testing.

A second fundamental relationship for predicting times of failure under repeated sinusoidal stresses is a cumulative damage law which purports to predict the life of a specimen subjected to stresses at a variety of levels. The most common relationship is that generally attributed to Miner and hence often called Miner's hypothesis. It has only been in recent years that Miner's hypothesis has shown limitations. As a result of these observed limitations, many researchers are now seeking substitutes for Miner's hypothesis, both by the collection of statistically significant quantities of data and by an examination of the micro-physical phenomena which accompany fatigue failure. These problems will be discussed further in Section B.

In addition to classical damage-accumulation theory, it has been recognized that below some value of peak alternating stress, no fatigue damage is induced in the structure. If one plots the peak stress to which a sample is subjected versus the number of cycles to failure for ferrous metals, one finds that the S-N curve seems to become parallel to the number axis at some point. This point, for convenience, is usually taken at a figure of 10 to 20 million cycles. Even for a test where the strain is applied at 30 cycles per second, such a test represents approximately 200 hours of testing time. To check that an asymptote really exists, it would be necessary to extend such a test by at least two orders of magnitude. Neither the homogeneity of the samples available nor the feasibility of running 20,000 hour tests warrants such an extrapolation. Indeed, the inhomogeneities found would require many such tests and hence an inordinate length of testing time.

Where scaled testing is being done and where the excitation frequencies in the field are those associated with normal rotating machinery, the extension of the S-N curve is an unwarranted refinement from an engineering point of view. Only in some materials (such as the aluminum alloys) where fatigue failures out to hundreds of millions of cycles have proved important, is such extended range testing normally considered imperative. The absence in these alloys of any clear asymptote and their increasing use in the aircraft industry has made them the subject of much long-time testing. We will point out in the next section that some of the problems associated with sonic fatigue also impel the engineer to a recognition of long-time test programs.

B. Current Problems

The modern jet engine, to indicate one source of excitation which is of importance in present fatigue problems and will be even more important in tomorrow's fatigue problems, generates a time-varying pressure field in which the instantaneous amplitudes of pressure are essentially randomly distributed with a probability density distribution that is essentially Gaussian. In contrast to the older sinusoidal form of excitation, therefore, we no longer have an expressible relationship among the peak pressure, the root-mean-square pressure and the instantaneous pressure. Were fatigue a linear phenomenon, the

stresses resulting from such a pressure field would not disturb us very greatly. Linear behavior requires simply that the effect of a number of sine waves acting simultaneously be the same as the vector sum of the individual effect of the sine waves acting alone. The communication engineer has long made use of this rule in handling Gaussian noise, since Gaussian noise may be considered to be made up of an infinite number of infinitesimal sine waves having random phase distribution. Similarly, were fatigue a purely statistical phenomenon, as is the case with many collision-produced chemical reactions, the random nature of the exciting function also would cause us no concern. In such a phenomenon we are concerned only with the energy present in the exciting signal and the statistics or temporal distribution of that energy do not concern us.

Unfortunately, the data which exist at the present time indicate that fatigue is neither a purely statistical phenomenon nor a linear one. Indeed, it has been hypothesized that fatigue failure would not exist in a truly linear material. It must be evident that if the engineer is to make the same type of major advance in random excitation fatigue that Miner's hypothesis permitted him to make in fatigue problems associated with sinusoidal excitation, a similar form of scaling law will be required.

I think we may make the assumption that for a random signal where the fatigue life is long compared with one cycle of the mid-frequency of our excitation bandwidth, the fatigue behavior of a sample does not depend on the instantaneous ordering of the stresses applied. That is to say, the damage produced by a long-term random excitation will be the same as that produced by another long-term random excitation having the same statistical characteristics even though the cross-correlation function of the two excitations be identically zero for all values of τ.*

Having made the above assumption, we are now in a position to say that we can talk about noise-induced fatigue failure without having to talk about the fatigue failure caused by a specific time segment of a given noise. (Prior work has shown that the assumption of independence from temporal sequence holds true only when the fatigue life is sufficiently long to assure randomness in the length of excitation sample used.)

Despite this restraining condition, studies of fatigue under the action of random noise are still faced with a major problem. We would like, for scaling purposes, a relationship such as that shown in Equation (1):

$$\frac{N_1}{N_2} = f\left[\bar{S}_1, \bar{S}_2\right] \tag{1}$$

* Modern communication theory throws some doubt on this statement, but without it this field would be totally inaccessible.

Equation (1) states that the fatigue life N_1 at some value of $\bar{S}_1$ is a definable function of the fatigue life N_2 at some other value of stress, $\bar{S}_2$, where the symbol $\bar{S}$ is used to denote the rms value of the alternating component of stress. In the sinusoidal case, the fundamental hypothesis mentioned earlier tells us that this function is given by $[\bar{S}_2/\bar{S}_1]^n$ where the value of n depends on the material under test.

We would also appreciate having an equation of the form of Equation (2).

$$N_1(\bar{S}_1,\omega_1) = g[N_2(\bar{S}_1,\omega_2)] \tag{2}$$

Equation (2) tells us that there is a definable relationship between the fatigue life at some value of stress and some value of mid-band frequency ω_1 and the fatigue life at the same value of stress but at mid-band frequency ω_2. Again, referring to our sinusoidal case and considering N to be expressed in cycles rather than time, most of the data indicate that, over a reasonable frequency range, g is simply unity.

Our basic current fatigue problem is the fact that neither the f of Eq. (1) or the g of Eq. (2) is specifiable for random-excitation fatigue at this time. Freudenthal's work indicates that even for a simple laboratory test (the stresses are sinusoidal and the peaks are allowed to take any one of six values randomly determined) the simple form of the scaling function of Eq. (1) is no longer valid. Even these data, however, leave considerable question whether a true random noise would produce values which differ very significantly from those produced by such a simple laboratory experiment. Such experiments as Freudenthal's are, of course, the way in which we must ultimately hope to arrive at the scaling laws of Eqs. (1) and (2) for random excitation.

An additional problem facing us in the use of random excitation testing and its application to random-noise fatigue is the fact that most engineering work which deals with statistical processes deals primarily with central-tendency phenomena. It is highly likely that the significant attributes of a random signal, insofar as its effect on producing fatigue damage, may be handled more significantly by utilizing extreme-value theory than by central-tendency theory.

An example may serve to illustrate this point. Figure 1-A shows the S-N curve for an ideally brittle material when tested using a sinusoidal excitation. For any value of stress above S_o the material is good for one cycle. For any value of stress below S_o the material has an infinite life. Such a curve is what we mean by a non-fatiguing material. If we now subject this material to an alternating stress

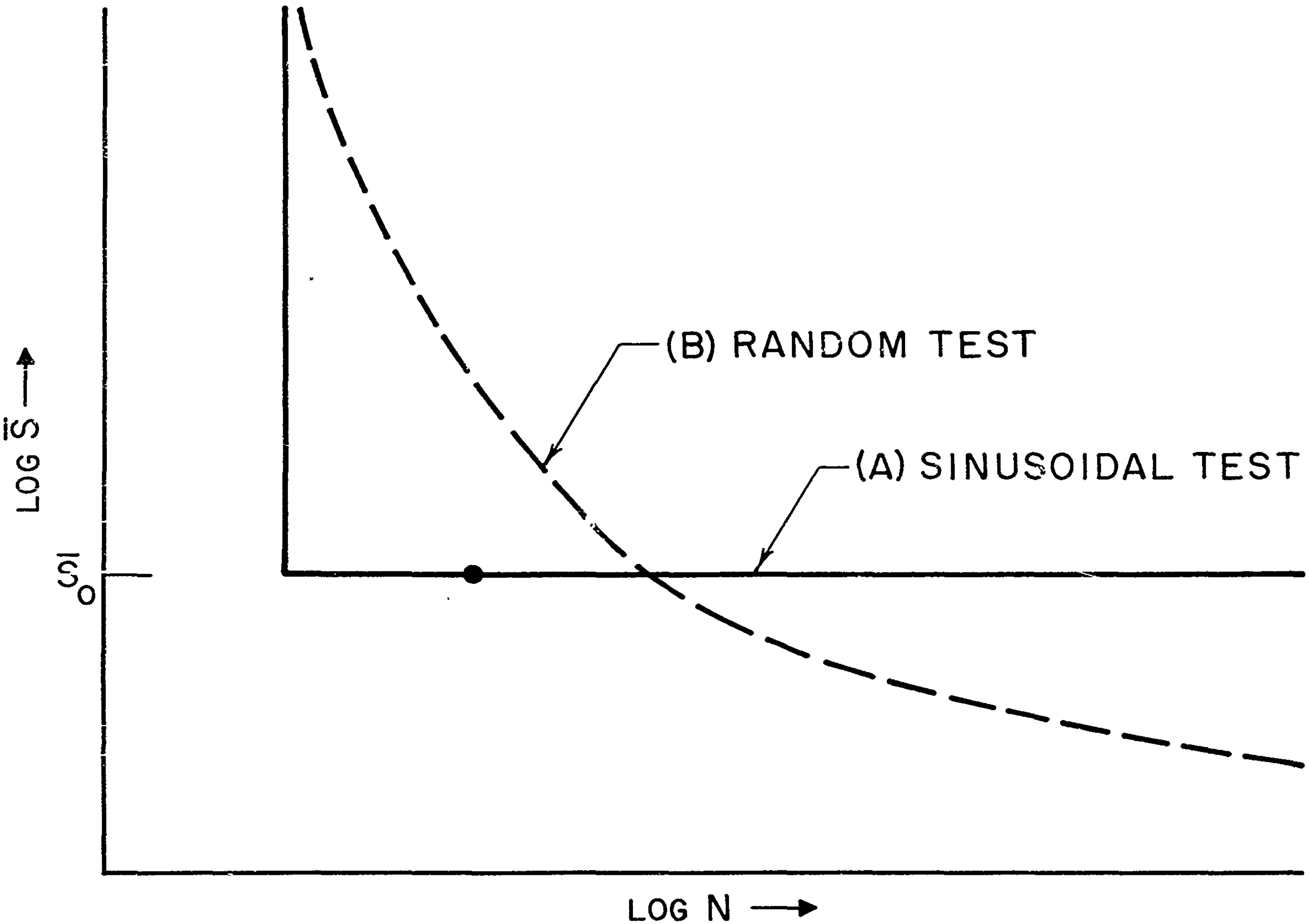

FIG. 1 S/N CURVES FOR AN IDEAL, BRITTLE, NON-FATIGUING MATERIAL UNDER TWO CONDITIONS OF TEST.

which has an ideal Gaussian distribution, our data may have a fairly large scatter, but when we plot the average value of N obtained at any stress we get quite a different curve such as that shown in Fig. 1-B. The shape of this curve arises from the fact that no matter how small the root-mean-square value of stress gets, for a true Gaussian distribution there is always a finite probability of the stress exceeding some threshold value. Similarly, the Curve B is asymptotic to the vertical portion of Curve A, because no matter how large the root-mean-square value of stress is made, there is a finite probability that you will go through at least one cycle without exceeding the threshold value. Obviously, then, the curve of Fig. 1-B has a shape which is dictated far more by the extreme-value statistics of the excitation than by the central-tendency portion of the probability curve.

If we do not confine our attention simply to a threshold case as we have here, but consider the influence of some hypothesis such as Miner's, we get even more pronounced results, since such a hypothesis says roughly that the importance of a given stress reversal is proportional to the amplitude of that stress reversal. Under such conditions the tails of the probability curve are multiplied by the values of their abscissas and a bimodal distribution results with an even greater emphasis on the stresses distantly displaced from the central value.

We have shown that it is important to consider the tails of any distribution with which we work but we have not shown that these tails are not automatically similar for any noises which we may consider. Under the section on acoustic problems we will discuss one aspect of sonic fatigue which severely affects this question. For the present, however, we need only consider normal testing techniques. Ballantine has shown that the normal peak-to-rms distribution for a white noise after it has passed through conventional electronic equipment rarely exceeds four. In normal central-tendency work such a limitation is not a severe one, because there are only about six chances in 100,000 of finding a peak which is greater than four times the rms value for a true Gaussian function. Where, however, we are interested in extreme value behavior, we must rewrite that last sentence to read: "Fully six out of every 100,000 cycles will be more than four times the rms value in height." Since these values are of high importance because of their amplitude, it may do well to insure their presence. In combustion noise, for example, recent unpublished work has indicated that pulses are present at least as far out in amplitude as six σ's.

If our work were to indicate that we are interested in instantaneous power input, we would be concerned with the quantity p^2 times the normal probability function. This quantity, which is the bimodal Maxwellian distribution, of course, places an ever-increasing emphasis on the extreme value quantities. It is evident, then, that when two random excitations are compared, it is not only necessary to be quite specific about their Gaussian distribution shape near the origin, but one must also be extremely careful in specifying the peak-to-root-mean-squared ratio for each of the two noises.

A more esoteric but by no means less significant aspect of the signals which must be taken into account when attempting a scaling law for random excitation is the phenomenon of band limiting. A Gaussian distribution specifies nothing about the conditional probabilities which are determined by bandwidth limiting of a random excitation. A true white noise, of course, has a spectral energy density which is independent of frequency. Such is not necessarily the case for practical noises. There is some indication that the spectrum level of the noise produced by a jet engine drops off, at very high frequencies, at a rate approximately equal to 3 db for each doubling of frequency. Such a drop-off, of course, implies a relatively constant octave-band spectrum. While the ordinary theory of damage accumulation for sinusoidal excitations assumes that the function g in Eq. (2) is unity, such an assumption is unwarranted where derivative-dependent phenomena become important. If, for example, we take a very simple modification of the standard S-N curves and use for our abscissa the quantity t, or time to fatigue failure, it is evident that the high frequency stress alternations become important. Indeed, were the response from our sample independent of frequency as far as stress pattern is concerned, the specification of equal octave-band levels over the spectrum would, for a curve using t as the abscissa, impute the major importance to the high frequency end of the spectrum.

Naturally, the high frequency end of the spectrum is often damped by the characteristics of the structure itself. There is, however, reason to believe that a series of electronic component failures some years ago was caused by an excitation in the region of approximately 1 megacycle. If, in addition to using t as the abscissa, we postulate that there are certain types of fatigue in which heating plays a major role, it can be shown, for viscous losses, that the high frequency terms again will assume a predominant role. Evidently then, it becomes important to specify any bandwidth restrictions on our stress excitation in deriving a scaling function for random excitation.

In sum, we can see that, in order to derive a scaling law which will permit the same type of flexibility of laboratory testing that exists for sinusoidal excitation, we must develop a knowledge of fatigue behavior under the conditions of changes in rms level, peak-to-rms ratio, and band limiting. Naturally, such a simplified analysis has assumed the existence of a normal distribution of amplitudes. Should we find, as field measurements progress, that we are dealing with distributions which have non-zero coefficients of skewness or kurtosis, we may be forced to re-evaluate the universality of our scaling functions.

Evidently then, the greatest single fatigue problem facing the field at the moment is the derivation of a scaling function which will permit testing with other than a one-to-one duplication of field conditions in the laboratory. Whether such a scaling function can be derived by simple empirical means such as was the case with Miner's hypothesis, or whether it will require a reasonable, fundamental understanding of

the micro-physical phenomena accompanying fatigue is a moot question. The tremendous number of variables in random testing as compared to sinusoidal testing, however, would indicate that pure empiricism is perhaps a very long procedure.

III. SONIC PROBLEMS

A. Background

The coupling of a sonic field, which is a spatial distribution of time-varying pressure, to a structure has long been recognized as a problem having engineering importance. For example, such coupling determines to a large extent the transmission loss of sound in passing through solid partitions, the detectability of submarines, the behavior of microphones and the ultrasonic cleanability of surgical equipment. Obviously then, a great deal of work has been done in analyzing this coupling. For simple structures, there are exact analytical solutions which relate the spatial distribution in pressure to the spatial distribution of stresses in the structure. Such analytic solutions are generally applicable only to pressure fields whose time and space functions are relatively simple analytic expressions and to solids whose boundaries can be made to coincide with the coordinate axes in a system displaying both orthogonality and relative analytic simplicity.

Unfortunately, the design of aircraft, and the design of structures in general, is such that we may not place on the designer the onus of forcing his boundaries to coincide with such axes. Similarly, the sound field generated by combustion and by the power plants now under consideration, as well as by the aerodynamic turbulence encountered by the moving aircraft, are such that they are neither analytic nor simple. Indeed, as indicated in the previous section, the sound fields must be described as statistical functions not only of time but of space. While the preceding section discussed the problems inherent in going from a known stress distribution having a statistical time function to a fatigue life, the present section will describe some of the difficulties in going from a statistical sound field to a known stress distribution. It will then be evident that the problem of going directly from a measured statistical sound field to an estimated fatigue life is still somewhat far off.

B. Current Problems

In the early days of room acoustics, the solution of the eigenvalue functions for a rectangular room, while analytic, proved to be only marginally useful in analyzing the behavior of such rooms under the influence of such quasi-random functions as speech and orchestral music. To handle these problems and the problems engendered by rooms which were not simple orthogonal structures, reverberation theory, the theory of randomness, concepts such as frequency and spacing index and diffusion were introduced. With the aid of these assets, it became possible to analyze the "average" response of a room. Average responses, like root-mean-square pressure, are a measure of central tendency. Very few investigators were interested in the form of analysis which would permit of extreme-value calculations. Bolt in this country and Meyer in Germany did perform the type of analysis which permitted the calculation of peak pressures in the room, although it did not serve to

locate them. In fatigue analysis, of course, we are interested in the concentrations of stresses which will occur in a structure and in the location of those stress concentrations.

Dyer and others in the field have shown that the problem of waves travelling in a solid medium interrupted by holes, boundaries or other changes in property, can be treated as a scattering problem and the stress concentrations calculated. Again, however, such calculations have been done for relatively simple structures and relatively simple time functions. In order for them to be applicable to the type of fatigue work now facing the aircraft field, such analyses must be extended, either by theoretical work or by experiment, to actual physical structures and actual time functions. Fortunately, it appears that these two problems may be somewhat separable in that if we know the time function we may analyze it in terms of its components, and if we know the reaction of a structure to simple excitations, we may synthesize it for a more complex excitation. Naturally, such a synthesis can be done only in the linear region at the present time, whereas fatigue takes place primarily in a non-linear region. Indications are, however, that the non-linear effects which are present will permit our simple synthesis procedure to arrive at a somewhat conservative over-estimate of the maximum stresses which may be encountered.

The linearity described above is a linearity of the structural medium rather than a linearity in the air. When we analyze the effect of air-non-linearities on the sonic fatigue problem, we find ourselves in much worse shape. If we consider just one aspect of the atmosphere, we can see how this difficulty comes about. The peak pressure in a sound wave whose rms value is 194 db is equal to one atmosphere. Thus, the negative peaks of a sound wave can never exceed this value. We immediately see, then, that a random acoustical signal having a high amplitude will have an inherently negative skew form whose degree of skewness will be a direct function of the wave amplitude. More important, for any phenomenon (such as fatigue) which depends on extreme-value behavior, the negative tail of the distribution function must go to zero at a value of σ corresponding to 194 db. Any scaling effort will be forced to take cognizance of this asymmetric distribution function.

Now let us look at another characteristic of the aerodynamic non-linearity. If we examine a pure sine wave, which we will assume for the moment has a plane wave front, such a wave propagates through the atmosphere with no diminution of amplitude other than that produced by atmospheric absorption and with no measurable wave shape distortion, so long as the amplitude of the alternating pressure is very small compared to the static pressure of the atmosphere. For the case where the alternating component of pressure is not small compared to ambient pressure, such simple propagation does not occur.

Striking photographs have been taken by Allen of the wave form of a plane sinusoidal wave of high intensity as a function of distance. These photographs indicate the same type of phenomenon one gets with ocean waves approaching a beach. The waves change from a sinusoidal shape to a sharply triangular shape and attenuate rapidly. The shape of the wave is a function of the distance through which the wave has propagated. Here again we have a problem which faces those interested in scaling. The distortion-distance function which a high amplitude wave undergoes is a function of frequency and represents a transfer of energy from the low frequency part of the spectrum to higher frequencies. Evidently then, those of us who wish to either scale or reproduce sonic phenomena in the laboratory are going to have to take into account the propagational distortion. Not only that, but we will have to take into account the altitude at which this distortion takes place since it is a marked function of altitude as well as frequency. It will not, therefore, be sufficient to put a microphone on an aircraft wing, record the sound, and play it back through a loudspeaker because by the time the sound has gotten from the loudspeaker to the test specimen, it will no longer have the same wave shape, spectral distribution, or statistical distribution.

Evidently then we are faced, as far as sonic work is concerned, with several problems. Among these we may list (1) the development of experimental or analytic techniques for determining the stress distribution in a structure from a given sonic field, (2) a method of generating, for fatigue simulation studies, a field having the same extreme-value distributions as the practical fatigue-producing sound field, (3) should the central values be important (as indicated by Freudenthal's interaction coefficient term) we must extend item 2 to include a simulation of the central-value distribution, (4) develop a method for scaling sound fields in such a fashion that the scaled field has the same distribution as the unscaled field despite the inherent tendency to increased skewness, (5) before using the scaled fields of item 4 as well as the results of item 1, demonstrate that acoustically generated fatigue can be correlated with fatigue as generated by the known statistical stress distribution as in Section II-B.

Siren test facilities have been and are being designed and constructed as a tool to aid the engineers, physicists and mathematicians in the solution of these problems.

IV. SIREN FACILITIES

A. Nature of the Field

In structural work, it appears that the spatial distribution of pressures in a sound field will be of extreme importance in determining the degree of coupling between that field and any given structure. As a result the siren facility now under design for construction at Wright-Patterson Air Force Base and the facility already largely completed at Bolt Beranek and Newman Inc. have been designed with an eye toward maximum flexibility of spatial shaping of the sound field. The sound sources can be used either in a progressive wave section or can dump their energy into a reverberant field.

When used in either mode, auxiliary sources will be available for partial shaping of the field.

It is imperative for reverberant-field testing, that the normal modes of the reverberant chamber be spaced as nearly uniformly as possible and with as close a spacing as can be achieved. These requirements insure the achievability of as nearly a uniform transmission coefficient as is possible. For this purpose, the volume of the room must be made large and the wall irregularities must be carefully controlled. The Wright Field facility will have a reverberation chamber with a volume of approximately 1/6 of a million cu. ft. and with "bumps" on three mutually orthogonal surfaces. For progressive wave simulation, the facility will have two modes of operation. One utilizing a "horn" and another utilizing a bare siren bank operating into a large anechoic volume.

The "horn" is designated in quotation marks since the dimensions of the horn are such, and the amplitude of the acoustical wave is such that ordinary horn theory becomes virtually meaningless. The horn will have a cross-sectional area of approximately 40 sq ft in the throat down which will pass approximately one million watts of acoustic power. The resulting sound field, which for pure tones will have an rms sound pressure level of 174 db, will suffer significant non-linear distortion in this confined area. By controlling the siren phasing as described later, waves which have essentially plane wave fronts will be generated. The horn is large enough so that auxiliary sirens may be mounted in the throat itself for additional irradiation of the specimen where cross-mode interaction is to be studied.

The large reverberation chamber is designed so that three surfaces can be covered with an absorptive material having an absorption coefficient of approximately 0.8. The high intensities involved, and the resulting spectral shift in energy, are such as to make this coefficient of absorption a sufficiently close approximation to an anechoic chamber for use in progressive wave testing. In this mode of operation the horn may or may not be used, and the bank of sirens may radiate directly

into the anechoic space. The specimen being tested can be brought as close to the sirens as desired or as far away as desired in order to simulate the level of spatial distortion which is desired for the program. Again, auxiliary sources can be placed in the anechoic space and the siren bank can be phase-controlled to simulate intersecting modes for studies of simplified spatial field distributions.

The provisions being made for spatial-distribution control are such that it should be possible to work up progressively from a simple plane-progressive wave to a complex random reverberant field in gradual steps. If the research program is to achieve a solution of the problems mentioned in Section III-B, such a logical progression of complexity will be a necessity.

B. Sources

It was realized early in the design phases of this program, that in order to arrive at predictive techniques, it would be necessary that the sources be capable of generating a wave which could undergo the same logical progression in complexity in the time domain which the facility provided in the space domain. In addition, for the reasons described in Section III-B, we realized that it was necessary to provide a source system capable of functioning over a wide dynamic range. These considerations have led to the design of a siren complex utilizing some relatively new techniques in siren design.

The sirens are capable of operating at low amplitudes with a pure sinusoidal output. This degree of complexity represents the minimum which was desired. Under these conditions initial investigations can be made of the energy transferred from a sonic field to a stress pattern in a structural ensemble.

The next logical progression will be to study this transfer of energy as the bandwidth of the source is increased, applying perturbation theory to the expected result. In order to perform such experimentation, each siren has associated with it a modulator which is, in effect, a very low frequency siren in series with the actual siren. For reasons which will be clear a little later on, the "rotor" of the modulator is oscillated rather than rotated. By driving the modulator with a sine wave, the output of the final siren becomes three sine waves relatively closely spaced in frequency. The two smaller side bands are spaced from the central frequency by an amount equal to the frequency of oscillation of the modulator plate. Thus, it becomes possible to change the siren output gradually from a pure tone to a group of closely spaced pure tones. By controlling the speed of the main siren, the center frequency of this ensemble can then be swept through a considerable frequency range in order to analyze the frequency dependence of the energy transfer.

Once the narrow band analysis has been performed for groups of sine waves, as described above, the next step in complexity will be to determine the energy transfer for narrow bands of noise. For this

purpose the modulator plate is oscillated by means of a tape-controlled hydraulic amplifier. The tape, which will contain low-frequency band-limited random noise, having a specified clipped peak to rms ratio, will cause the siren to produce a complex wave having a bandwidth twice that of the band-limited noise and having superimposed at the center a single frequency. Again, by sweeping the speed of the main siren, this band can be made to move through the frequency spectrum.

Naturally, such a band is relatively complex to handle and it would be desirable to eliminate the pure tone located at the center of the band. In an attempt to achieve such suppression, it is planned that the phase shift between any two sirens will be controlled by special control circuitry to a high degree of precision. If two adjacent sirens have their modulators driven from the same hydraulic output but have their individual outputs (in the absence of modulation) 180° out of phase, then along the plane which is the perpendicular bisector to the line connecting the two sirens, the sirens will function as a dipole for the pure tone components but as coherent monopoles for the noise component. Along this plane, the amplitude of the pure tone component should be zero while the amplitude of the noise should be maintained. Such a "suppressed carrier modulation system" is common in communication work and should permit the generation of a narrow band of noise whose mid-frequency can be swept through a wide frequency spectrum.

In order of ascending complexity, we then have the problem of generating wider band noise. The Wright Field siren facility is currently designed to use a bank of 25 sirens which will yield twelve pairs of sirens having axial symmetry around a line normal to the siren bank. If the narrow bands of noise can be produced by the technique described in the preceding section, these twelve banks will be fed with different bands of noise and with their center frequencies spaced apart an amount equal to the double bandwidth of the noise. Such a technique should permit generating accurately controlled bands of noise having a bandwidth in excess of 600 cycles and having the capability of being swept through the overall frequency spectrum.

Should research show that wider bands of noise are required to secure an accurate simulation of fatigue damage, provision has been made for utilizing a new form of siren designed by Von Gierke and Coles of the WADC Aero Medical Laboratory. An alternative to this addition would be a modulated air horn, providing such a horn is developed to have a significantly greater bandwidth than achievable by present techniques. Since the suppressed-carrier technique wastes half the power put into the system, the ultimate capability of the siren bank will be approximately 3 db lower in this mode of operation.

Again in ascending order of complexity, it is expected that the preceding techniques will be used as a function of level and that the level will be gradually increased to the maximum producible output. Both facilities are designed to produce an intensity at maximum operating conditions of 25 kilowatts per sq ft in the mid-range of operation.

C. Plan of Operation

Current plans for the operation of the facilities call for a progressive analysis of the transfer of energy from a sonic field to a structural field utilizing the combined gradual progressions of field complexity and time-function complexity. A similar progression in the complexity of structures will be pursued, moving from a very simple two-dimensional panel to complex three-dimensional arrays.

By gradually widening the distribution function of the noise fed into the modulators and by utilizing the techniques of electronic distribution-function shaping, it is to be hoped that this progression can be carried to the state where a meaningful scaled sonic fatigue test can be performed on practical structures. It is apparent to all who are working on the program, and must be made apparent to others concerned with the field, that the problems lying between the present state-of-the-art and the achievement of such an ultimate goal are sufficient that a major effort extending over a relatively long time will be required before this goal is reached.

TEST METHODS FOR CONDUCTING
FAIL-SAFE CERTIFICATION PROGRAMS

By

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The problem of completing a satisfactory aircraft structure fail-safe certification program involves conducting tests to prove the airframe is fail-safe under reasonable service damage situations. Prior to tests, tolerable loads in combination with the extent of damage must be established by analysis and reference to service experience with aircraft of similar types.

Test methods are presented for applying loads and for inflicting simulated service damage from causes such as fatigue, corrosion, penetration of critical structures by projectiles, and inadvertent damage during maintenance operations. Tests are conducted under probable service loading conditions of pressurization, gust or maneuver loadings, using full-scale or component tests depending upon the immediate situation at hand and background information available. Lockheed C-130 tests have included full scale fuselage tests supplemented by a number of component and element tests.

The test results must finally be related to actual service difficulty expectancy through life and fail-safe analyses. If the operational future of the aircraft type is readily predictable this latter requirement is rather easily achieved. If many unknowns exist or operational employment of the aircraft is likely to be highly variable then conservative evaluations of test results are in order.

BACKGROUND OF FAIL-SAFE CONCEPT

The fail-safe concept of structural design was somewhat like Topsy - it just grew. Shortly after World War II considerable attention was focused on fatigue problems of commercial aircraft and upon the rather large amount of ignorance existing then concerning methods of analysis and design to avoid fatigue failures in service. This problem was met head-on by such people as McBrearty of Lockheed, References 1 and 2, Strang of Douglas, and Vollmecke of the C.A.A. (now the F.A.A.) by advancement of the fail-safe design concept. This concept was simply to make structures tolerant of fatigue failures as opposed to attempting to design so that fatigue failures would never occur. Many considerations suggested this approach - the many unknowns in the fatigue design problem, possible sources of damage other than fatigue, the inherent fail-safe characteristics of monocoque construction, and the promise of a minimum weight solution, to name a few.

Paralleling, and in some cases preceding commercial design considerations, military authorities became intensely interested in providing fatigue resistant and fail-safe structures. Examples of early considerations were the requirements for dual control systems, battle damage resistant structure, and the encouragement of monocoque construction. Later developments in the military field, particularly as military aircraft accumulated relatively high operational times during peace-time training periods, included sponsoring extensive fatigue resistant design and test programs.

The joint thinking of commercial and military aircraft design personnel supported by efforts of the NACA (now NASA) led to the promulgation of Civil Air Regulations, effective March 1956, which stipulate that civil transport aircraft must either be designed to be fail-safe or be demonstrated to have an extremely high level of fatigue resistance. Similar fail-safe and fatigue requirements are now under consideration by the military for incorporation in a structural design specification of the 8600 series. These specifications will eventually replace the 5700 series of specifications.

FAIL-SAFE DAMAGE AND LOADING CONDITIONS

Verification by test that a structure is fail-safe or tolerant of service failures of structural elements must be accompanied by considerations of the circumstances under which fail-safety must be assured. The loads plus the extent of damage to be tolerated must be established by negotiation and by reference to experience, specification and the structural configuration. The Civil Air Regulations establish the minimum fail-safe load requirements for civil airplanes at approximately 80% of design limit flight loads plus normal maximum fuselage pressure. No fail-safe load requirements are established for military aircraft at

present although proposed MIL-A-8861 does specify such requirements. Neither the CAR's nor the proposed military requirements are specific as to how much of the structure should be considered damaged. It has been Lockheed's policy over the years to make conservative assumptions as to both fail-safe loads and structural damage. We, generally speaking, have had no difficulty negotiating approval of our proposed substantiation procedures.

Safety in flight has been the primary concern in developing fail-safe civil designs. Fail-safety in military designs has emphasized mission completion in addition to safety aspects. Both of these primary considerations then enter the selection of critical structural elements.

Let me cite you a recent case where our C-130 demonstrated a degree of fail-safety. I quote from a Field Service report: "The damage included a hole 2.75" x 2" on the left side of the fuselage in the prop plane area. A longitudinal crack also propagated from the hole in both directions for a total length of 13 inches". This crack stopped at transverse rings and resulted only in partial loss of pressure.

A second recent case that did not actually result in structural failure but does demonstrate the importance of possible damage sources other than fatigue, is extracted from another Field Service report: "Aircraft 3205 returned from a mission with corrosion caused by acid on the right-hand forward section of the cargo compartment floor. Sulphuric acid was spilled from a container in an unmarked box. The box was inadvertently loaded in the aircraft so that the box was upside down. The acid remaining in the bottle leaked out on the floor and caused a corrosive action on various floor parts, the chine cap assembly, and small areas of the fuselage skin".

The above examples are but two of many reports we have received during our extensive experience with civil and military aircraft. Reports have included fatigue, corrosion, battle, inadvertent, and, yes, even advertent damage, and have led Lockheed to provide fail-safe structures with structural elements damaged as shown in Table 1. It should be noted that the structure mentioned is critical for flight loadings - no landing gear parts are included.

TABLE 1

EXTENT OF DAMAGE REQUIRED TO DEMONSTRATE A
FAIL-SAFE STRUCTURE

FUSELAGE (Pressurized or Unpressurized)

1. Any element of an external longeron or stiffener severed.
2. Any ring severed in combination with a fore and aft skin crack of length equal to the ring spacing.
3. Any internal longeron severed in combination with a circumferential skin crack of length equal to the frame spacing.
4. A circumferential or longitudinal skin crack of 15 inches in length.
5. One broken element in a door frame.
6. Failure of any door hinge or latch.
7. Failure of any element of a windshield post.
8. Any single member severed in the wing-fuselage attachment.

WING

1. Failure of the tension portion of any beam web.
2. Any beam cap element severed.
3. Any surface integral skin section severed.
4. Any wing skin or a combination of stringer and skin between stringers severed.
5. Failure of any single element of any wing joint.

EMPENNAGE

1. Vertical and horizontal stabilizers.
   a. Any beam cap element severed.
   b. The tension portion of any beam web severed.
   c. Any surface skin panel or stringer severed.
2. Empennage to fuselage attachment.
   a. Failure of any single bolt, fitting or actuator in the vertical stabilizer or horizontal stabilizer to fuselage attachment.
   b. Failure of any single fitting in the horizontal stabilizer to fin attachment (if appropriate).

Once the test loads and the extent of damage have been established analysis and tests are used as tools in the certification procedures.

METHODS OF TEST

Fail-safe testing is still a developing art even though methods of applying loads for which fail-safety is desired are straight forward and, for the most part, follow well established techniques. Methods of generating damage vary from company to company and are developing with time. In selecting the method of inflicting damage an important consideration is whether or not the damage is to be inflicted while the structure is under load. Some philosophical judgment enters this decision and discussions with procuring and certifying agency personnel are usually necessary to resolve all points. Lockheed's policy has been to generate damage while the structure is under load unless past experience has shown dynamic effects to be negligible.

Fail-safe tests start early and continue thru the development of designs at Lockheed. In the early stages, tests of various structural elements and materials are completed and crack propagation characteristics at various loads with various initial types of damage are observed. As the design progresses components are subjected to similar tests and finally complete full-scale assemblies are subjected to tests where analysis based on past experience and earlier tests indicates a marginal condition exists. Sometimes the background data are so limited that analysis is considered unreliable. In these instances, fail-safe demonstration tests are conducted for civil types and are recommended for military types.

Examples of some recent element, component and full scale tests appear in Figures 1 - 20. These figures, with the notes appearing on them, are essentially self-explanatory and show loading techniques, damage generation techniques and some results of tests at Lockheed's Georgia and California Divisions. It should be noted that the means of generating damage illustrated include saws for pre-load type damage, and remote control equipment including a guided circular saw, a swinging knife, and knife-like projectiles for cuts under load. Other methods of creating damage have included repeated load tests, falling blunt objects, hammers, and projectiles in various shapes. Boeing has used a guillotine-type knife as well as procedures similar to Lockheed's. Other companies appear to be using similar techniques although the writer does not profess to be current with all the methods in use in other companies.

As was mentioned earlier, repeated load or fatigue tests often provide fail-safe demonstrations. The hydro-fatigue test of the C-130 fuselage conducted by Lockheed, under the auspices of the USAF, led to quite a few demonstrations of fail-safe structure after fatigue failures. Figure 21 shows the overall test set-up and Figures 22 - 25 illustrate several of the fail-safe failures. This test also afforded several examples of non fail-safe failures. One of these cases is illustrated in

Figure 26. In this instance there was considerable evidence of a fatigue crack having existed for some time, in fact our inspectors had reported incipient skin cracks, however, the test was continued with the result shown. Needless to say, C-130's in service have incorporated changes to make the structure fail-safe in this area. Another point to mention at this time is that the original C-130 was not intentionally designed fail-safe.

ANALYSIS OF TEST RESULTS

Full scale test results are usually rather simple to analyze - the structure is either fail-safe or not fail-safe. If it is not fail-safe then the test results must be reviewed to establish the most economical way to make the structure fail-safe. If there is no major weight reduction program under way, a successful fail-safe result is generally the end of the analyses.

Element and development type tests represent more complex analytical problems. Generally, as some of the reference titles indicate, Lockheed attempts to extract the maximum in useful data from each test. Our practice has been to use elaborate strain gage configurations to trace crack propagation rates, evaluate dynamic effects, failure sequence, etc. IBM carding of data is extensively used and analysis results are generalized as much as practicable. We consider much of our analytical results and data proprietary in nature, hence no attempt will be made to present these results here. Your attention, however, is directed to the reference list of this paper. Some of the published references should be quite helpful in formulating test procedures as well as assisting in design problems. Lockheed would also welcome specific inquiries concerning our procedures and will be glad to answer any that we can.

CONCLUDING REMARKS

a. Lockheed considers fail-safe design a most important facet of aircraft design. We are devoting as much attention if not more to fail-safe design as to fatigue resistant design, in our effort to assure a safe and operationally reliable aircraft.

b. Fail-safe design has not led, in itself, to excessive weight penalties as most Lockheed monocoque designs are inherently fail-safe and require little additional weight to approach the 100% fail-safe assurance level.

c. Providing a fail-safe design at the same time assists materially in providing a fatigue resistant design since both crack initiation and propagation can be inhibited by careful and judicial placement of material.

d. The test methods now available for conducting fail-safe certification programs are quite adequate and convincing and represent considerable improvement over earlier techniques.

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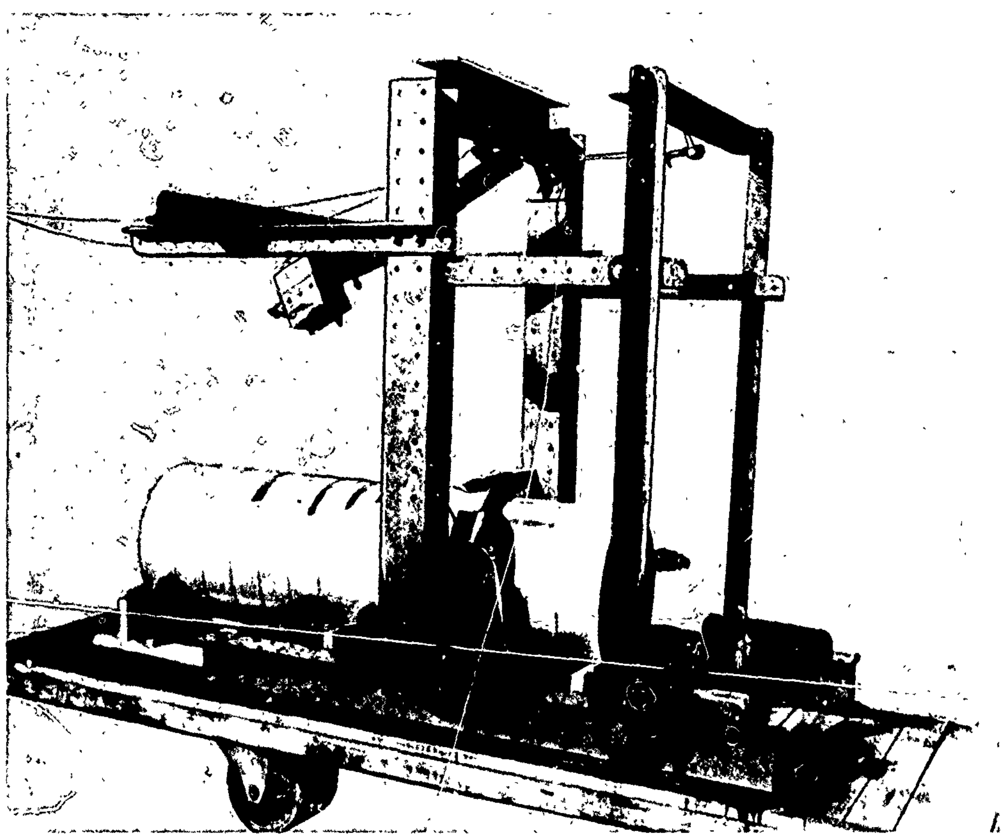


Fig. 1

Fail-Safe Developmental Test

Cylindrical specimens are pressurized with air or water - service damage is simulated by a swinging knife - Note specimen has failed in a non-fail-safe manner.

Fig. 2

Successful Fail-Safe Test

Specimen was tested as in Fig. 1 - Crack was confined by crack stoppers.

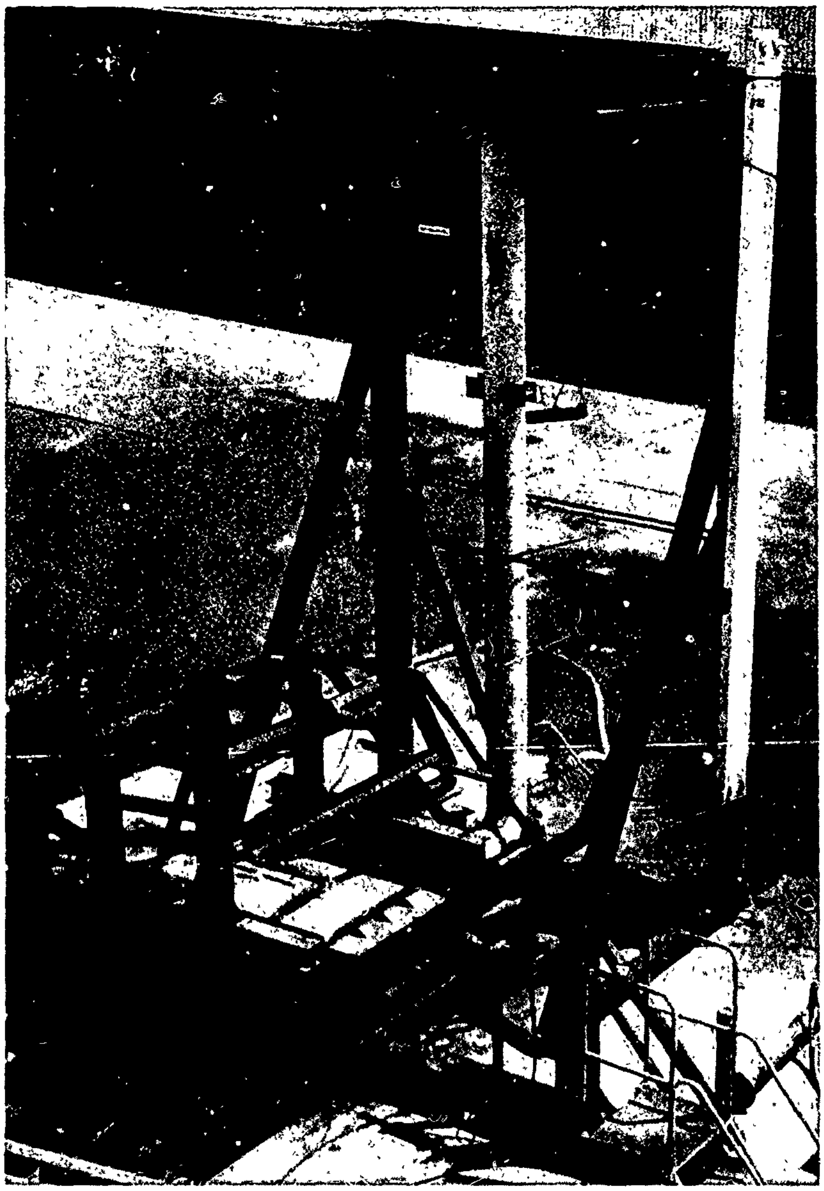


Fig. 3

Fail-Safe Developmental Test Set-Up

'Pillow' type specimens are pressurized with water. Longitudinal loads are applied with hydraulic jacks. Damage is inflicted by swinging or falling knife or by an air-driven circular saw. Internal pressure is maintained for extended times by 'head' of water supplied from reservoir of about 30 times the specimen capacity.

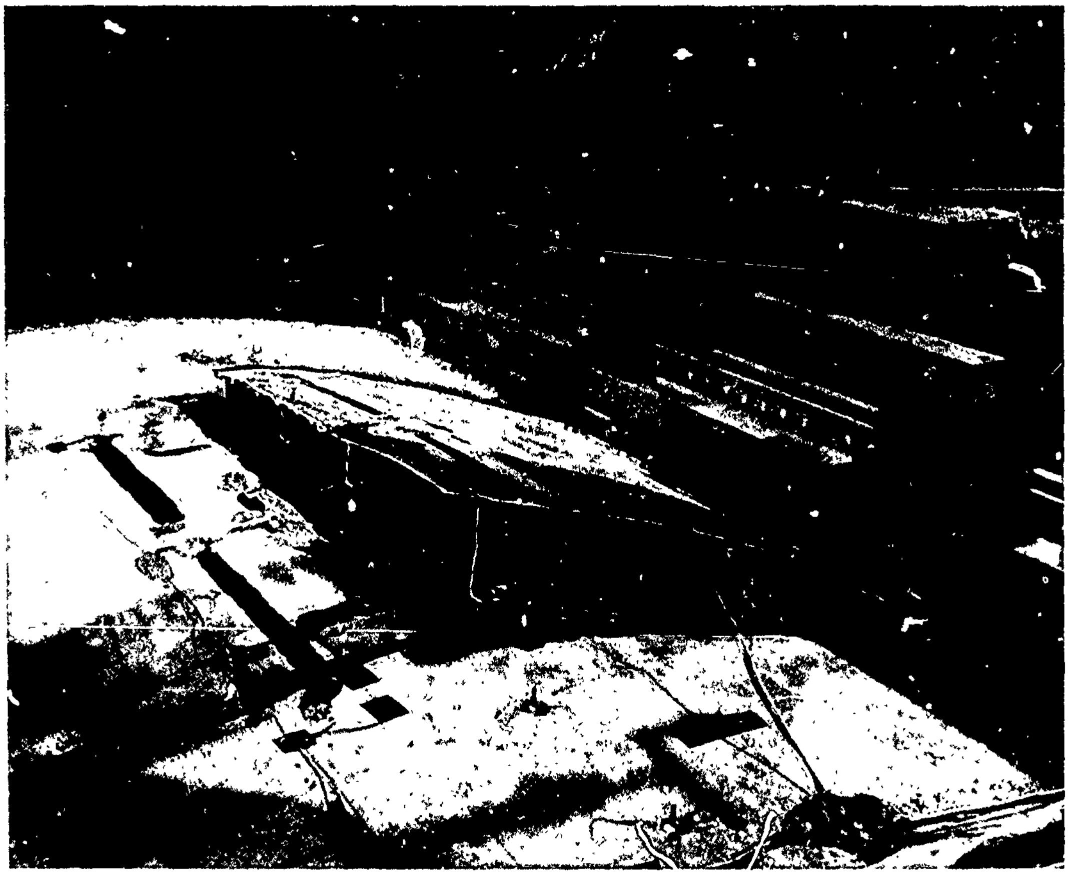


Fig. 4

Non-Fail-Safe Result of 'Pillow' Type Test

Note that skin and 'crack-stopping' rings failed in this test using the apparatus described in Fig. 3.

Fig. 5

Fail-Safe Result of 'Pillow' Test

Ring and skin combination was such that failure did not progress even after a 40-inch cut using apparatus of Fig. 3.

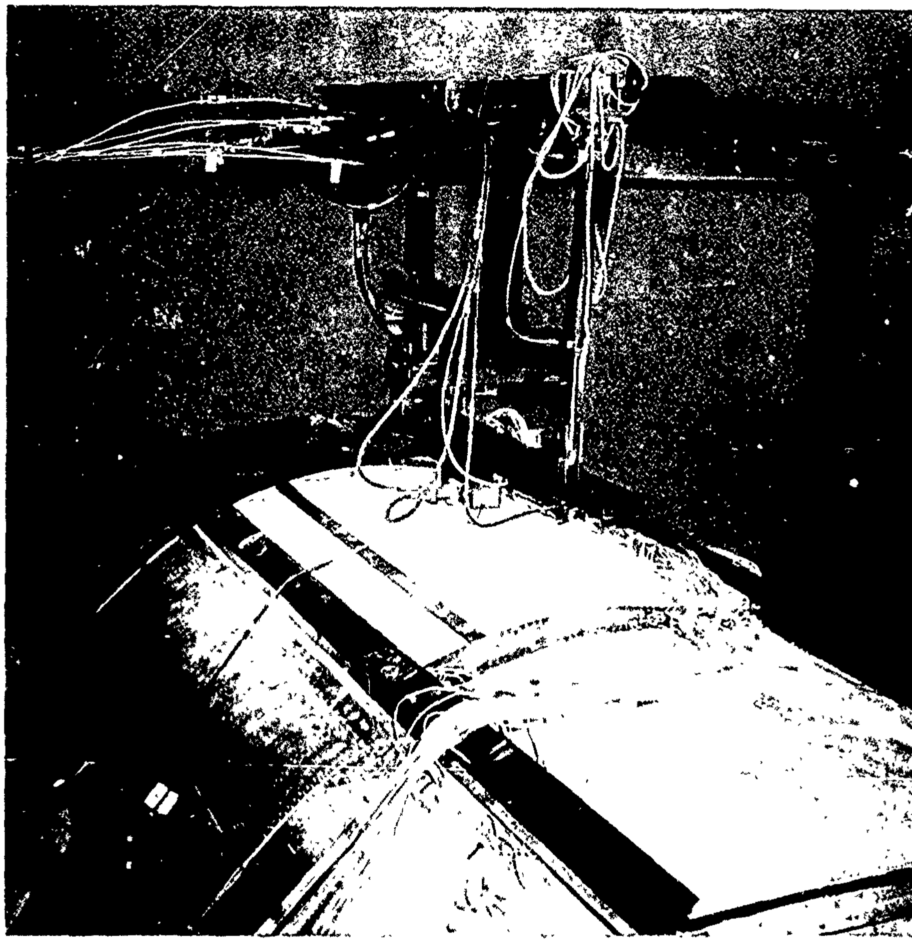


Fig. 6

Full-Scale Fuselage Test

Picture shows top of C-130A fuselage. Entire fuselage was air-pressurized to 7.5 psi, damage being inflicted by a remote control air-driven 8-inch diameter circular saw 1/8-inch thick. Test area is shown framed by unloaded members to confine failure should an explosive type failure occur.

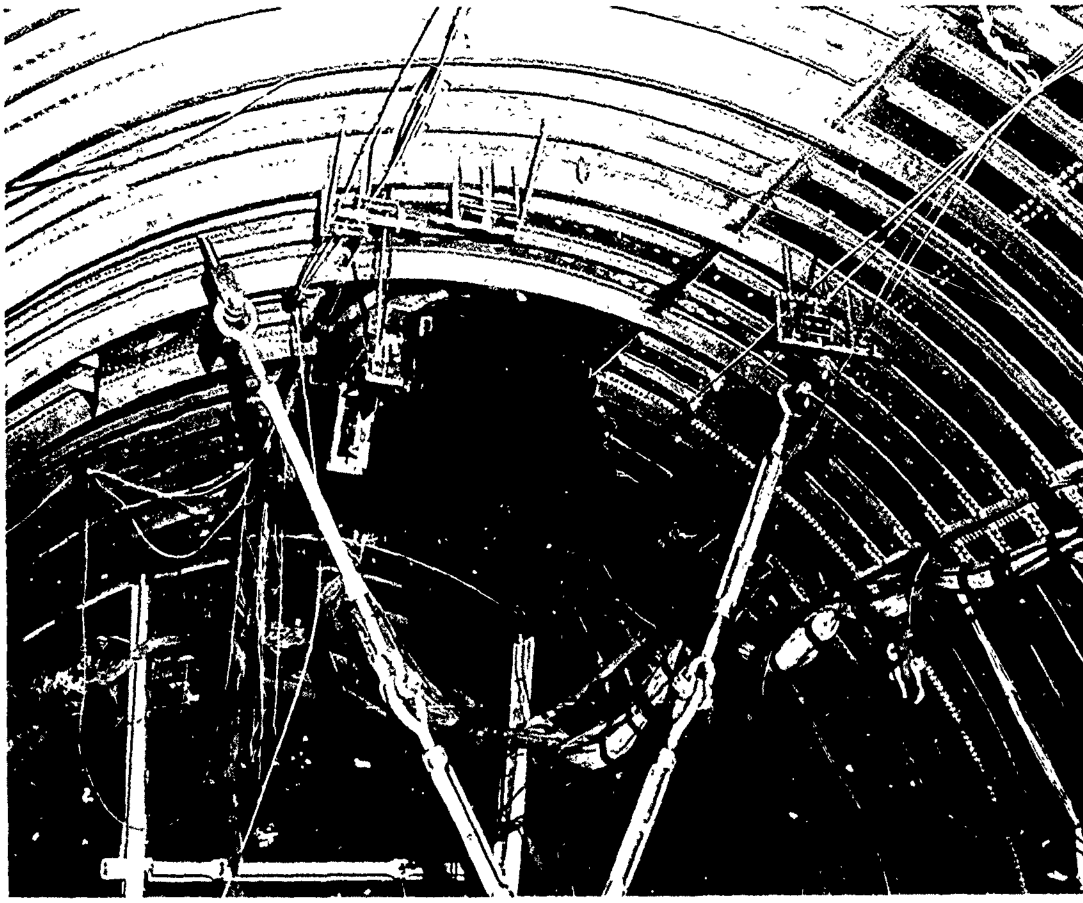


Fig. 7

Full-Scale Non-Fail-Safe Result

View is from interior of C-130A fuselage. Skin and frames failed as expected under combined 7.5 psi internal pressure and fuselage bending after a 13-inch skin cut just as frame was being severed. This result led to reinforcement of all C-130A's in this area.

Fig. 8

Close-up of Air-Driven Circular Saw

Note saw cut in skin - cut also extends thru the frame straddled by the skin cut. This was a 'fail-safe' result of a C-130A full-scale test.

Fig. 9

Explosive Failure of C-130A
Fuselage Top

This series of pictures, taken with a movie camera at 30 frames per second, shows a non-fail-safe failure of the pre-damaged fuselage top as pressure and airloads were increased to expected failure values. This failure confirmed analysis procedures and led to reinforcement of this area of the shell.

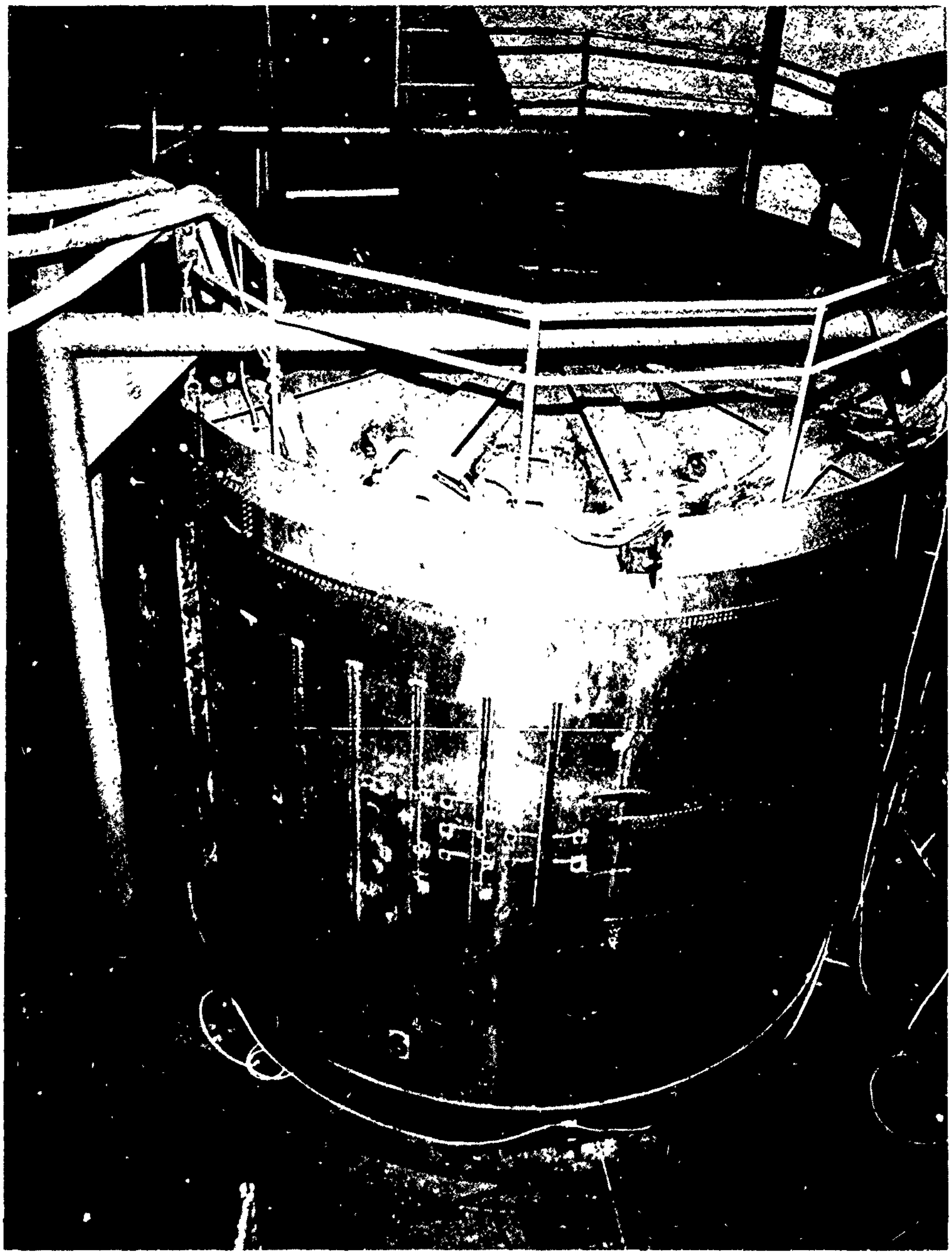


Fig. 10

Fail-Safe Developmental Tests

The picture shows the 85-inch radius air-pressurized cylinder used for tests designed to develop optimum ring-stiffener-skin combinations for pressurized fuselages.

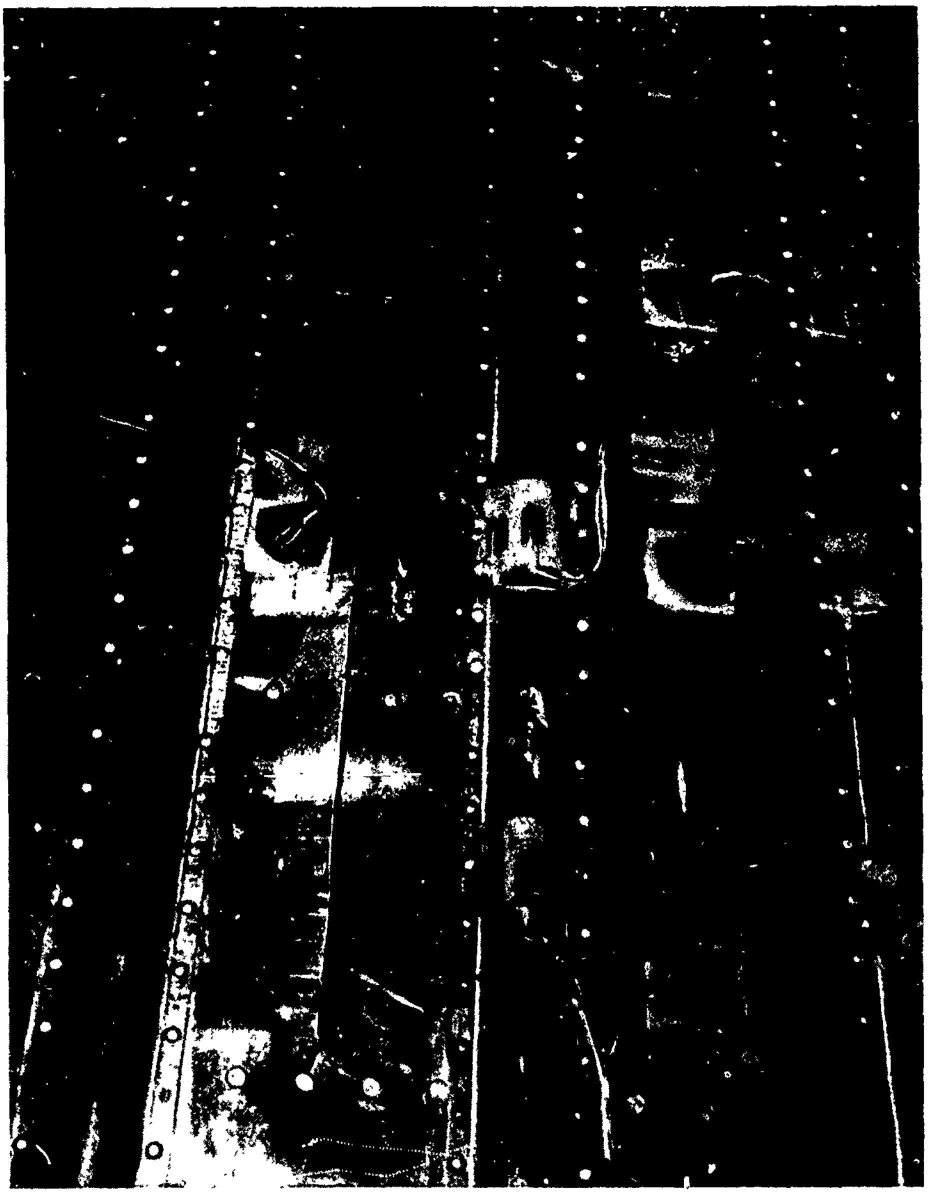


Fig. 11

Fail-Safe Test Instrumentation

This picture shows some of the strain gage instrumentation used to evaluate stress distributions, crack propagation rates, load redistributions, dynamic effects, etc. Also shown is a 'fail-safe'test result.

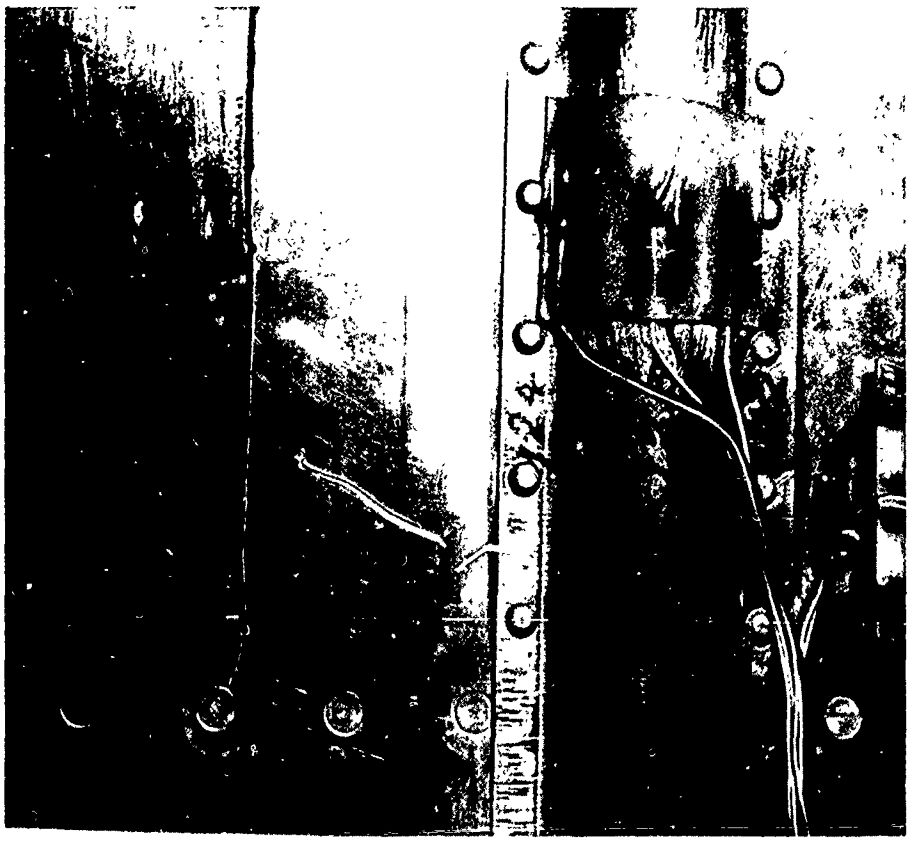


Fig. 12

Development of Fail-Safe Failure

Note small holes indicating ends of succeeding jeweler-sabre-type-saw cuts - Also note fine crack extending to rivet hole from lower end of saw cut. Successive saw cuts were followed by pressure cycles until the crack propagating itself under load stopped at a crack stopper ring.

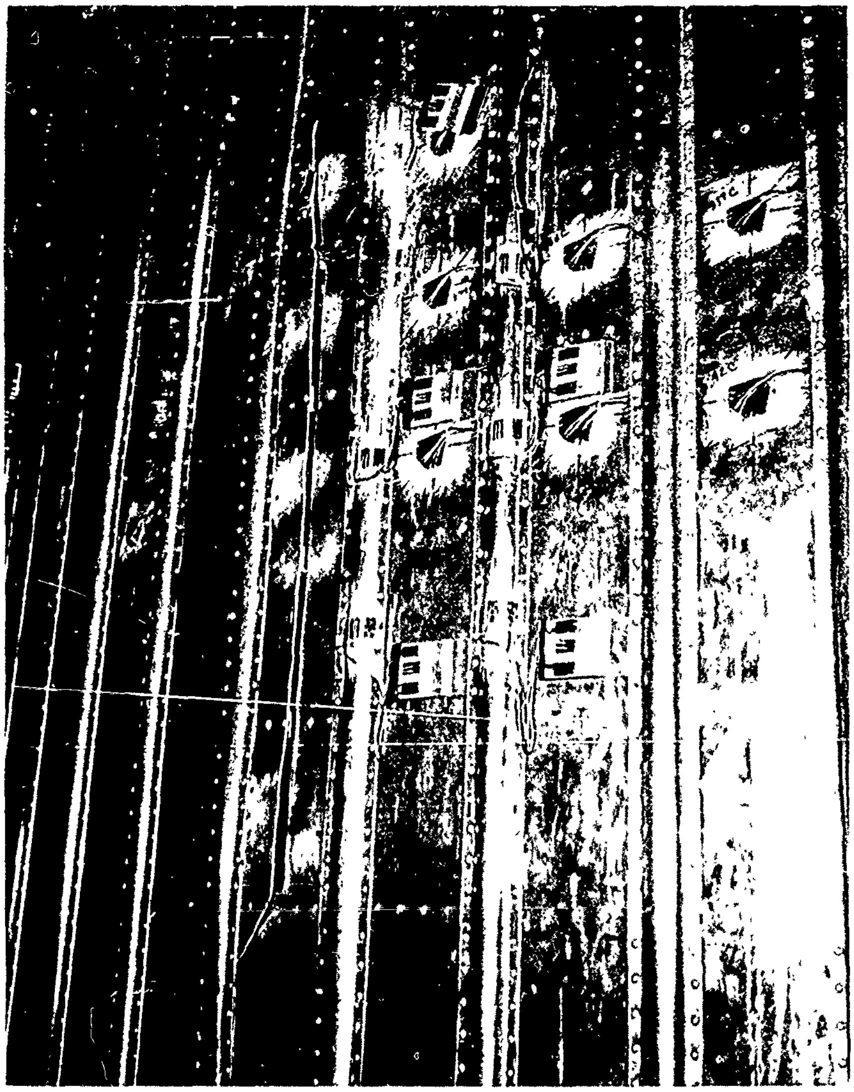


Fig. 13

Fail-Safe Developmental
Test Result

Crack developed in previous Figure was intentionally extended cutting crack stopping rings until pressure could no longer be maintained with an air supply of capacity several times that of the C-130 pressurization system.

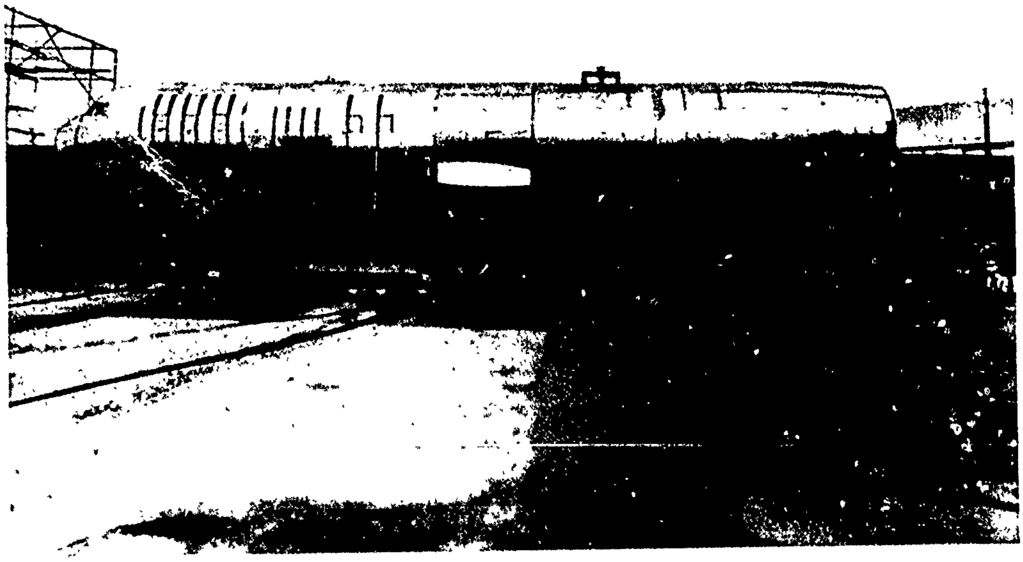


Fig. 14

Electra Full-Scale Fail-Safe Test Set-Up

This picture shows the general test set-up for Lockheed California Division Electra fuselage fail-safe tests. The fuselage was pressurized to normal maximum operating pressure and loaded correspondingly to critical airload conditions. Fail-safe damage was inflicted under these loads.

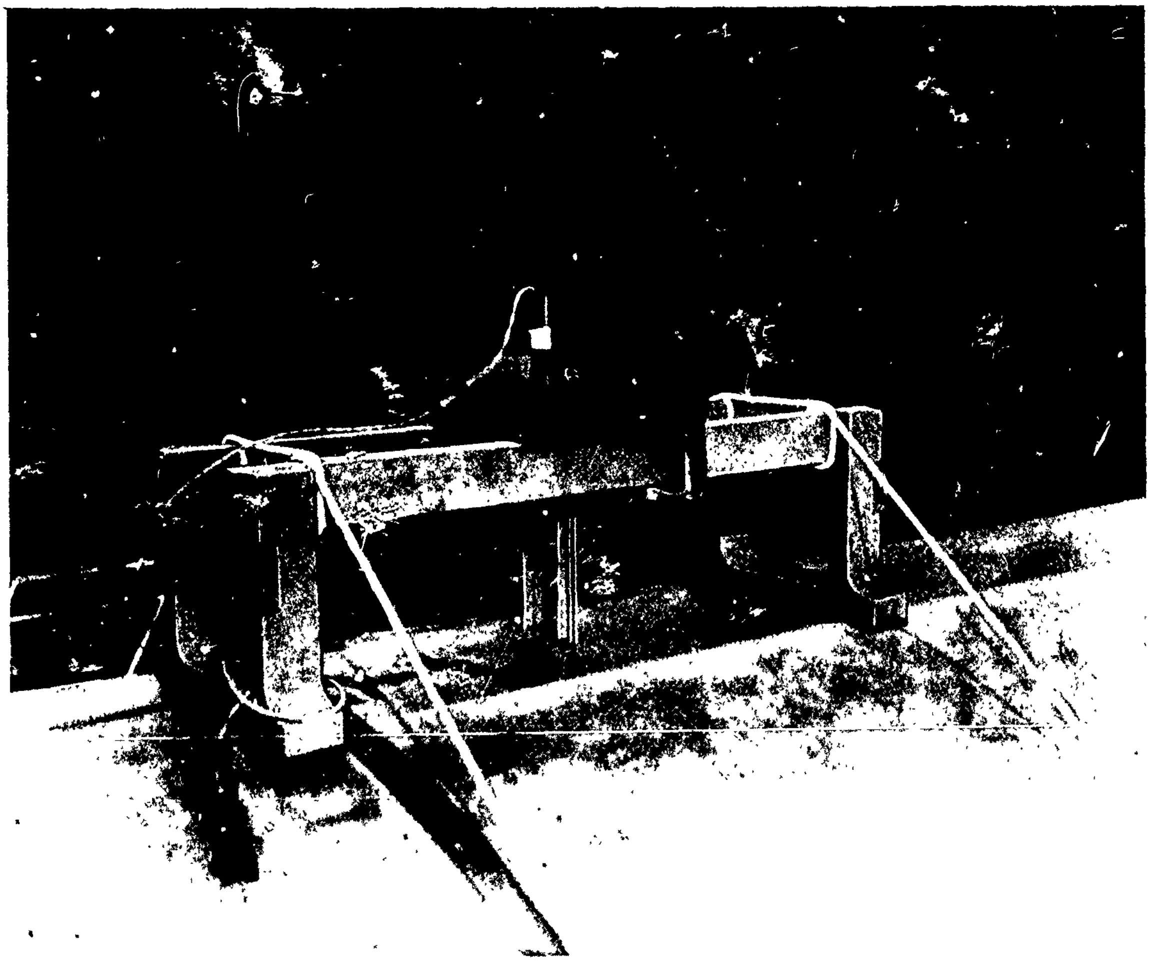


Fig. 15

Electra Spear Gun

This figure shows the spear gun and its support for Electra tests. The gun fires a spear-like knife through loaded structure. The knife is caught inside in a rag-filled box. Powder is allowed to blow through damaged area for photo recording purposes

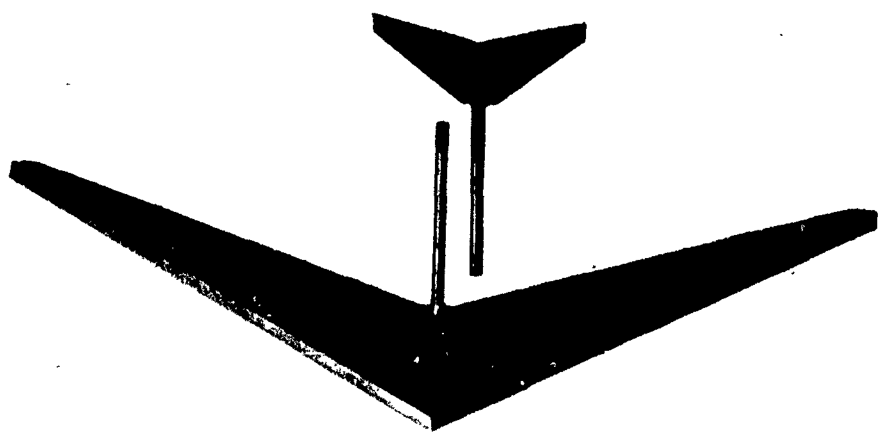


Fig. 16

Fail-Safe Damage Spears

This figure shows 6-inch and 24-inch spears used to shoot through Electra fuselage skin, stringers and rings.

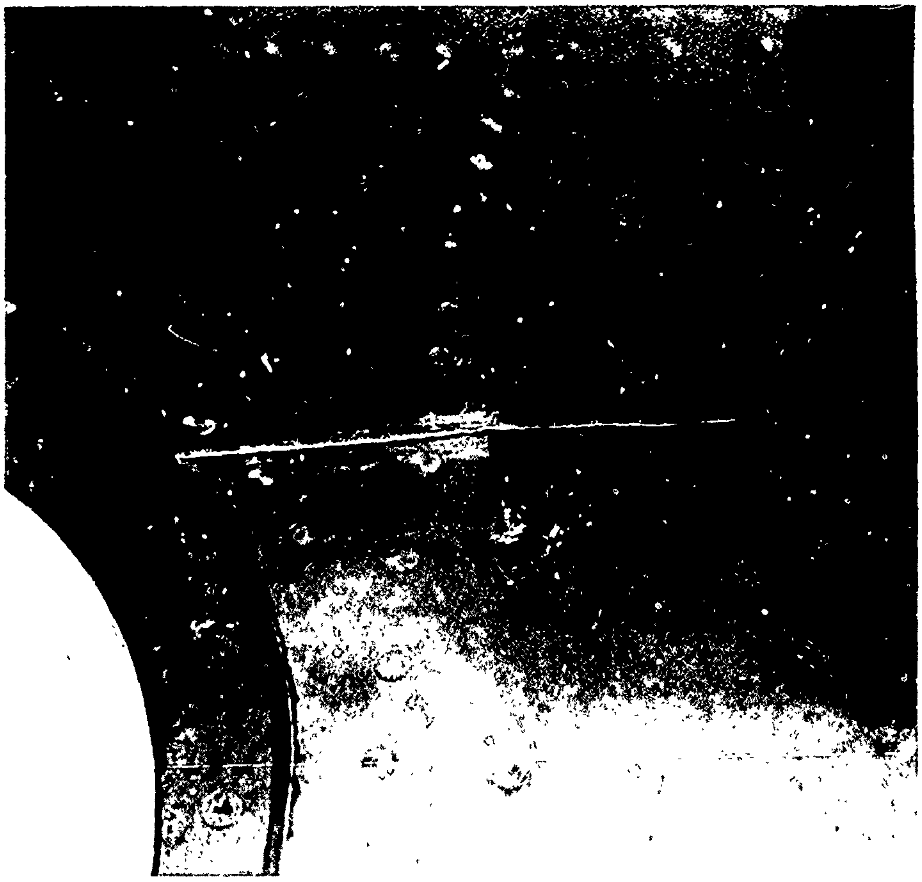


Fig. 17

Electra Fail-Safe Damage

This picture shows fail-safe damage inflicted adjacent to an Electra window. A 6-inch skin cut plus a cut ring was sustained. In this instance, cuts were made prior to loading because dynamic effects had proven negligible in earlier tests.

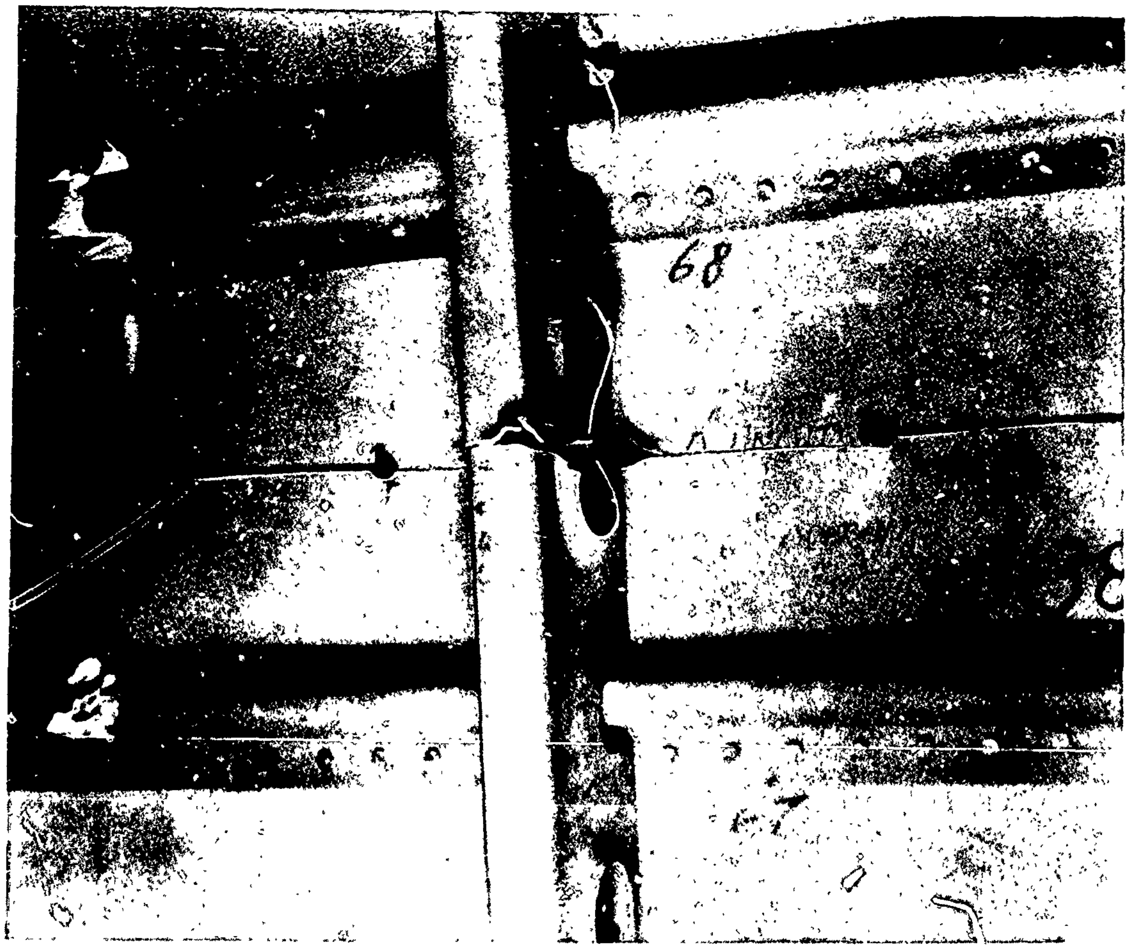


Fig. 18

Electra Fail-Safe Damage

This picture shows Electra fail-safe ring and skin damage inflicted under required fail-safe loads. Area between holes at ends of saw cuts was severed by spear-knife. Overall length of cut was 12 inches; portion cut under load was six inches long.

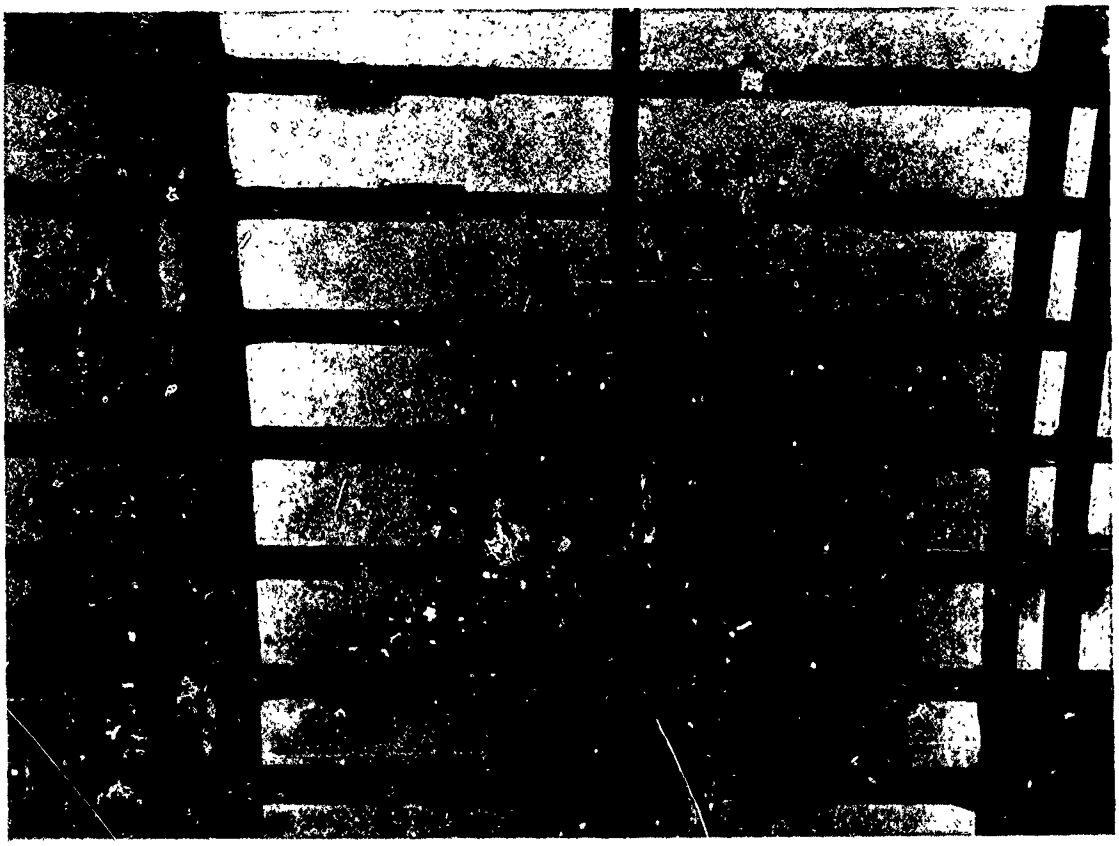


Fig. 19

Electra Fuselage Pre-Load Cuts

This is a view from the interior of the side of the Electra fuselage in the propeller plane. Note the pre-cut skin and stringers forming a one foot by six foot panel. Uncut areas were severed under fail-safe fuselage pressure and air loads by one six-inch spear and two 24-inch spears with the result shown in Figure 20.

Fig. 20

Fail-Safe Damage in Propeller Plane
of Electra

This figure shows the fail-safe damage generated under load in the final test of the elaborate series conducted on the Electra fuselage by Lockheed's California Division. 125% of fail-safe airloads were supported by the fuselage damaged as shown.

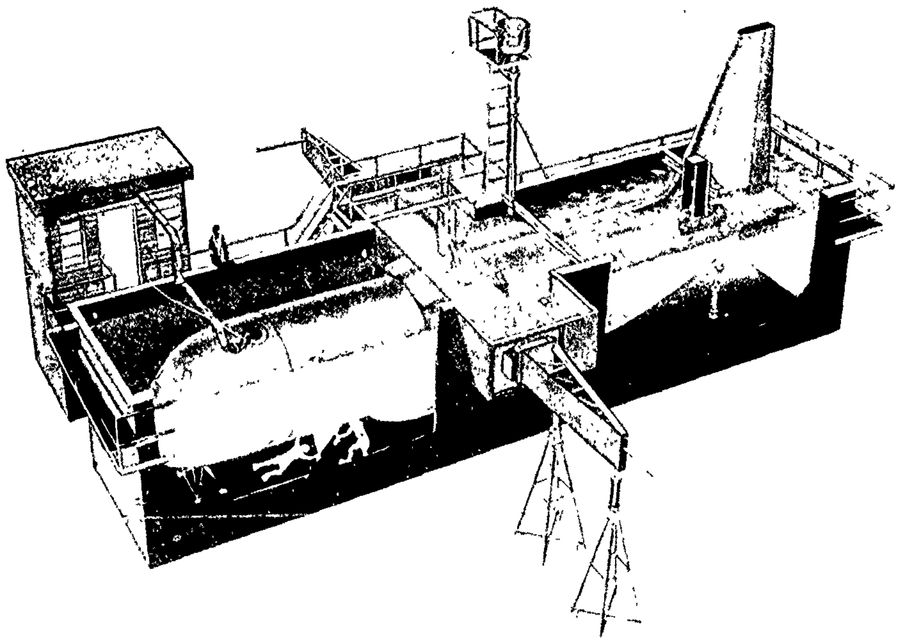


Fig. 21

Over-All View - C-130A Hydro-fatigue
Test Set-Up

20,000 cycles of fuselage pressure plus fuselage air loads corresponding to 60,000 flight hours were applied in this repeated load fatigue test.

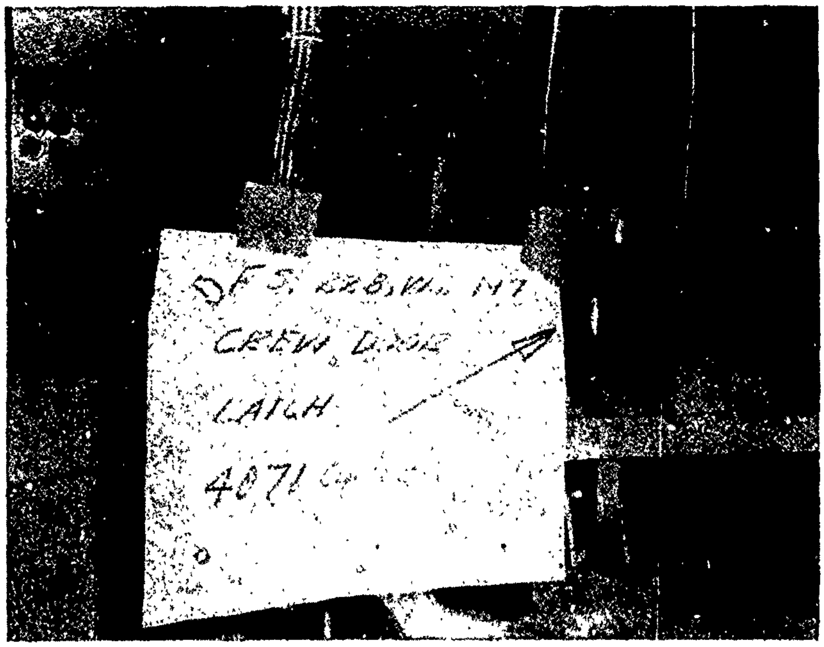


Fig. 22

Fail-Safe Failure of C-130A
Crew Door Retention

This failure occurred at 4071 cycles (equivalent to about 12,000 flight hours) of the fuselage test. Note failed latch. Remaining latch of total of two retained door. However, pressure could not be maintained in fuselage at normal operating level.

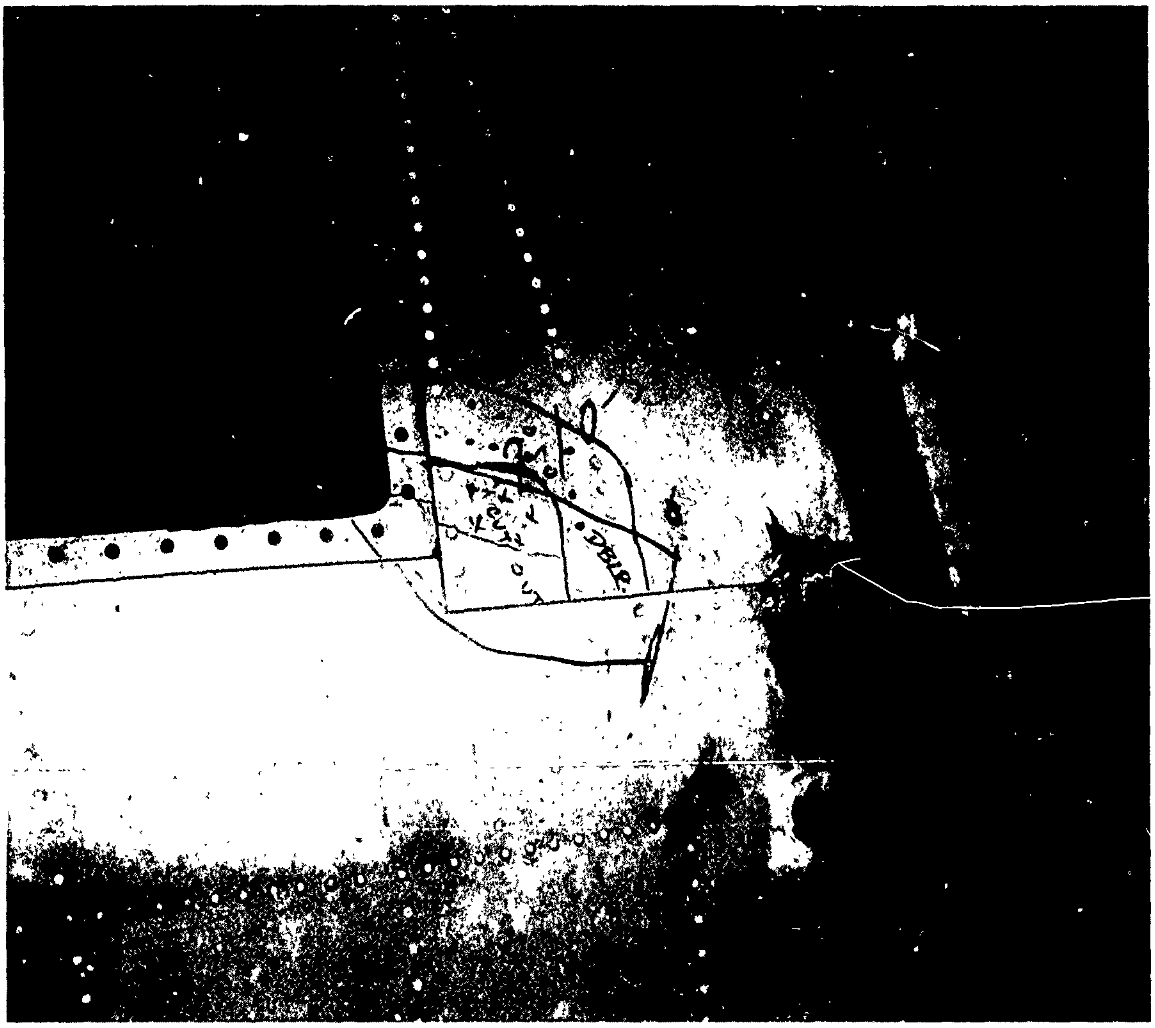
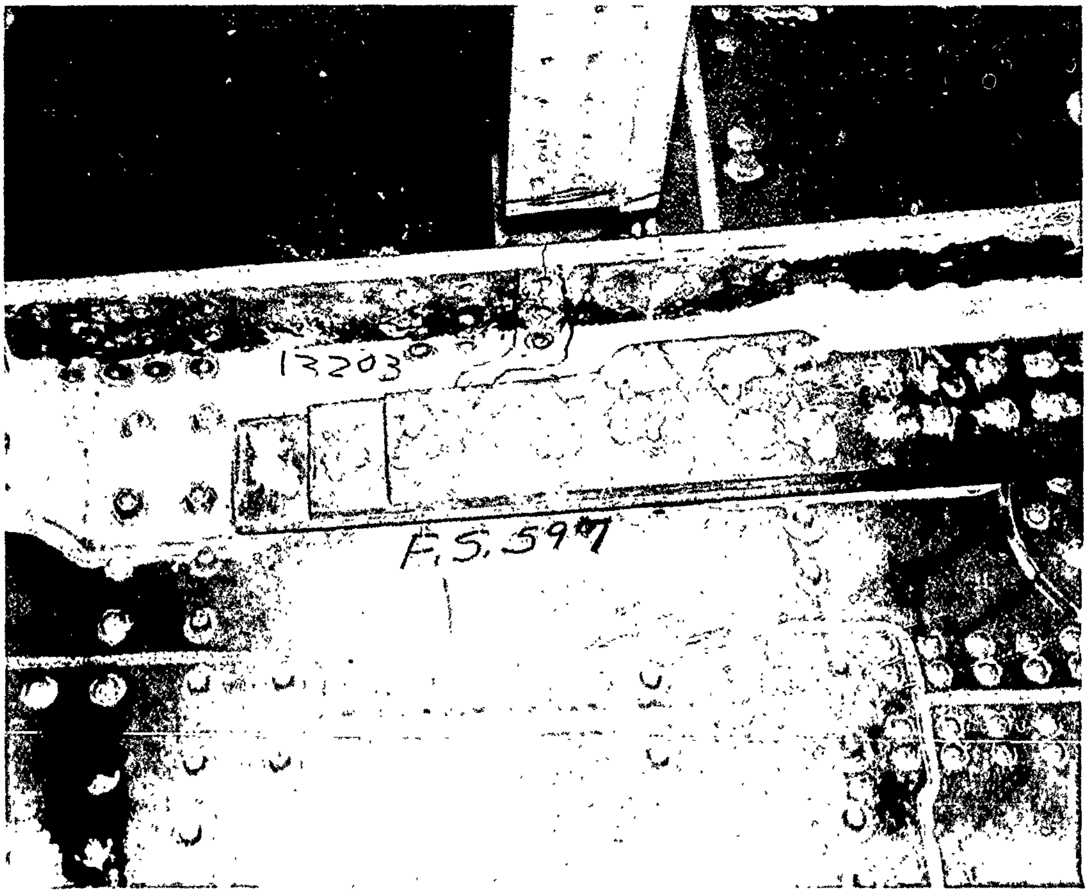


Fig. 23

Fail-Safe Failure of C-130A Pilots Side
Window Framing

Note crack in skin - Window frame and under frame supported load. Crack was found during an inspection - no pressure loss accompanied this failure.



Fail-Safe Failure of Major C-130A Longeron

This failure during the hydro-fatigue test was discovered by inspection. Flight and pressure loads had been sustained in the order of 1,000 cycles (3,000 flight hours) after failure began and the structure still retained its ability to carry all test loads. Reinforcing plates added after failure are shown.

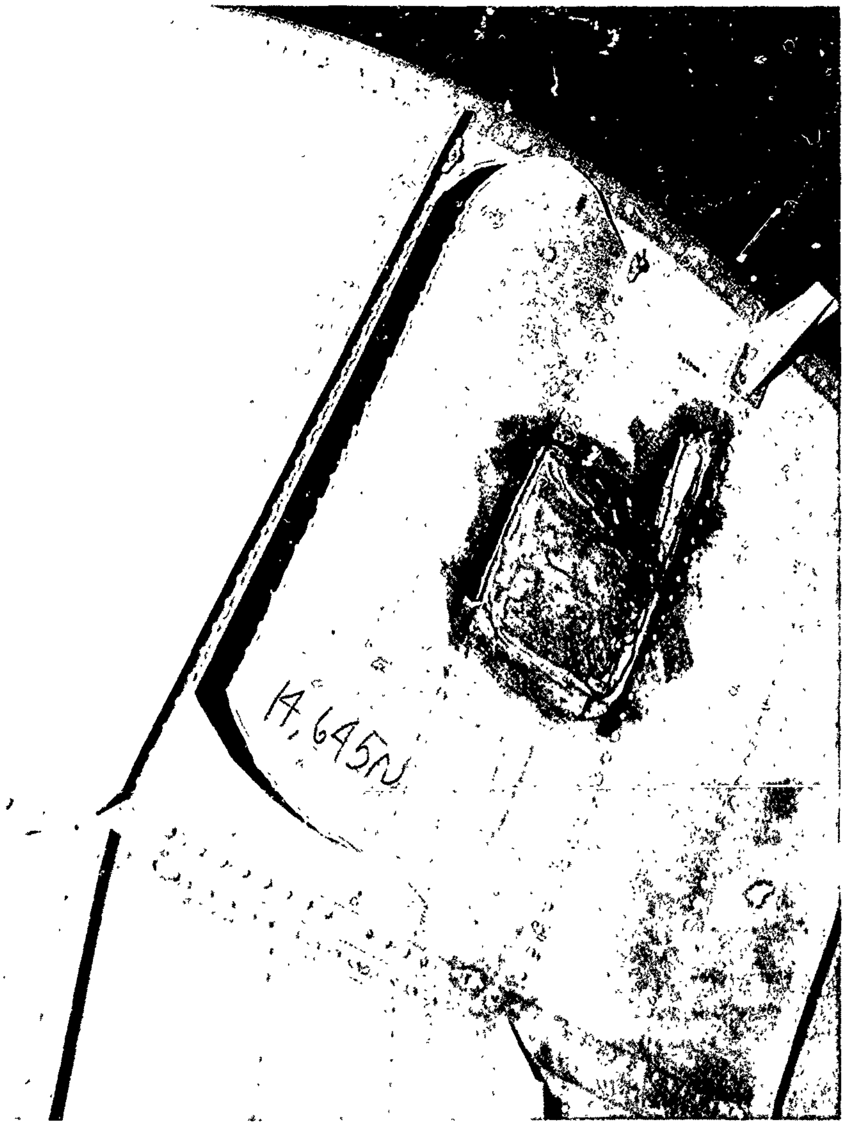


Fig. 25

Fail-Safe Failure of Flat Pressure
Bulkhead C-130A

This failure occurred after 14,645 cycles without warning and resulted in loss of pressure though flight load strength was not affected.

Fig. 26

Non-Fail-Safe Failure of C-130A Fore Body

View is of fuselage looking down and forward. Failure shown essentially severed nose from after body. Cracks started in upper body skin and butt-type manufacturing joint angles and propagated around nose. No crack stoppers had been provided in this joint in the original design. All C-130's have been modified to eliminate future failures of this type in this area.

TECHNIQUES OF TEMPERATURE SIMULATION IN FULL SCALE FATIGUE TESTING

By

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ABSTRACT

This paper presents facts and figures on test equipment, methods and techniques developed to simulate temperature conditions in full scale structure fatigue testing at Convair San Diego.

Typical flight temperature requirements are shown, problems relating to their simulation on full scale test specimens are brought out, and a method for the thermal simulation is presented.

Subjects given special attention are as follows:

1. Extending the state of the art in the use of infrared lamps.

2. New specimen coatings to improve absorptivity.

3. Use of electric arc heaters to simulate high density heat requirements.

4. Forced cooling in thermal simulating.

The equipment and techniques described will simulate mach 5 stagnation conditions for large specimen areas (infrared lamps) and mach 9 stagnation condition areas for smaller areas such as nose cones and leading edges (plasma jet). Thermal equilibrium conditions exceeding

mach 9 at 100,000 feet can be simulated.

INTRODUCTION

Much has been written and said about the myriad of problems related to structural fatigue; also to temperature problems that we encounter in designing, building, and operating our flight vehicles of today. Rather than to repeat what has already been disclosed in this area, the subject of thermal simulation in full scale fatigue testing is highlighted. However, since this topic is closely related to other test considerations, they too may be touched on briefly from time to time in developing this story. The most well known and publicized items in this field will be briefly reviewed. More time will be spent on those items that may help advance the state of the art. Even the subject of thermal simulation is broad. Since the author is directly concerned with the testing end of this business, and test engineers are usually provided with temperature, load, time histories which they apply to structure, that area concerned with thermal test facilities and equipment will be treated.

The ultimate goal for the test engineer is to be able to apply actual load and temperature spectrums in true time to any product. This must be accomplished simultaneously, continuously, automatically, and accurately in conducting life tests to prove and improve structure. Data must also be recorded and processed in the same fashion. Ground tests run in this fashion would provide the answers to many of our structural design problems without the danger of complete loss of extremely costly specimens. Also, information from such tests can be obtained and used before it's too late to change a structure.

Many compromises are made in view of technical and financial considerations but great strides have been made in this activity by our industry. This paper presents some advances made in this direction.

REASONS FOR HIGH TEMPERATURE STRUCTURES TESTING

We've learned from past experience that the only practical method for determining the structural soundness of an airplane is by conducting rigorous ground tests in close simulation of actual operating conditions. (Flight testing is more accurate but too costly and too late.) We must continue to follow this practice. Up to now, we've been concerned mostly with loads, but recently heat has come into the picture. Along with the load conditions encountered in the past, we must now find out under controlled conditions how heating and cooling affects our structure.

Heat sources, such as aerodynamic heating, radiation from the sun, and engine heat, encountered by our present flight vehicles present a complicated system of thermal gradients. Add stresses caused by various combinations of heating and loading; combine the variable reduction in material properties caused by heat; toss in rapid cooling if you may; and you have a structural design problem for which analytical solutions are virtually impossible.

In a recent test we discovered that when a critical part of a missile was subjected to repeated loads and programmed heat, the measured stresses

were increased by a factor of 3 during a rapid 300°F temperature rise.

Where fatigue is concerned, we must be able to answer such questions as:

1. Does the effect of creep outmode fatigue?

2. Does metal fatigue increase the creep rate?

Since these are dependent on time, temperature, and stress, along with other variables, such as, discontinuities in the built up structure, only tests of full scale specimens subjected to load, time, temperature spectrums will give us the answers to these questions.

Since the designers must know the combined effects of heating and loading on a structure before a flight article is available, thermal simulation in ground tests is mandatory.

GENERAL FACILITY DISCUSSION AND GROUND RULES

The performance of aircraft, missiles, and space vehicles is increasing at a rate which is pushing the state of the art for simulation of load and heat in structural tests.

Stagnation temperatures of 6000°F, and heat fluxes of 2000 $BTU/FT^2/sec.$ and loads of 100g have been mentioned in connection with some projects. (1)

Since these requirements along with the accompanying time history exceed the capabilities of most present test apparatus, new equipment and facilities are needed for this simulation.

In order to provide proper facilities for this work, we must continue to study testing problems, develop and evaluate facilities and equipment, and explore new test methods and techniques. (2)

There is no universal method for thermal simulation in testing. General methods of heating are well known. In review these are conduction, convection, induction, and radiation. They all have a place and each of us has his reasons for choosing one heating approach or another. (3)

Individual orgainizations have different requirements, problems and approaches to thermal simulation in full scale fatigue testing, but one way to get indoctrinated in this business is to have a program and a schedule. In this way some problems have built in answers. This is what happened at Convair San Diego:

Test Requirements (simplified)

1. Simulate aerodynamic heat and load on full scale airplane fuel tanks while filled with JP-4 fuel under pressure.

2. Fuel cooling during test was also required.

3. A typical test spectrum for one of these tanks is as follows:

Test No.	Loading Condition	Skin Temp. (°F)	Fuel Temp. (°F)	Cycles
1.S.	Limit Load	Amb.	Amb.	1 Static Test
2.S	"	-65	-65	1
3.S	"	70 to 160	70 $\pm$ 20	1
4.S	5.33 X Limit Load	70 to 216	40 to 110	1
1	0 - 4.7g	Amb.	Amb.	2400 Fatigue Test
2	"	-65	-65	500
3	"	160	70 $\pm$ 20	1200
4	0 - 3.6g	70 to 216	40 to 110	240
5	0 - 4.7g	Amb.	Amb.	2400
6	"	-65	-65	500
7	"	70 to 160	40 to 110	1200
8	"	Amb.	Amb.	2400
9	"	160	70 $\pm$ 20	1200
10	0 - 2.25g	Amb.	Amb.	75000
11	0 - 2.25g	-65	-65	75000
12	0 - 4.7g	Amb.	Amb.	To Failure

Notes:

1. The pressure in the fuel tank was cycled from 0 - 15.5 psi for each load cycle.

2. Temperature was programmed during test No. 3.S, 4.S, 4, and 7.

3. Temperature gradients through the skin were maintained in other conditions as noted in table.

A temporary test site in a remote area was developed to conduct these tests because of the fire hazard and the long lead time required to set up such a facility in more congested areas. Portable equipment including power transformers were rented. Liquid CO_2 provided the fuel cooling and pressure. Three saturable core reactors were used to program heat through infrared lamp banks to the specimens.

Figure 1 shows this site.

Figure 2 shows one of the specimens.

Setting up this small temporary test site gave an indication of some of the major problems in setting up thermal simulation test sites:

1. Special provisions must usually be made for providing the large quantities of electrical power required for such work.

2. Insurance underwriters must be consulted and satisfied with special facility designs.

3. Providing for thermal simulation costs lots of money.

4. Long lead times are required in getting equipment. Many items of equipment are not standard and must be developed.

5. Off site locations for these tests are a strong possibility along with the problem of providing services (power, gas, water, guards, etc.) especially when full scale specimens are tested while containing fuel under pressure and temperature.

From programs such as this and other extensive studies, general ground rules for a high temperature facility were evolved as follows:

1. Provide a facility to be used for the structural testing of missiles and aircraft components under simulated aerodynamic heat and load conditions.

2. Tests of a hazardous nature such as the heating and cooling in conjunction with loading of pressurized tanks filled with fuel must be performed safely in this facility.

3. Heat and load must be programmed into the specimen simultaneously, continuously and in simulation of true flight history.

4. Test data such as temperature, pressure, deflection and strain must be recorded instantaneously and processed automatically.

5. The construction of the facility to be such that for any given year a certain overall test capability exists:

   A. To reduce cost of funding in a given year.

   B. To take advantage of improvements in the state of the art in subsequent years.

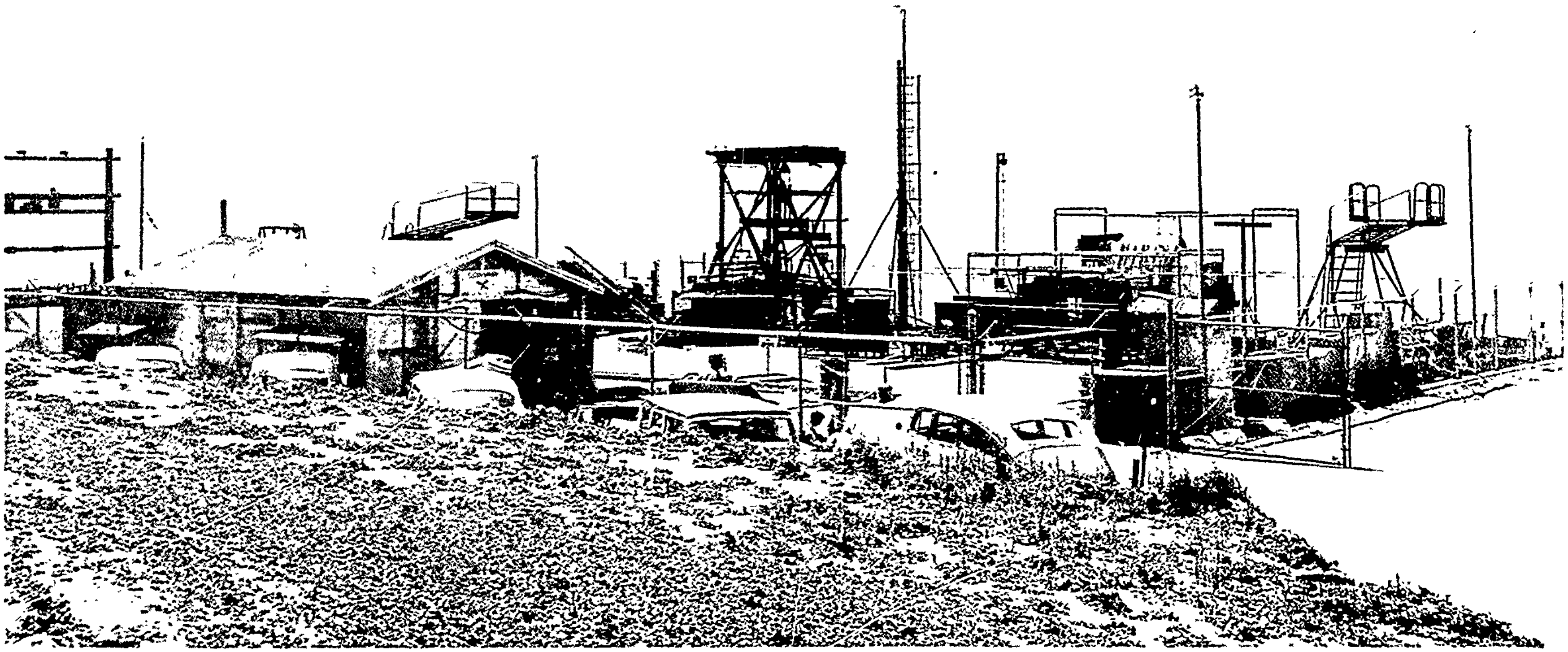

Figure 1 - Temporary High Temperature Structures Test Facility

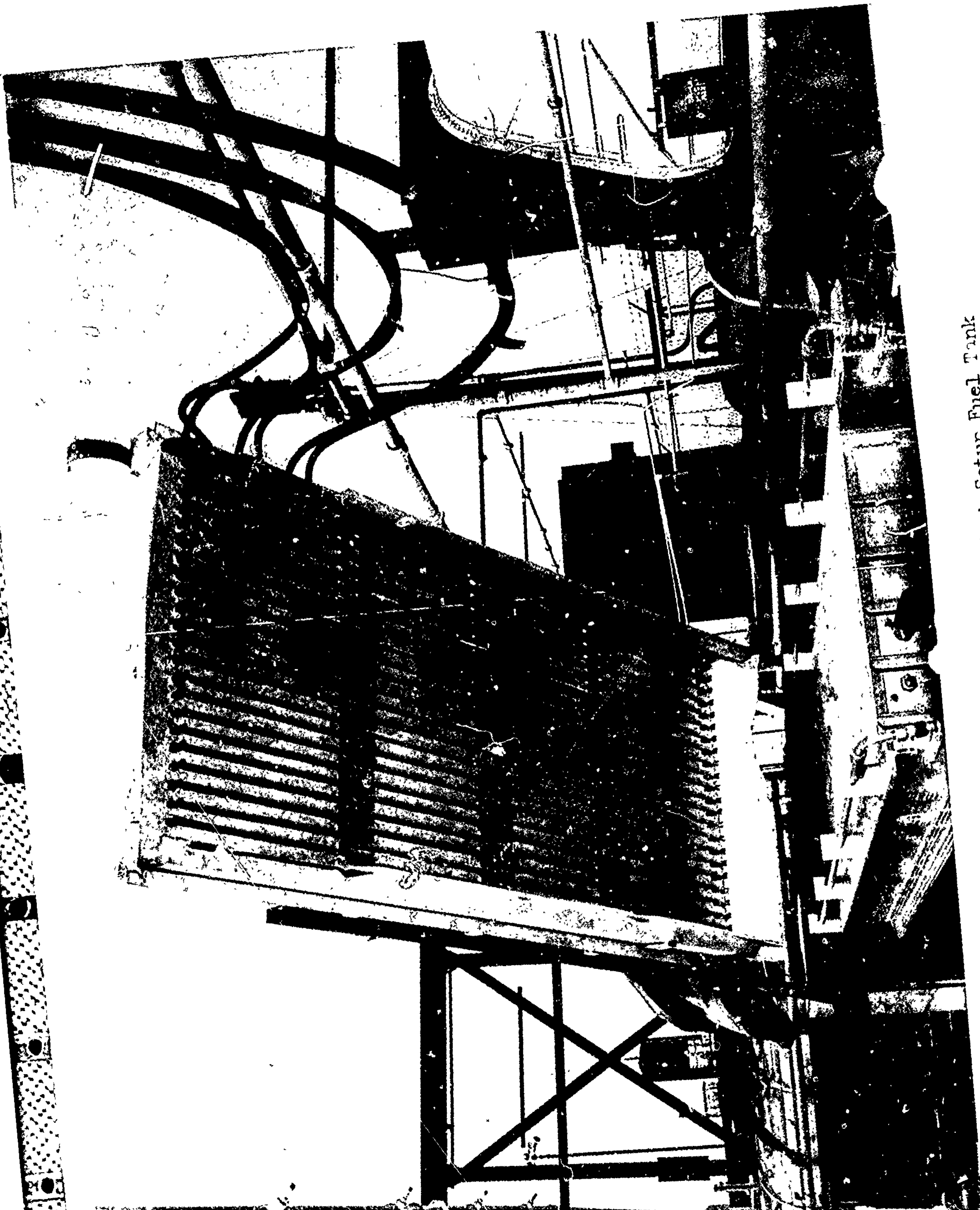

Figure 2 - Test Setup Fuel Tank

6. Design to allow for easy expansion. All equipment must be compatible.

7. Be prepared to put on any combination of heat and load on structure as required (primary heat source being limiting factor).

8. Provide temperature controller and programmer that can be used to control batteries of heaters surrounding the specimen.

9. Infrared lamp heaters will be used whenever possible. These to be built up around specimen as required.

10. Supplementary heaters to be developed to heat those areas beyond the range of infrared lamps.

11. Specimens of varying sizes up to whole vehicles must be handled in this facility.

12. Provide for programmed cooling.

FACILITY COSTS

Once the facility requirements are known, the first obstacles encountered are:

1. <u>What should a basic facility consist of?</u>

A general approach for deciding what is necessary in providing a thermal simulation capability is to take a look at your needs and your budget, and then decide on a basic building block which will give an overall capability for solving today's problems today and to which future building blocks can be added later.

The basic building block might consist of electrical power and multiple channels of temperature and load programmers, controllers, and recorders. Actual heaters and special items of equipment are acquired as needed, and are tailored to the individual job. Specimens are enclosed with batteries of heaters. The heaters are divided into groups proportional to the calculated heat required. Each of the heaters is connected to its own control channel and programmed to follow a curve which represents the heat variation for that section over the duration of the flight mission. If adequate test areas and equipment buildings are not already available and suitable, they should be added. In this system a test capability exists from the time the first dollar is spent and the latest advances in the state of the art can be utilized when adding to the facility.

For the sake of economy test programs may have to be limited in size to fit the capacity of the facility. Consideration should also be given to tests of components instead of the whole article; one half of an airplane instead of the whole; and overloading transformers for short periods of time. Existing equipment and facilities should be included in this basic block if at all possible.

2. <u>What does a basic facility cost?</u>

Many of us have done our own research into devising, building and using various types of equipment. Also many of us have been fortunate enough to benefit by the extensive work that the Air Force, Navy, Army, other government agencies and private industry have financed in this field, and have been able to buy elaborate control and programming equipment at a fraction of the cost of development. Even so, a great deal of money is required for this work.

Some basic facility costs are listed below:

1. Electrical power - substation and feeder line - $35/KVA.

2. 450 KVA ignitron power controllers - $3,750 each.

3. Computer programmer - $6,250 each.

4. Recorder - 400 channel tape channel unit - $800/channel.

5. Power cables, instrumentation wire and heaters - $20/KVA.

6. High bay load reacting structure with reinforced tiedowns - $28/sq. ft.

7. Load programmer - $5,000/channel.

8. Electric power bill (12,000 KVA) - $6,000/month.

9. Related equipment rentals:

   IBM 727 tape transport - $560/month.

   IBM 526 summary punch - $95/month.

   IBM 704 - $80/hour.

For pricing out a facility building block let's take a 6,000 KVA power base, approximately the power required for the mach 3 simulation of a manned interceptor, and see what it would cost to conduct a full scale fatigue test on such a vehicle.

Specimen cost and labor is omitted.

1.	Power - 6,000 x $35	$210,000
2.	Ignitron controllers 12 x $3,750	45,000
3.	Computer programmers 12 x $6,250	75,000
4.	Automatic recorder 400 x $800	320,000
5.	Heaters, thermocouple wire, cable etc. 6,000 x $20	120,000

6. Test building 100' x 100' x $30 $300,000

7. Load programmer 36 x $5,000 180,000

8. Electrical power bill for one year 36,000

9. Special equipment rentals for one year
IBM equipment 8,000

10. 704 computer time $80/hr. when used for sorting and processing test data - assume 2 hrs/week. 8,320

Total $1,302,220

Any special equipment is extra, such as, a 40 ton CO_2 system ($40,000) and CO_2 sells at $80/ton.

ACTUAL FACILITY FROM BUILDING BLOCK APPROACH

At Convair, San Diego, the building block approach has been used over the last couple of years in expanding existing facilities to provide a thermal simulation capability for full scale testing as follows:

1. General

Known equipment was surveyed and in many cases equipment specifications were written around existing apparatus.

Example:

Research Inc. temperature programmer controllers

In other cases where existing equipment did not meet our requirements, specifications were written for new equipment.

Example:

Convair, San Diego Computer Lab Data Acquisition and Interpretation System.

2. Size

The Laboratory consists of buildings and test equipment for accommodating large test specimens of the following combinations:

A. Two airplanes with wing spans or fuselage length of 90 ft.

B. One airplane and one large 120 ft. missile.

C. Two large 120 ft. missiles.

3. Brick and Mortar

A two story air conditioned block house is provided for housing personnel and test equipment.

Test buildings are reinforced concrete floored, load reacting structures.

4. Power

A 600 volt, 3,000 KVA system is presently installed, with overload capabilities to 6,000 KVA for 5 minute periods.

5. Power Controllers

12 sets of ignitron power controllers allow simultaneous and independent control of 12 separate areas. These controllers contain two ignitron tubes each, wired in universe parallel to give a single phase A.C. output with voltage controlled by varying the firing angle within each cycle. Ignitron controls shown in Figure 3 were chosen because of their rapid speed of response.

6. Computer Programmers

The nerve center is the programmer. Essentially, this computer delivers to each control device in "real time" the load, temperature, heat flux or other variable predetermined for it.

The controllers are coupled to programmers which contain analog computers with A.C. outputs. Two control consoles allow programming from one to twelve channels each. (See Figure 4.) Programming can be accomplished as follows:

A. $h(T_{aw} - T_s) - BT_s^4 - \frac{NYEI}{K}$ = error signal.

B. $h(T_{aw} - T_s) - BT_s^4 - C\frac{dT_s}{dt}$ = error signal.

C. Heat flow versus time.

D. Temperature versus time.

E. Thermal soak at fixed temperature or heat flow.

The first two modes are variations of the convective heat flow equation, in which h, the heat transfer coefficient, and T_{aw}, adiabatic wall temperature are fed to the computer by curves drawn on input drums. T_s, specimen temperature, is fed back by thermocouple. BT_s^4 is a radiant heat loss. In Mode 1, the desired Q is continuously compared with the actual Q as measured by an efficiency (N) times the power (YEI) being put out by the ignitrons. In Mode 2, the desired Q is compared with actual Q as measured by a mass-specific heat factor (C) times the rate of specimen temperature rise $\left(\frac{dT_s}{dt}\right)$.

Figure 3 - Ignitron Power Controllers

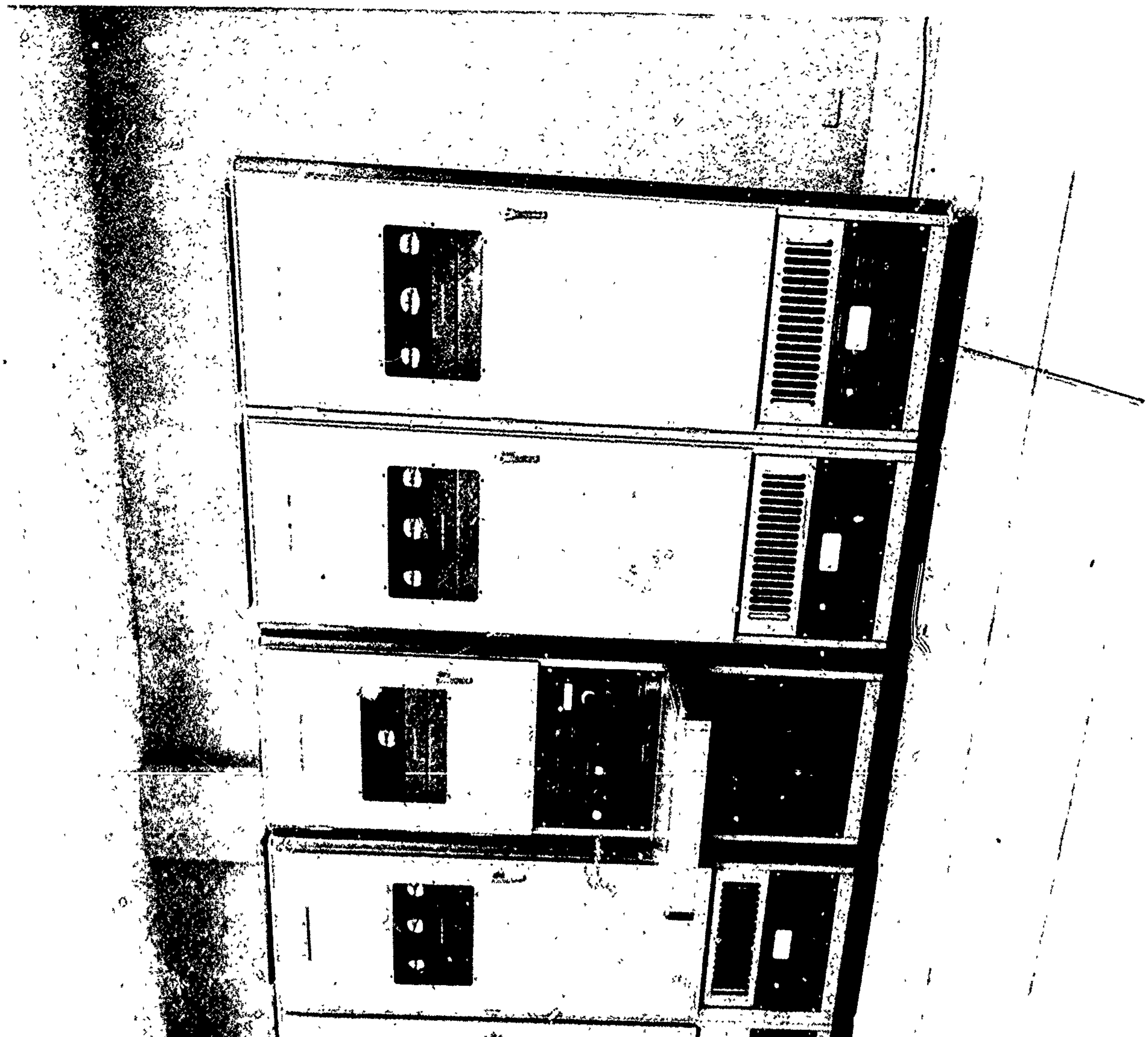

Figure 4 - Temperature Programmer Computer

In Mode 3, heat flow is programmed by a single input curve, and the actual Q is measured as in Mode 1 or 2.

Thermal simulation with the computer control can best be explained by explaining Mode 1 with reference to Figure 5.

A. Incoming power is received by

B. The ignitron power unit which controls the power delivered to the

C. Heaters surrounding a test specimen.

D. A thermocouple located on the specimen measures surface temperature, T_s, which is

E. Recorded on tape in the Data Acquisition center and is also

F. Used in an R.C. network to provide the radiant heat loss correction $BT_s{}^4$

G. T_s is also combined in a summing network with a

H. Signal representing T_{aw}, the adiabatic wall temperature.

I. This combined signal, $(T_{aw} - T_s)$ is combined in a

J. Multiplying circuit with the signal

K. h, the heat transfer coefficient for a particular flight profile.

L. The factor $h(T_{aw} - T_s)$ combined with the radiant heat loss factor $- BT_s{}^4$ represents the power desired which is compared in the computer-controller with

M. A measure of the power being instantaneously delivered.

N. Any unbalance in these two values shows up as an error signal to the controlling equipment which drives to reduce the difference to zero.

7. Specimen Cooling

A 40 ton carbon dioxide storage tank is installed for applying liquid or gaseous carbon dioxide for specimen cooling; also for inerting area around specimen containing fuel. (See Figure 6.)

8. Data Recorder

Tests are of no value unless data is recorded and put into a usable form in a reasonable length of time. This is how this can be accomplished. A data acquisition system was designed and built under the direction of the Convair Computer Laboratories. It will record signals from 400 end instruments (strain gages, thermo-

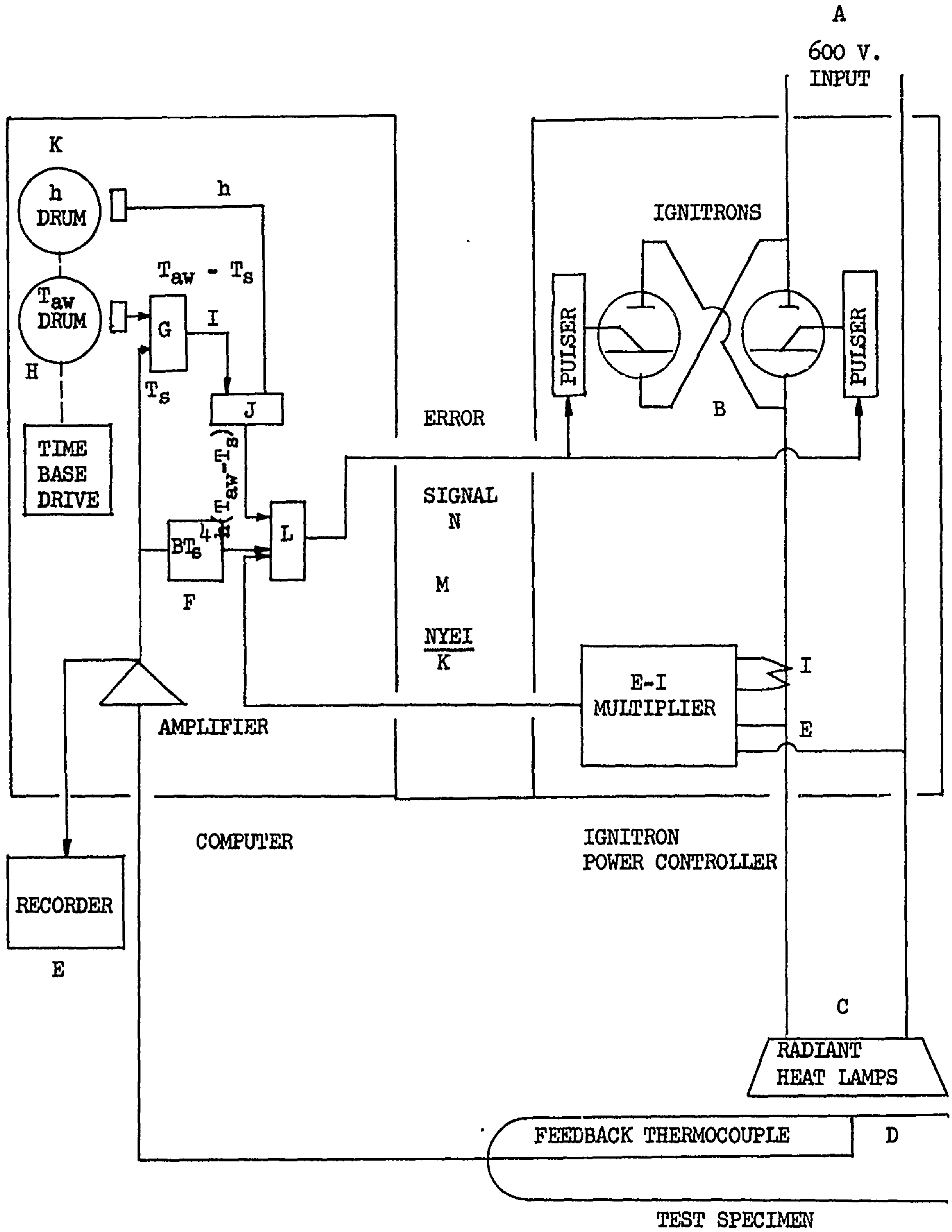

Figure 5 - HEATER PROGRAMMER CONTROLLER SCHEMATIC

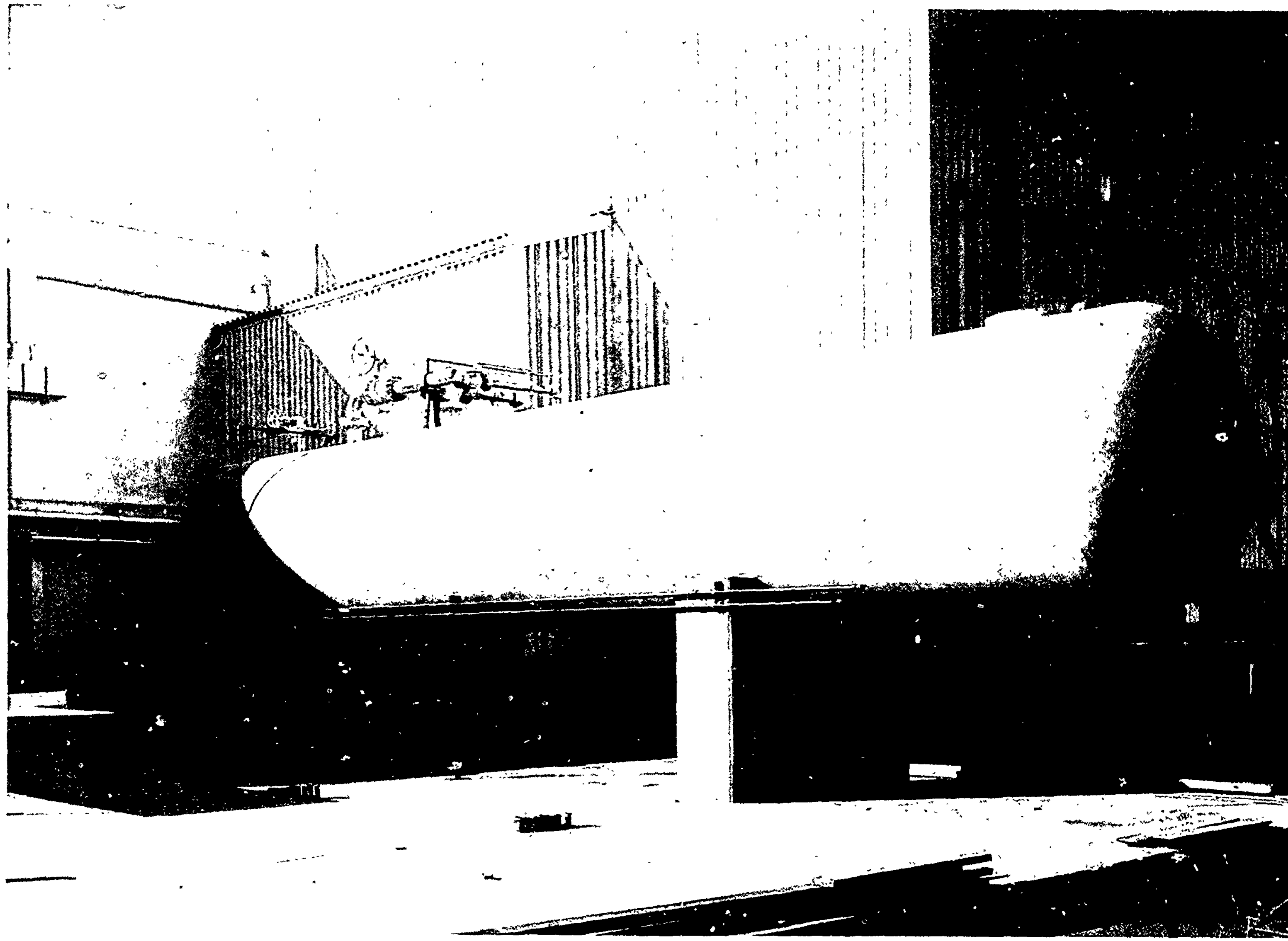

Figure 6 - 40 Ton Liquid Carbon Dioxide Storage Tank

couples, deflection transducers, accelerometers, pressure pickups) at a maximum rate of 1600 channels per second. The signals are digitized, stored on magnetic tape, then fed to an IBM 704 computer for data sorting and processing. Eighteen channels are monitored during test, by continuous analog plots on x-y plotters. (See Figure 9.) Figure 7 shows the signal conditioner. Figure 8 shows the tape recorder and Figure 9 shows the x-y plotters.

9. Load Programmer

A load programmer is presently being built that will operate on the same time base as the temperature computer programmer.

10. Safety Provisions

To offset any additional hazards in tests involving fuels, protection by inert atmosphere is provided. This consists of closing off the immediate area surrounding the test specimen by reasonably gas-tight curtains or sheet metal curtain walls and purging. Carbon dioxide, the most probable purging gas, is approximately one and one half times as heavy as air. Once the test cell has been purged and filled with CO_2, a very small continuing stream will keep the specimen in an inert atmosphere. Pans and low dikes under the specimen are used to prevent spilled fuels from flowing out of the purged areas. CO_2 is also used as the pressurizing media for fuel tanks. A 40 ton tank provides the CO_2 reservoir. Observation of the specimen is by closed circuit T.V. CO_2 analyzers are used for detecting and monitoring concentrations of CO_2 inside and around the test cell. While under purge conditions, no fire is anticipated; however a water deluge system is available as is other fire fighting equipment. Drain trenches running from the test area to a 35,000 gallon fuel sump are also provided to handle large quantities of fuel or simulated fuel in case of a major specimen failure when full of fuel.

The overall facility is shown in Figure 10.

SPECIAL PROBLEMS

Once the basic facility is established, problems presented in thermal simulation are those of providing the particular heater or cooler that can be controlled from the basic programmer and that will provide the proper input to the specimen. These heating devices along with related reflectors and specimen coatings must be compatible with loading and instrumentation equipment to allow for true time, heat, load simulation and recording. Also, we must continue to improve on heating efficiences. Results of some studies of these devices are presented next:

1. Typical thermal simulation requirements

Studies have shown that heat inputs must be provided to meet the following requirements:

Vehicle	Max. Heat Flux	Max. Temp.
A. Supersonic jet transport	.5 BTU/FT^2/sec.	600°F

Figure 7 - Signal Conditioner Of Daisy Recorder

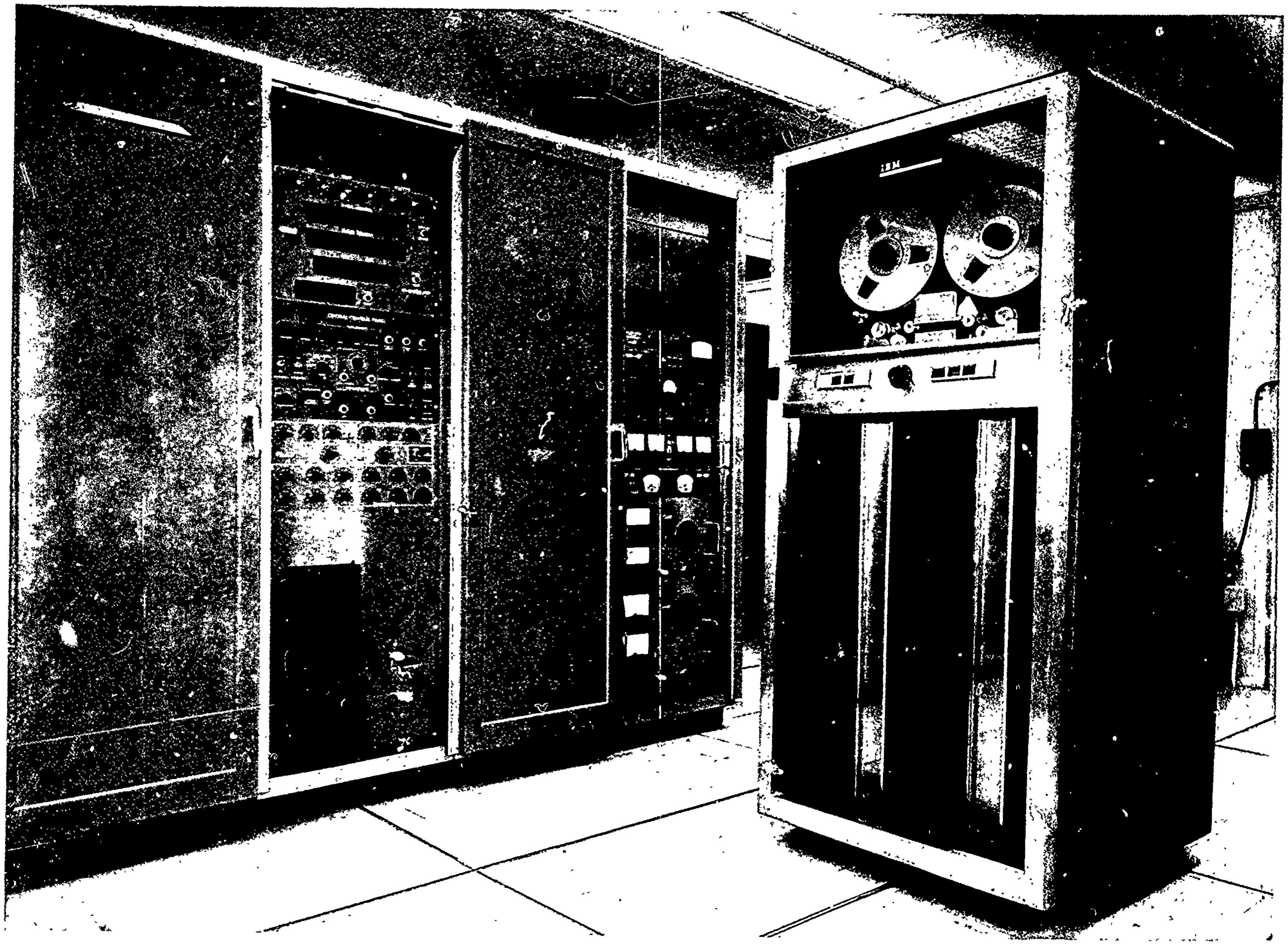

Figure 8 - Information Selector and Tape Recorder

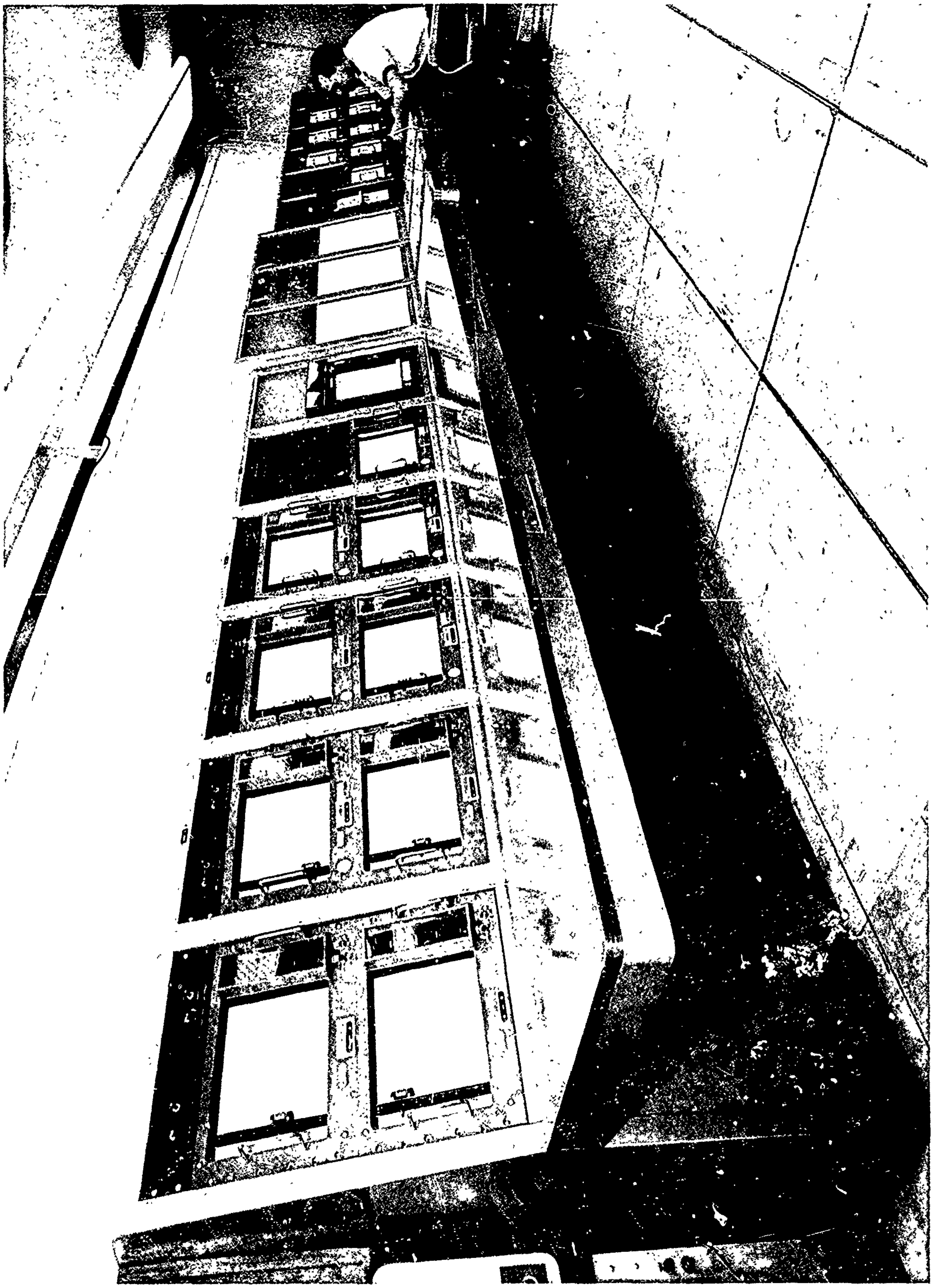

Figure 9 - X-Y-Plotters For Recorder

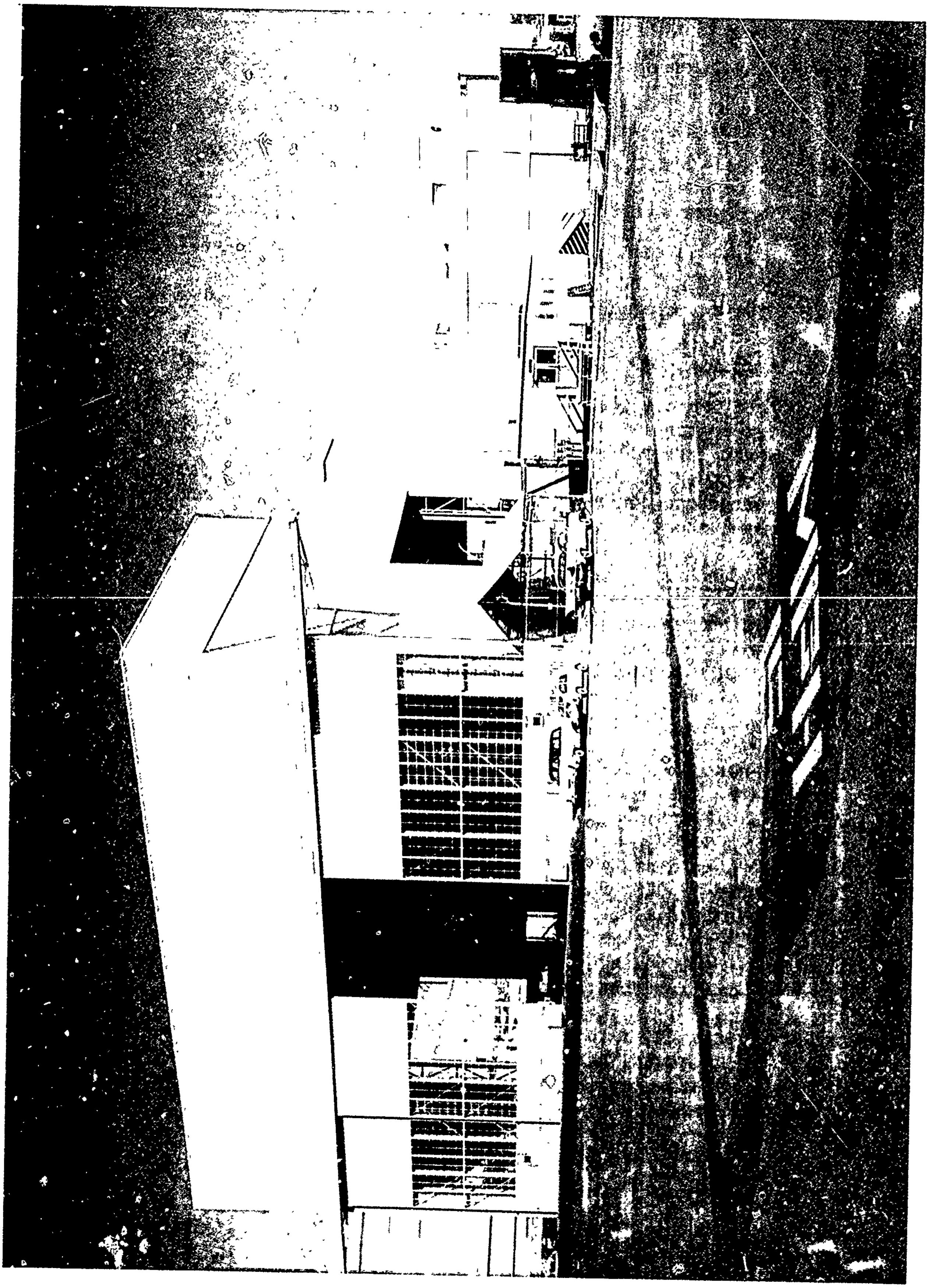

Figure 10 - Convair Hi h Temperature Structures Lab

Vehicle	Max. Heat Flux	Max. Temp.
B. Advanced manned interceptor	6 $BTU/FT^2/sec.$	1000°F
C. Advanced bomber	2.8 $BTU/FT^2/sec.$	1000°F
D. Manned satelites - space vehicles	100 $BTU/FT^2/sec.$	3000°F
E. Unmanned interceptor missiles	300 $BTU/FT^2/sec.$	3700°F
F. ICBM nose cones	2000 $BTU/FT^2/sec.$	6000°F

G. Figure 11 shows a temperature profile for an interceptor missile.

2. Heater Development

Many methods of heating are possible and several methods have been investigated for producing high temperatures. However, this study was centered around quartz-enclosed tungsten filament infrared lamps because of their low thermal inertia, and electric arc heating because of the extremely high energy flux rates obtainable. Also, both can be powered and controlled with our existing equipment and lend themselves to being arranged about a specimen undergoing loading.

A. Infrared Lamp Heaters

The use of commercially available infrared quartz lamps for thermal simulation is well known. The G.E. T3 lamps have a tungsten filament enclosed in a 3/8 inch diameter clear quartz tube containing argon gas. They come in various lengths and power ratings. They may be used with commercially available reflectors that come in many assortments which will heat to about 150 $BTU/FT^2/sec.$

Many times large contoured clad aluminum reflectors can be made and used for lamp holders such as the ones shown in Figures 2 and 25. These are satisfactory for heat rates up to 15 $BTU/FT^2/sec.$ The reflector material limits the temperature range.

Figure 12 shows a high density heater now in use. This heater has no special cooling provisions, used 12 G.E. T3 1000 watt bulbs stacked in two rows on 1/2 inch centers, has a gold plated stainless steel reflector, and produces specimen absorptivities of 160 $BTU/FT^2/sec.$ (See Figure 13.) It is rated for 240 volts and was driven to a little over 500 volts. This heater can only be used at this rate for limited periods because of overheating of reflector and lamp ends.

Three materials - alumina, fire brick, and insulating fire brick, were coated with cobaltic oxide and exposed to this heat flux for 10 seconds. The results are shown in Figure 14. The fire brick and the insulating fire brick melted. The bricks on which the three specimens were placed also melted. The melting temperature for the brick, insulating fire brick, and alumina were 2400°F, 3100°F, and 3600°F.

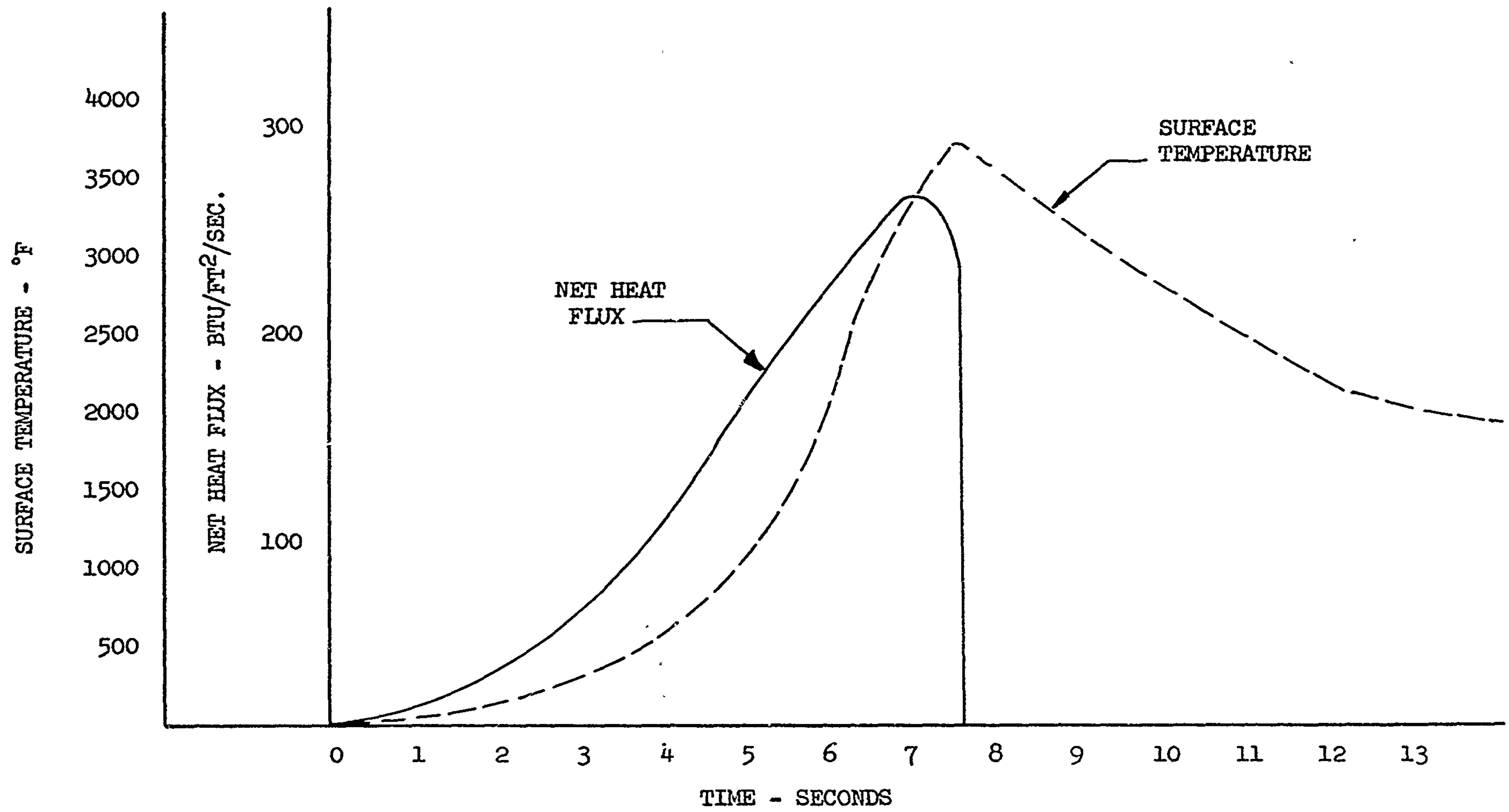

Figure 11 - TYPICAL PARAMETERS FOR HIGH TEMPERATURE STRUCTURAL TEST (INTERCEPTOR MISSILE LEADING EDGE)

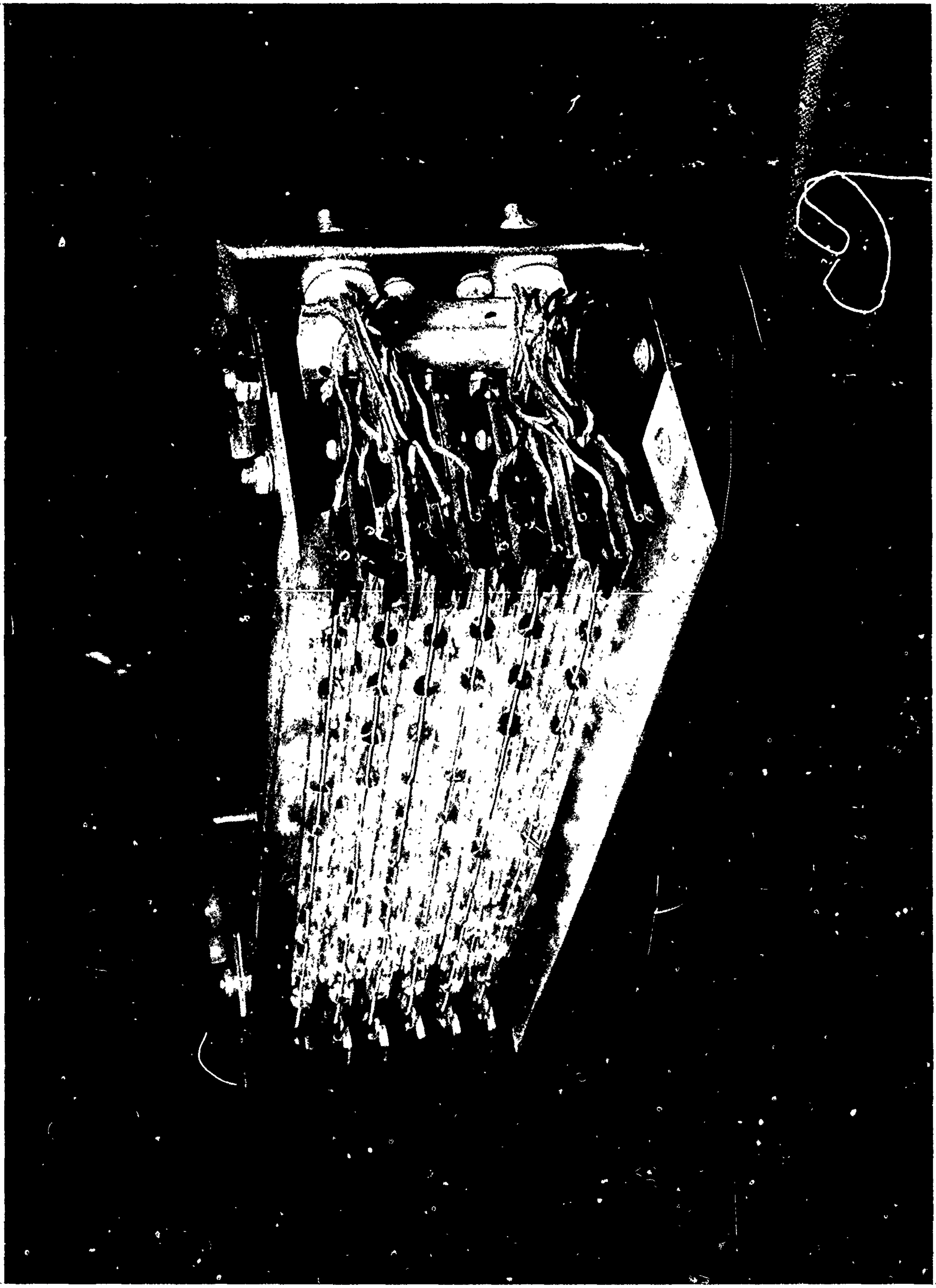

Figure 12 - Infrared Lamp Heater - Hi Density

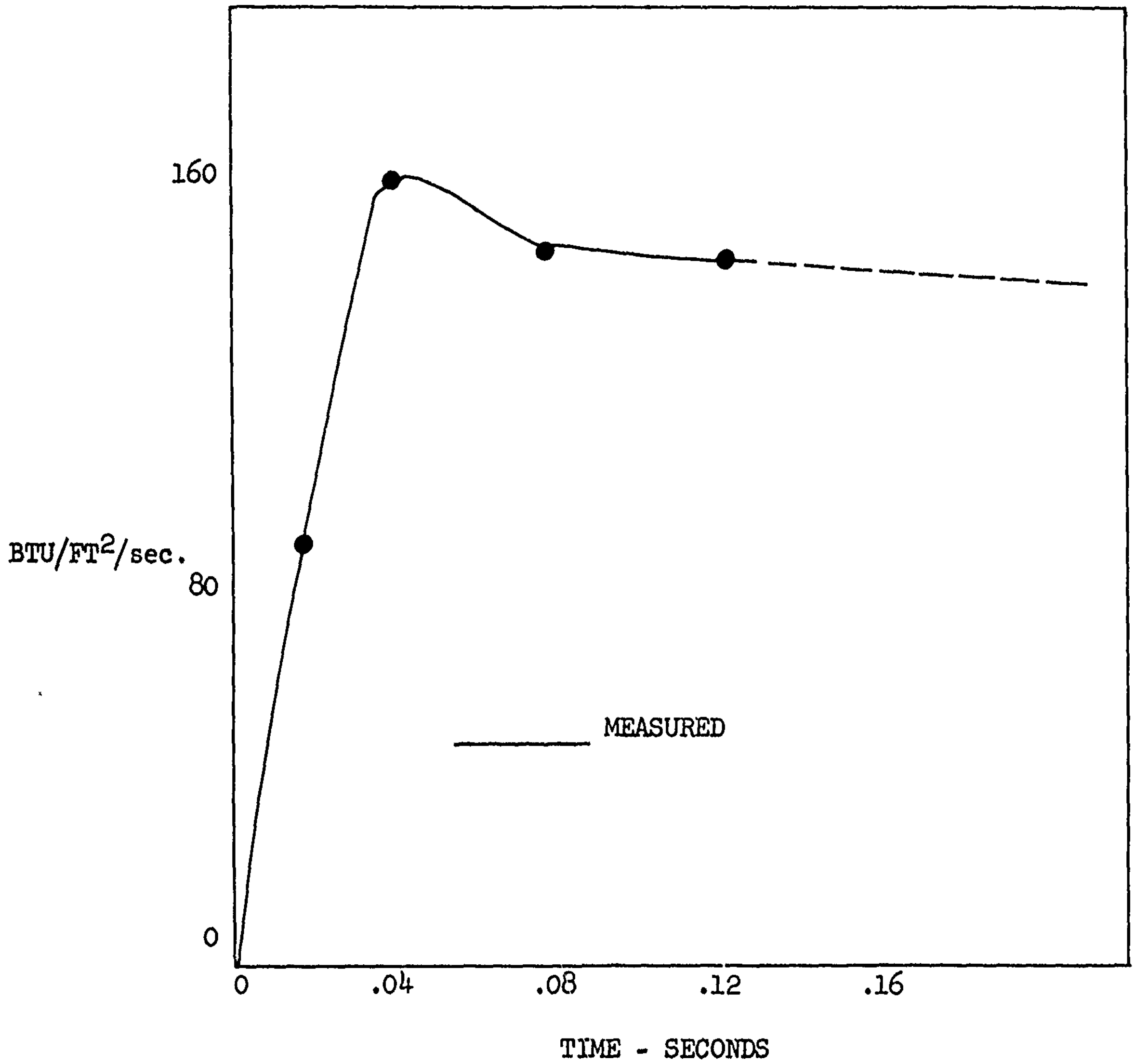

Figure 13 - HEAT FLUX OUTPUT FROM FLAT HIGH DENSITY LAMP BANK

Figure 14 - Specimens Melted By Infrared Lamps

A new high density lamp bank has been built and is being evaluated. It is similar to the first one but has the following changes:

1. It is designed for long usage at high density and high temperature.

2. 2000 watt bulbs are used in place of the 1000 watt bulbs.

3. The reflector and lamps are cooled with air or carbon dioxide. Reflector coatings deteriorate above 1500°F.

4. Lamp ends are cooled by a continuous flow of water. Lamp end seals fail above 650°F.

This heater is expected to produce heat fluxes in excess of 200 BTU/FT^2/sec. for extended periods of time.

Interesting findings

1. When specimens being heated by infrared lamps vaporize, the vapor may coat the heat lamp allowing the quartz to heat up and melt.

2. A CO^2 atmosphere around heat lamps will absorb up to 22% of the heater output.

B. Electric Arcs

Electric arcs including plasma jets have been investigated for use in full scale structural tests where high heat fluxes in excess of those obtainable with infrared lamps are required. Temperatures of 26,000°F and heat fluxes of 2000 BTU's per square foot per second have been claimed.

a. Plasma Jet

Several plasma jets have been built and tested. A D.C. powered plasma jet capable of running continuously for over 2 minutes is shown in Figure 15.

Figure 16 shows the specimen absorptivity and temperature rise when subjected to this heat source for a very short period of time.

Another plasma jet capable of running on A.C. or D.C. power has been designed and built. See Figure 17. The nozzles are removable and sizes up to .75 inches throat diameter may be used. Figure 18 shows this heater in operation. circulating water cools the nozzle while the working fluid (argon, helium, nitrogen, air) cools the back electrode. The back electrode also serves as a regenerator in heating the working fluid before it reaches the arc chamber. Tests are being run on this plasma jet using ignitron power controllers as the power sourve. Air can be introduced through orifices surrounding the tungsten nozzle to reduce the plasma temperature where this is required.

A plasma jet which would be capable of heating a leading edge is being designed.

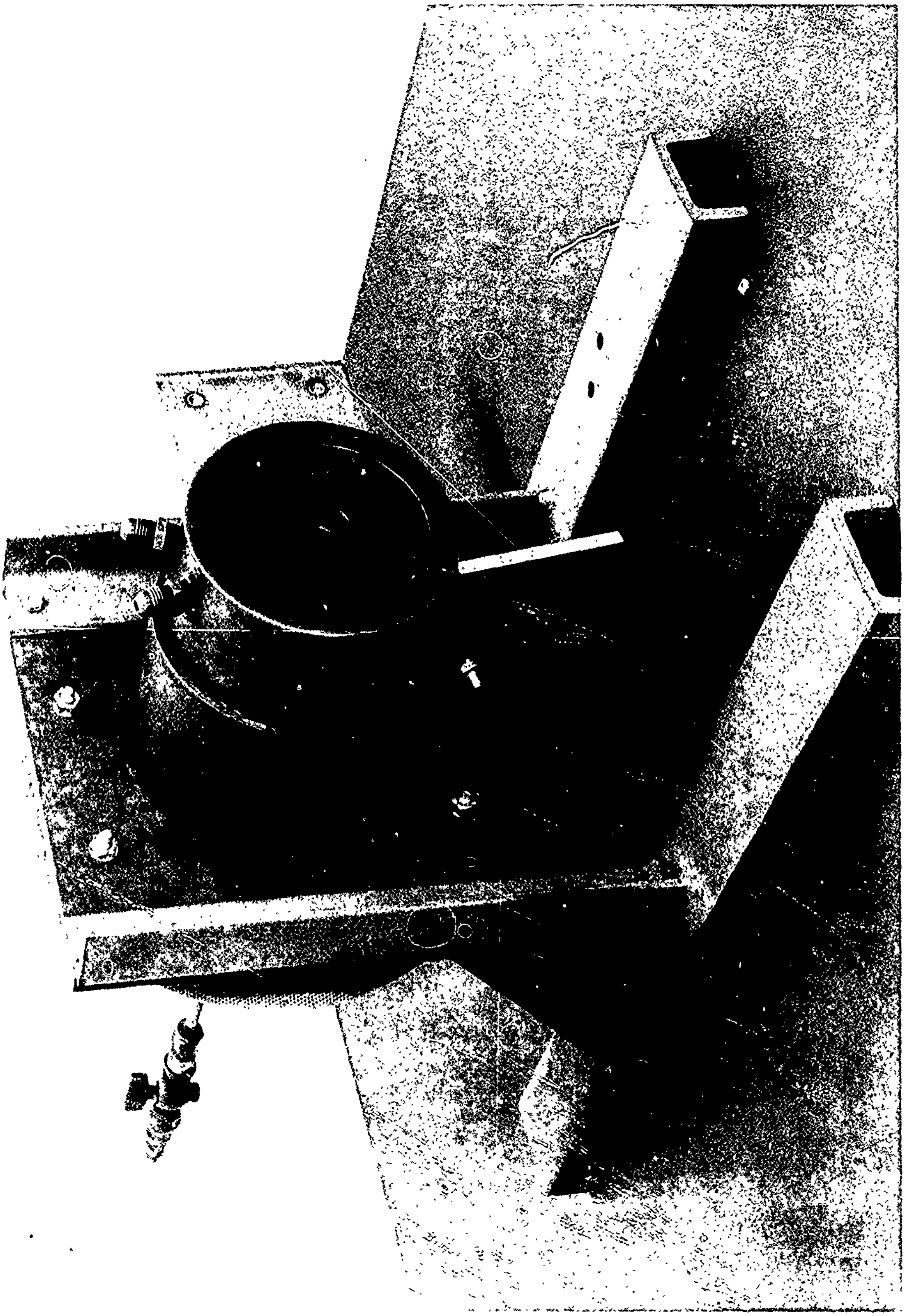

Figure 15 - Plasma Jet

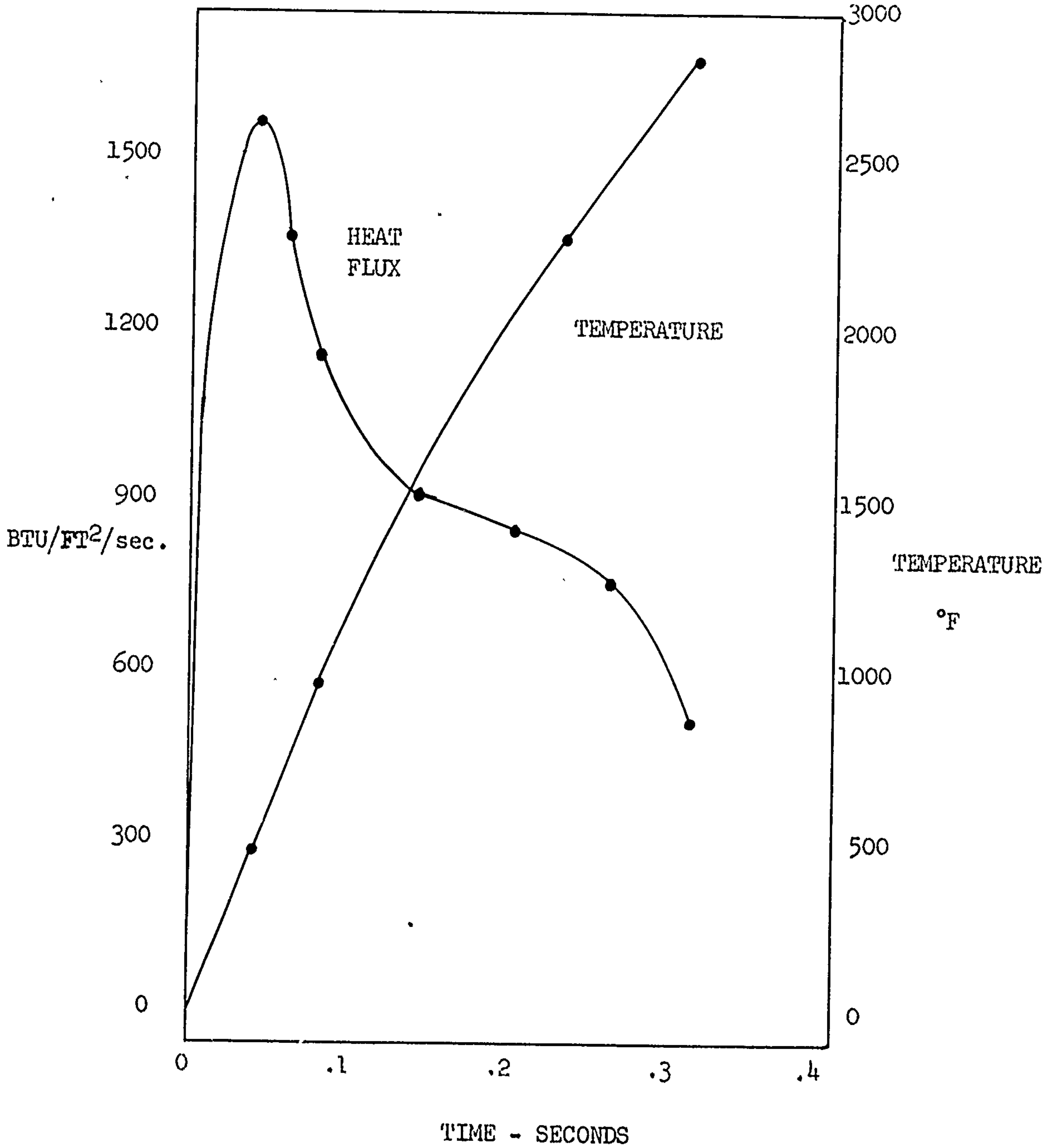

Figure 16 - ABSORPTIVITY AND TEMPERATURE RESPONSE OF MOLYBDENUM SPECIMEN IN PLASMA JET FLAME

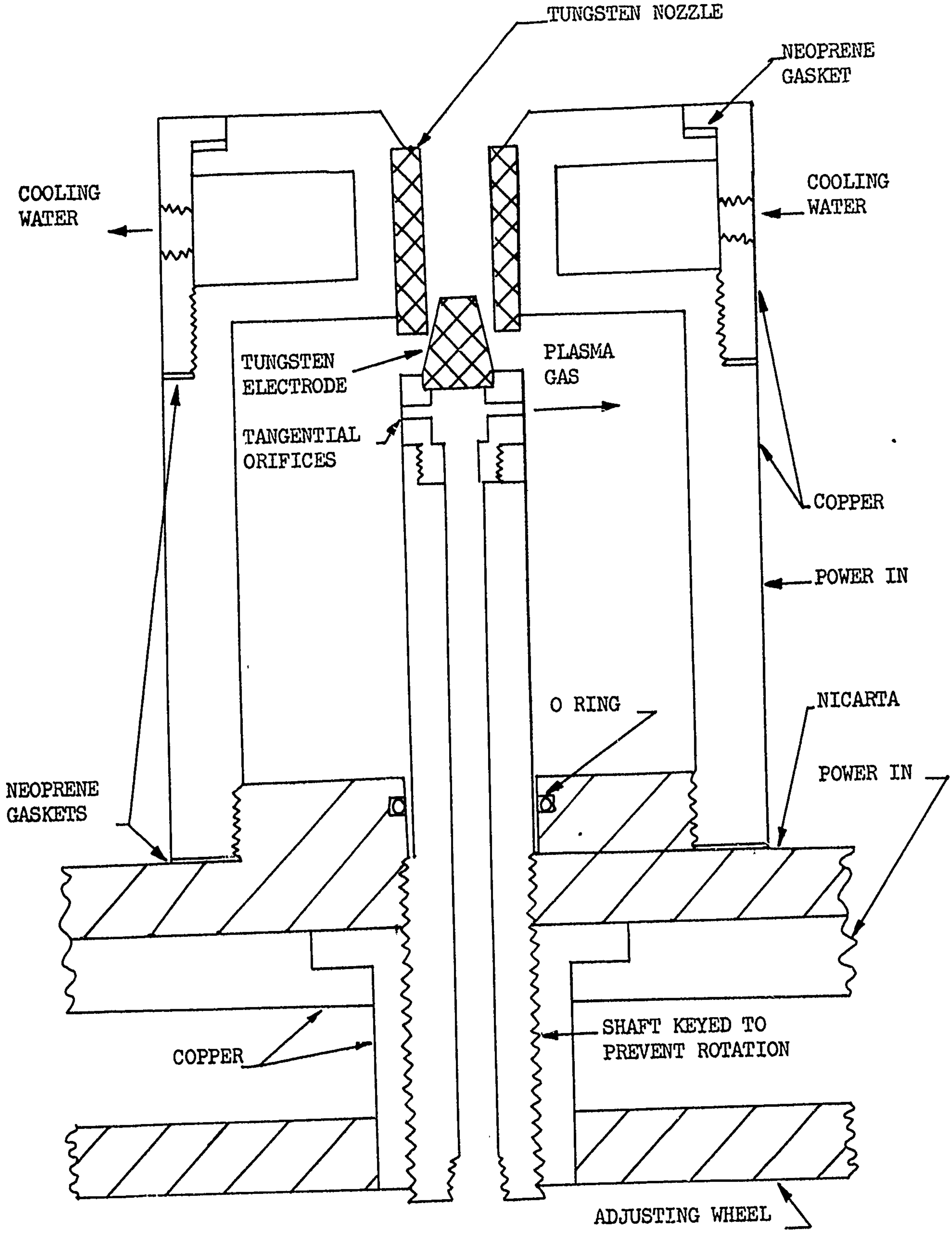

Figure 17 - A.C. - D.C. PLASMA JET CROSS SECTION

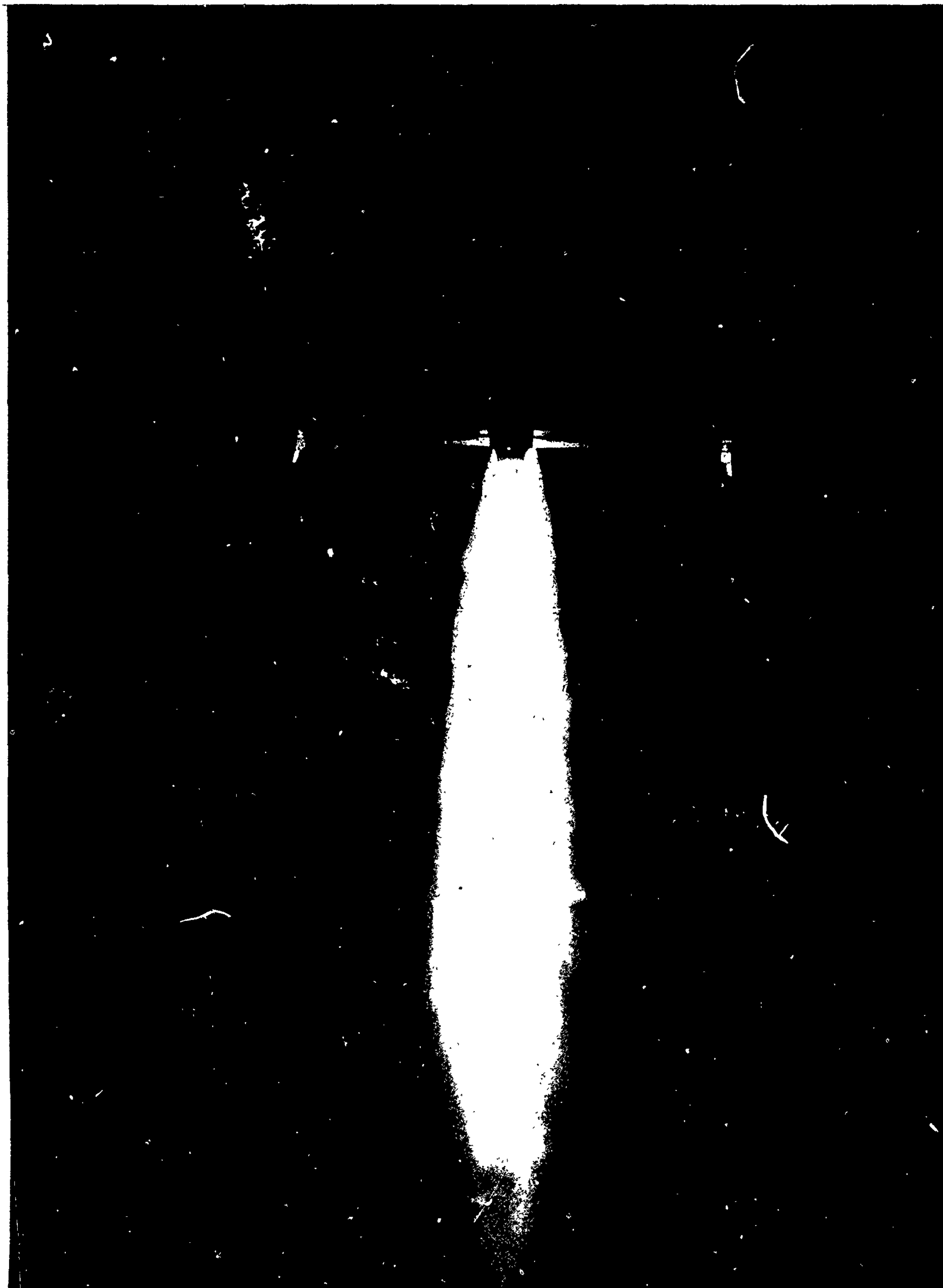

Figure 18 - Plasma Jet Flame

Its operation is essentially the same as other plasma jets except that it is not stationary, but oscillates along a prescribed path. The amplitude and frequency of oscillation are adjustable and a plasma flame can be swept back and forth along the edge of a specimen. Batteries of plasma jets along a leading edge or nose cone will accomplish the same thing.

b. Oscillating Arc

Investigation of other electric arcs for high density heating has been made. One that shows some promise for use in this work is the oscillating arc. Two electrodes were mounted on a platform and the platform was oscillated by attachment to an eccentric wheel as shown in Figure 19. Power was applied to the electrodes and the platform oscillated five times a second. A dense three inch tail flame was produced. One complete cycle of oscillation is shown in Figure 20. A leading edge specimen made of 0.025 stainless steel was put into the inner part of the tail flame. Heat rates of 385°F/sec. were recorded. This device produces a stable, uniform flame.

Electric arc heaters show great promise for use in thermal simulation in full scale tests in those specimen areas whose heat requirements exceed that for infrared heaters. Since these devices will normally be used for temperature in excess of those in the known thermocouple range, the heat flow mode of the programmer controller can be used for controlling input to specimens since it does not require a thermocouple feed back signal.

c. Vacuum Heat Tension Pad

A vacuum type heat tension pad for heating and loading a skin surface simultaneously has been developed and tested. This pad is capable of loading a surface up to 12 psi and heating it to 550°F at rates exceeding 20°F/sec. - 50 BTU/FT^2/sec. The pad is constructed with an arched roof and two sides. Two pipe fittings are inserted in the roof, one for vacuum attachment, and the other for electric power loads. See Figure 21. Inside the pad are four 500 watt G.E. T3 infrared lamps and an Alzac reflector. A 1/2 inch silicone rubber gasket separates the pad from the specimen and serves as a cushion as well as an insulator.

The operation is simple. The pad is placed on a specimen; internal vacuum is applied at the same time the heater is turned on. The maximum load that can be applied is 12.5 psi at which point the pad can be knocked off with a slight eccentric load. At 12 psi the pad is very stable.

To date the gasket material limits the use of the pad to 550°F. As better gasket material becomes available, a high density lamp bank will be incorporated and higher temperature and heat rates can be achieved.

This pad lends itself to use on light gage skins when loading and heating is within its limits, and when built in skin attachments are undesirable.

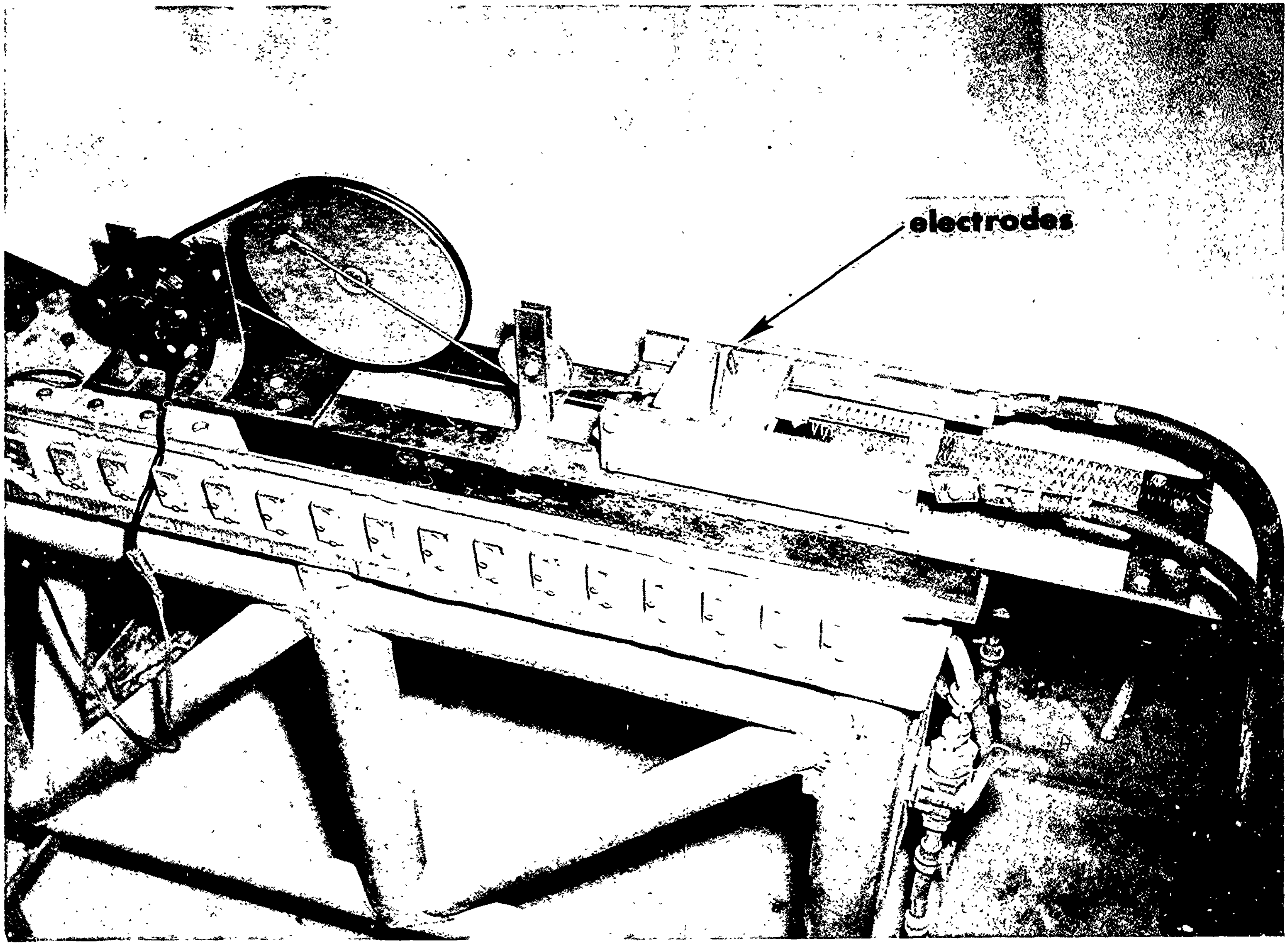

Figure 19 - Oscillatin- El ctric Arc Heater

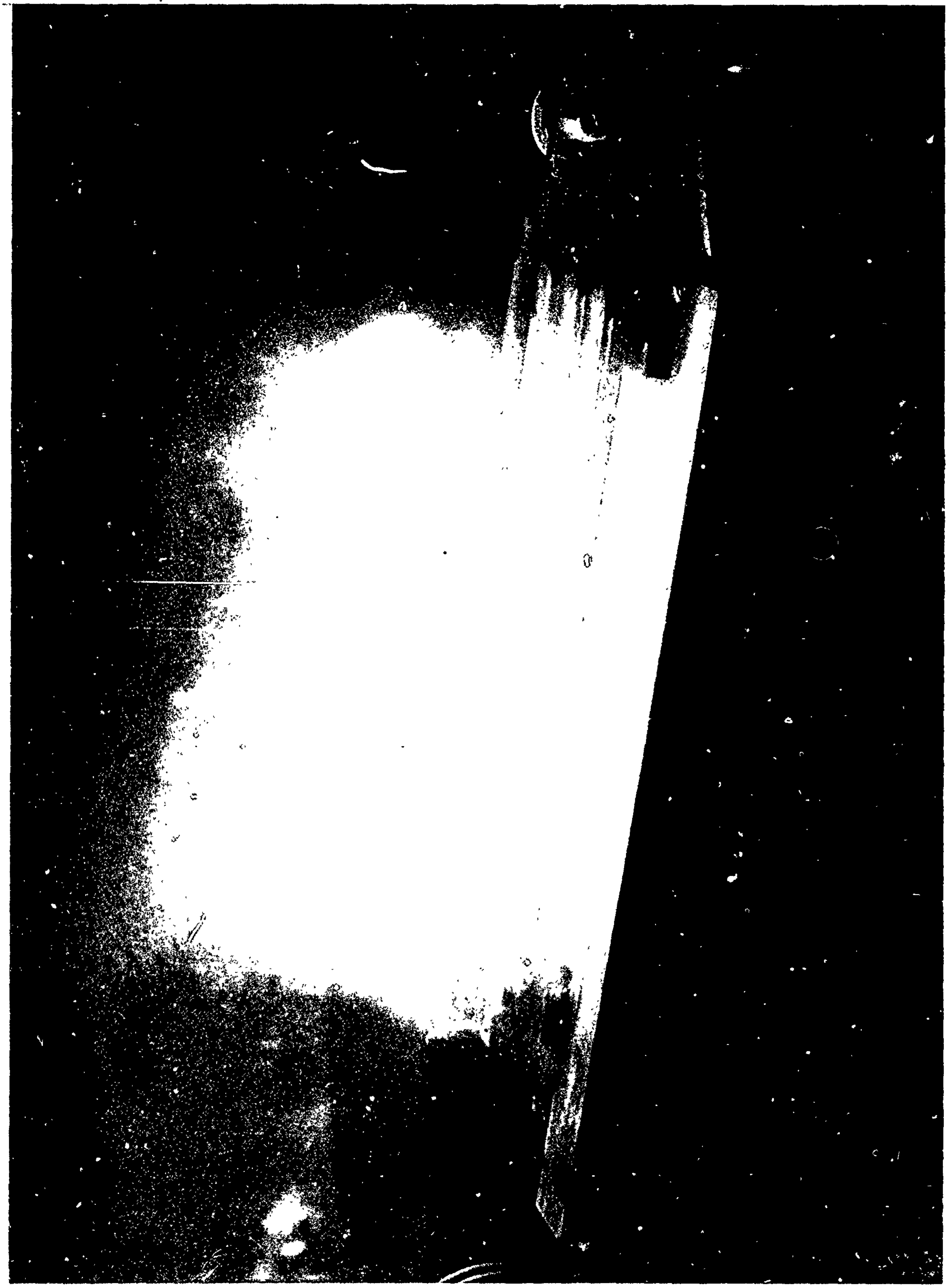

Figure 20 - Oscillating Arc Flame

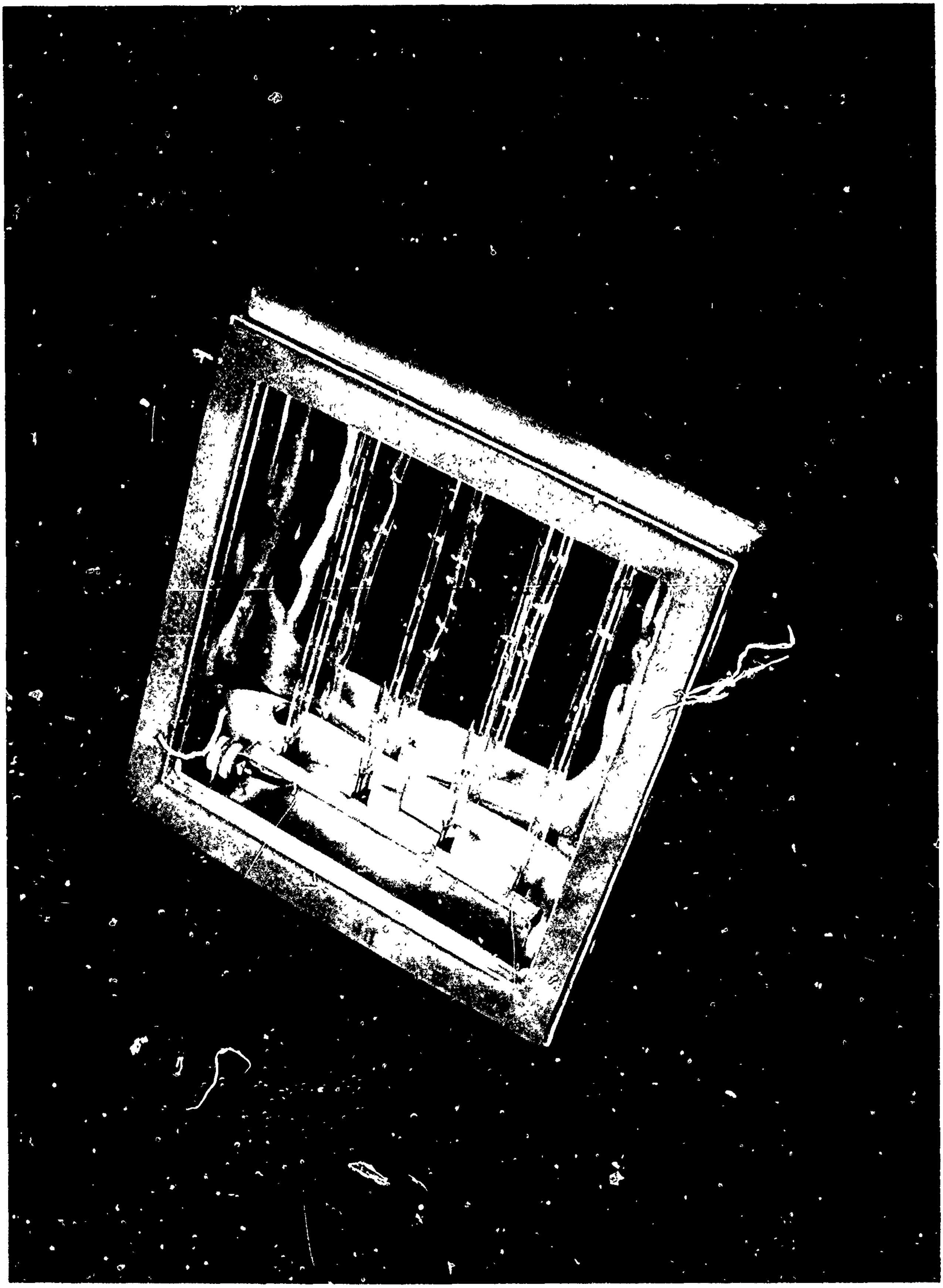

Figure 21 - Vacuum Pad With Infrared Heater

3. Reflectors for Lamp Banks

If efficiency of infrared heaters is to be improved, reflectors must be improved. Tests have shown that low density lamp banks which develop energy fluxes less than 15 BTU/FT^2/sec., can use clad aluminum for reflectors. The clad aluminum has a reflectivity of about 76% at the peak of the infrared lamp spectrum.

Aluminum sheet with a specular finish (Alzac) has a reflectivity of 89% and has been used for high density lamp banks for short periods of time. The low melting temperature of the aluminum (approximately 1100°F) limits the testing time to about 20 seconds for lamp banks producing 150 BTU/FT^2/sec.

The ideal reflector for high density lamp banks must have a high melting temperature, and good strength characteristics. Gold with a reflectivity of 97% is excellent for these requirements, but the cost is prohibitive for a solid gold reflector. A gold coated reflector is feasible and tests have been made to determine the best combination of gold and back-up material.

Gold has been both plated and vacuum metalized on copper and stainless steel. Copper, for the present, has been rejected because it is heavy and makes a cumbersome lamp bank. A thin sheet of stainless steel with a gold coating makes a good reflector.

A comparison was made with two identical 0.04 inch Type 321 stainless steel specimens. One was vacuum metalized with gold, and the other was coated with .0025 inch of an insulating ceramic and then vacuum metalized with gold. Both specimens were exposed to an energy flux of 100 BTU/FT^2/sec. for seven seconds. Figure 22 shows the temperature response on the back side (side not exposed) of each specimen. The ceramic insulates the steel from energy transmitted through the gold. It is desired to keep the back-up material on a reflector as cool as possible to retain its strength and prevent sagging. Our latest gold and ceramic coated stainless steel reflector for high flux rates uses air or gaseous CO_2 for forced cooling of the reflector.

4. Specimen Coatings to Increase Specimen Absorptivity

Specimen coatings must also be improved for the overall efficiency in thermal simulation.

Results of coating investigation are as follows:

Several materials have been investigated to find a suitable coating to increase specimen absorptivity. Those considered with graphite, porcelox, and oxides of iron, nickel, copper, manganese, and cobalt.

The evaluation was based on ease of application to a specimen, the ability to retain properties after continuous exposure to high infrared energy flux, and low reflectivity or high emissivity in the infrared range.

Tests run on black iron oxide (FE_2O^4) indicated a color change at 800°F from black to red brown (FE_2O^3). This change reduced its emissivity which makes this

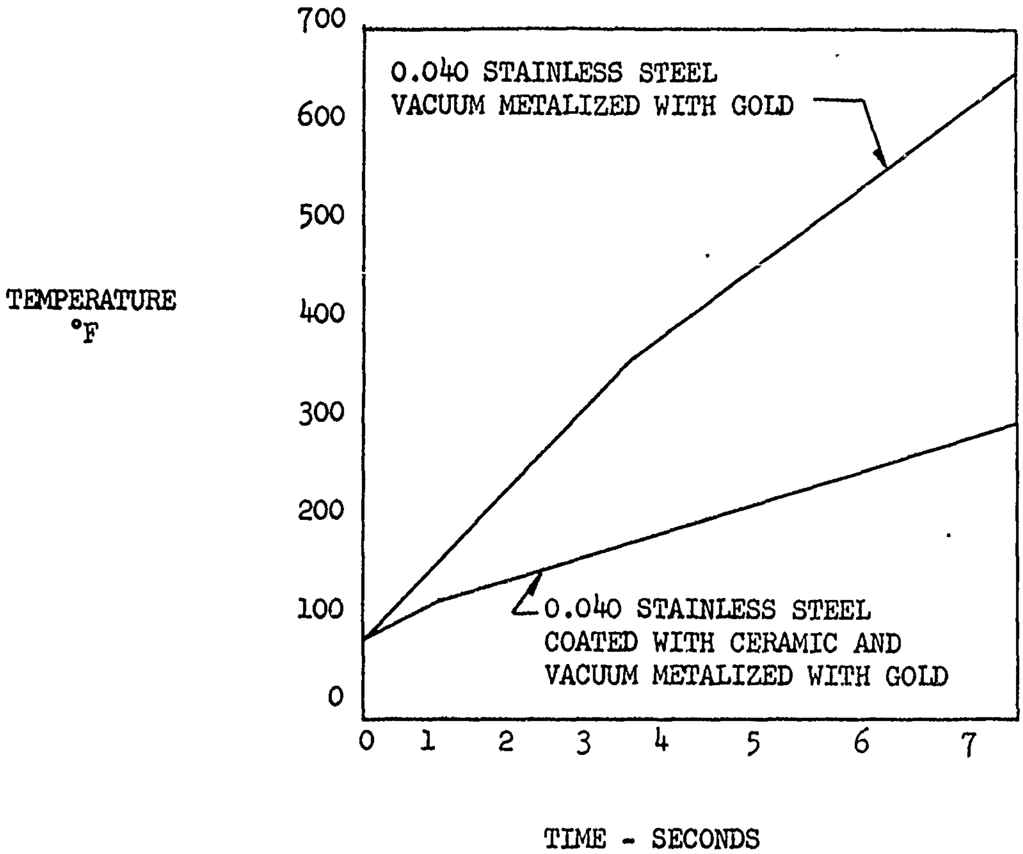

Figure 22 - TEMPERATURE RISE OF TWO STAINLESS STEEL SPECIMENS

material unsuitable for testing at temperatures over 800°F.

Oxides of nickel, copper, manganese, and cobalt were subjected to temperatures of 1400°F for 20 minutes. There was a slight color change in the oxides of nickel and copper, and a very noticeable change in manganese dioxide. All were to a lighter color. There was no change in cobaltic oxide.

Cobaltic oxide was compared to a 50/50 mixture of cobalt oxide and porcelox by applying each to the surface of three identical copper disks. All three disks were exposed to a heat flux of approximately 160 BTU/FT^2/sec. for about four seconds. The temperatures reached by the cobaltic oxide, 50/50 mixture, and porcelox were 1953°F, 1946°F and 1801°F respectively. (See Figure 23.)

The dry powdered oxides were mixed with various solutions to allow application to a specimen. Water, detergents in water, and alcohol were tried. Water does not readily mix with any of the oxides. The detergents mix well, but leave an organic film which smokes excessively at 500°F. Alcohol mixes well, and leaves no residue to smoke. The cobaltic oxide and alcohol solution, when applied to a surface and allowed to dry, has a powdery consistency which can easily be rubbed off. This is desirable for many tests, where an inspection of a heated surface, after testing, is required. However, a permanent type of coating which is not easily rubbed off is also needed, so that deflection probes and other test equipment may touch the specimen without removing the coating.

Porcelox is a good permanent type of coating even though it requires curing at about 250°F. The curing is necessary to drive off water, organic binders, and to fuse low temperature glasses which form the high temperature binders. It is purchased in a liquid form and is applied by brushing. It should be cured at about 250°F for approximately 15 minutes to drive off any water that happens to remain in the mixture, boil off organic binders, and fuse low temperature glasses.

The best coating found for increasing specimen absorptivity in the infrared range was the one made up by mixing cobalt oxide into porcelox in one to one proportions. It's easily applied, adheres well and can be used up to 2000°F.

5. Forced Cooling

To be able to round out our full scale thermal simulation capabilities we must provide means for forced cooling of specimens. Forced cooling is required when it is necessary to cool a hot surface faster than its normal cooling rate. This can happen in the following ways:

1. A high speed interceptor is cooled by deceleration (pull up maneuver) after an extended high mach number dash to the target. The deceleration allows heat transfer to a cooler boundary layer.

2. A ballistic missile could cool through radiation to space at very high altitudes. The cooling effect may be most critical since the heating cycle of a missile or aircraft is usually longer and the structure may reach equilibrium at a high temperature and then when cooled rapidly is subjected to high thermal gradients which may cause undesirable warping or buckling.

Figure 23 - Specimen Coatings

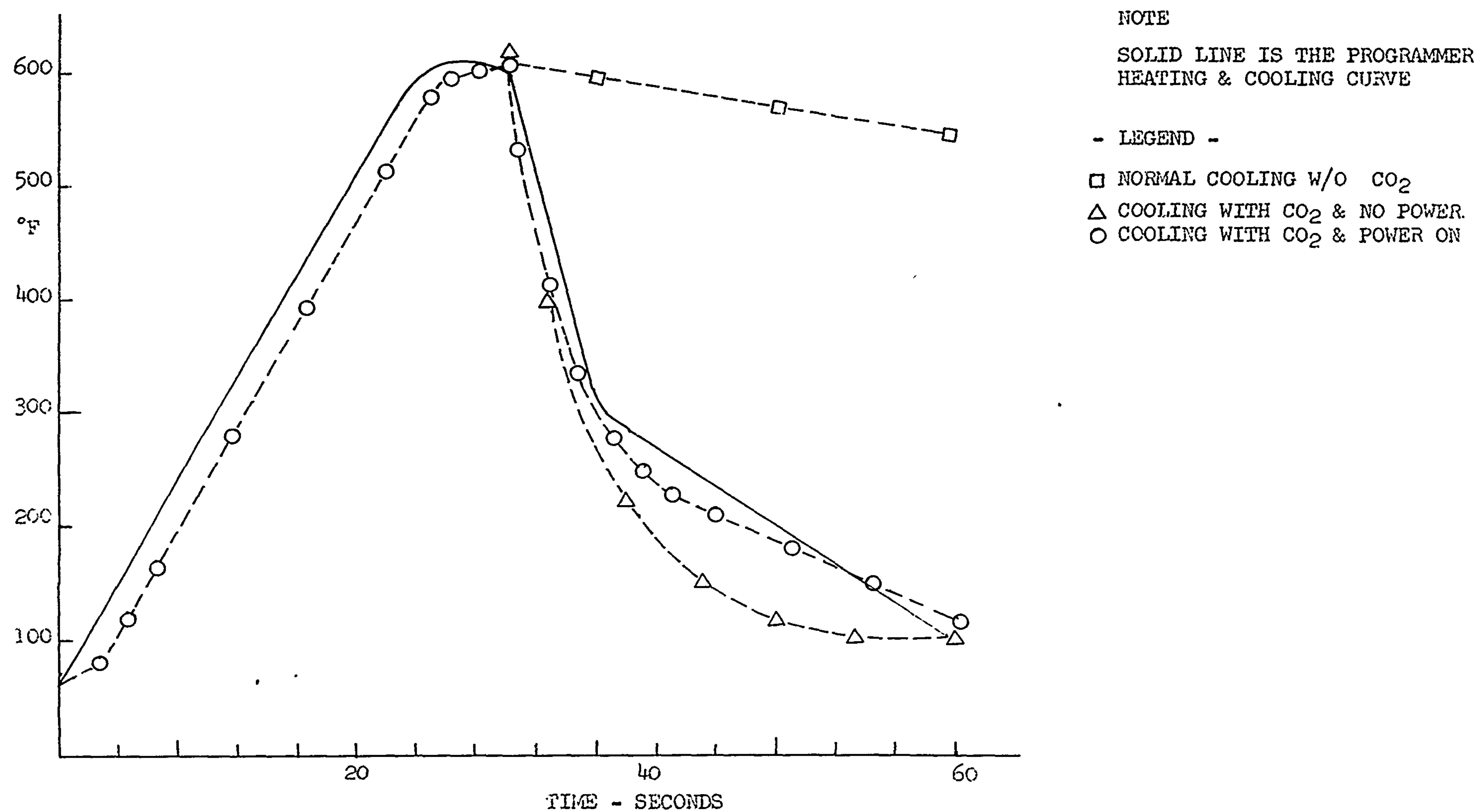

Figure 24 - CONTROLLED COOLING

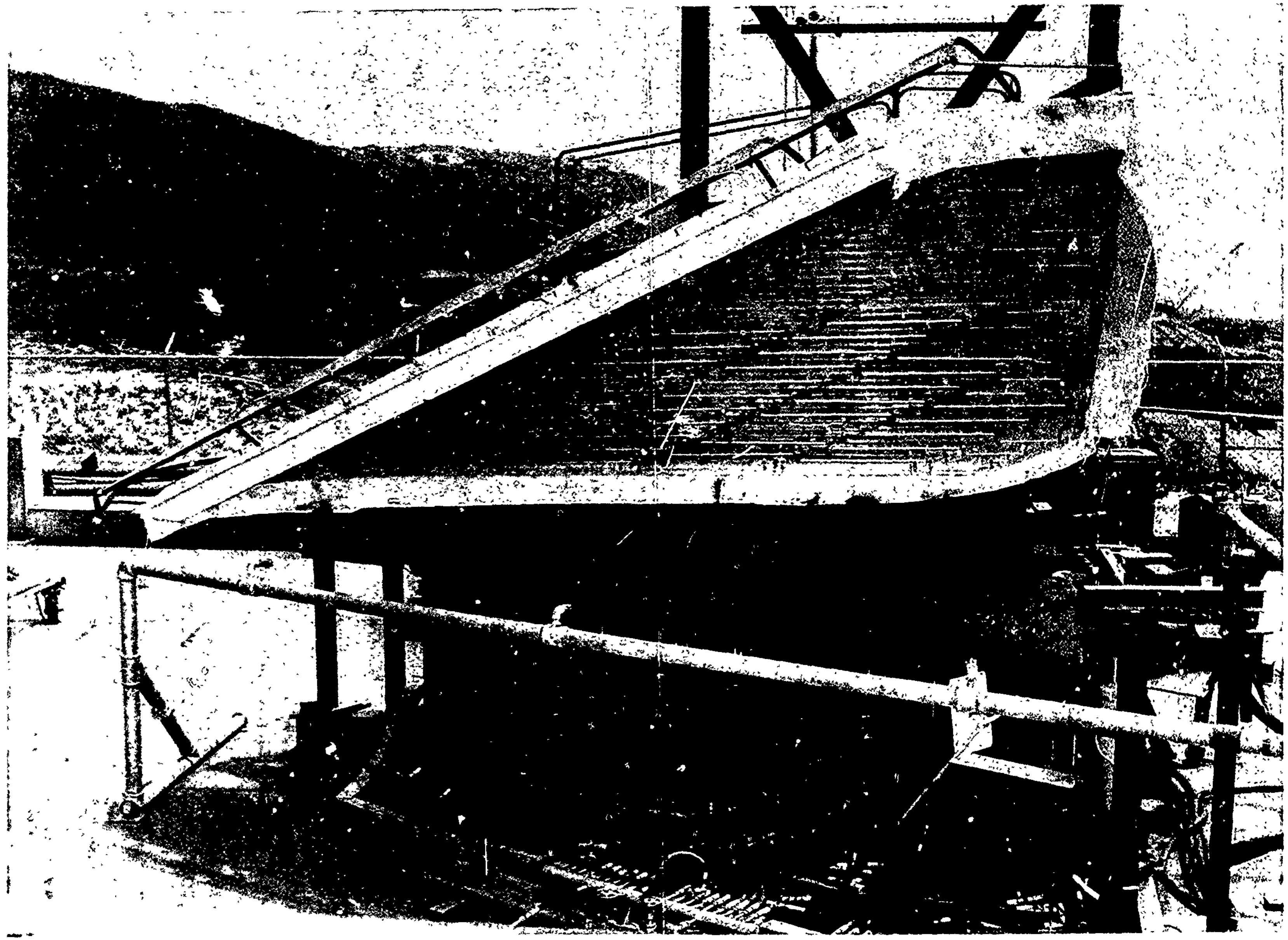

Figure 25 - Fatigue Test Of Full Scale Specimen With Thermal Simulation

Preliminary tests have indicated that CO_2 can be used for forced or programmed cooling. Maximum cooling rates have not yet been determined. However a method for obtaining controlled cooling was devised as follows:

1. In Figure 24 you'll note that when the heat power was turned off at the apex of the curve, the specimen temperature dropped off somewhat slower than desired.

2. A fixed orifice or orifices controlled by variable solenoid valves is placed so that when the temperature program computer calls for no power, the valves will be turned on to allow CO_2 to flow as a gas over the specimen.

3. The orifices were designed to provide an over abundance of cooling, forcing the specimen temperature below what is desired. This produces a heat demand signal in the computer which again supplies heat to the specimen.

   In this fashion the temperature computer programmer is used to program and close thermal simulation can be had even when forced cooling is required.

6. General

Test requirements from specimen to specimen vary greatly, such as, magnitude of specimen deflection (B-70 wing versus a missile fin) and magnitude of heat, that no universal heater is proposed.

The close relation of load, heat and measuring apparatus must be recognized and carefully reviewed when new heaters, loaders, recorders, and programmers are acquired or built. All these considerations must be taken into account when final devices are provided for individual test setups. A typical test set up for applying both programmed heat and load is shown in Figure 25. In this case since the deflection was slight and heat flow low the heat lamps were attached to one large aluminum reflector which rides up and down with the specimen as it deflects. Since the specimen is also full of fuel which cools the lower surface, heat is only applied to one side.

Note too, that in this particular test load was applied to the lower surface with compression hydraulic load cylinders.

SUMMARY AND CONCLUSIONS

1. Heat, load and time histories of todays flight vehicles occur in such combinations as to make analytical solutions virtually impossible for designers.

2. Ground tests in close simulation of operating conditions are therefore mandatory. Flight tests would be more accurate but too costly and too late.

3. These new test requirements have pushed the state of the art in their simulation to require a very costly expansion of existing facilities or acquisition of entire new facilities.

4. There is no universal method for thermal simulation in full scale testing nor do any two companies have the exact same problems nor approach them in the same way.

5. Thermal simulation capability may be added or acquired as needed in building blocks of basic power, multiple channels of temperature and load programmers, controllers, and recorders. Special devices surrounding the specimen to be used for the actual input to the specimen are designed to take care of specific requirements.

6. In fullfilling the above requirement we find that:

   A. Recent work in this field has provided us with equipment, know how, and facilities for thermal simulation.

   B. Heat programmers and computer controller equipment is almost an off-the-shelf item.

   C. Money is still hard to come by as are such items as load programmers.

   D. The use of infrared heaters is standard practice.

   E. We're still pushing the state of the art in some areas such as high heat flux (nose cone heat simulation), load programmers, rapid automatic data recorders and processers, absorptive and reflective coatings, and new heaters.

   F. Need for adding costly building blocks for heat capability can be reduced by concentrating on improving heater efficiency.

7. Infrared heaters can be used for thermal simulation up to temperatures of 3000°F, heat fluxes of 200 BTU/FT^2/sec. They are easy to control and can be made very efficient by use of gold/ceramic cooled reflectors in conjunction with test specimens coated with cobalt oxide/porcelox. Thermal conditions for all manned vehicles can be simulated with these heaters. All areas except nose cones and sharp leading edges of missiles can be heated in the same fashion since the peak heat flux for 90% of these bodies is approximately 25% of the maximum shown.

8. The infrared heater can be supplemented with electric arc heaters (plasma jets) for temperature and heat fluxes in excess of those above and up to 10,000°F and 1500 BTU/FT^2/sec. These heaters too can be controlled by the basic programmer computer controller.

9. Programmed or forced cooling is mandatory and feasible using liquid CO_2 expanded to gas through nozzles and sprayed over hot specimens. Accurate control is again maintained by the heat controller which is used to overcome an excess of cooling.

10. The best type of reflector discovered in these studies is stainless steel sprayed with ceramic and coated with gold.

11. The best type of specimen absorptive coating was found to be a 50/50 mixture of cobalt oxide and porcelox.

12. A mach 3 jet transport such as that shown in Figure 26 and proposed for operation by 1970 could be tested for fatigue, full scale, including thermal simulation using the equipment described in this paper. Four of the basic building blocks would be required, totaling 24,000 KVA and 48 programmer controllers. The airplane in this example would operate above 60,000 feet and would not be supersonic below 35,000 feet, is 211 feet long and has a 98 foot wing span.

13. The amount of power available still limits the size of the total specimen that can be tested.

14. Other approaches to this problem I'm sure lend themselves as well to thermal simulation and have been used successfully by others.

15. Many problem areas still exist. For example, these solutions still fall short of the ICBM thermal simulation. We must continue to push the state of the art in thermal simulation.

16. We are deeply indebted to those who have done the real spade work in this field.

Figure 26 - Future Jet Transport Requiring Thermal Simulation In Full Scale Fatigue Tests

LIST OF COMMON SYMBOLS

g	acceleration due to gravity (ft/sec^2)
C_p	specific heat (BTU/#/°F)
Q	heat flow BTU/hr.
T_s	surface temperature (°F)
T	temperature change (°F)
T_{aw}	adiabatic wall temperature (°F)
B	radiation constant (BTU/HR°F/FT^2)
A	area (in.^{2})
E	load voltage (volts)
I	current (amperes)
h	heat transfer coefficient (BTU/HR°F/FT^2)
Y	conversion constant $\frac{\text{BTU}}{\text{hr-watt}}$
C	thermal capacity BTU/°F/FT^2
N	thermal efficiency of heater %
t	time (seconds)
$\frac{dT_s}{dt}$	rate of temperature change °F/sec.
E	emmissivity

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THE DETERMINATION OF EQUIVALENT FATIGUE TEST LOAD SPECTRA FROM FLIGHT RECORDS

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The problem of deducing equivalent fatigue test load spectra from flight records so that the relationship between the damage produced in a test and that in actual operation can be quantitatively estimated is discussed. To date there is very little test data which compare the fatigue damage produced by completely random loadings with that of the derived ordered loadings. A summary of the type of research studies which are required to investigate the many parameters which are involved in constructing a test spectrum from flight records such as peak counting methods, loading block size, stress interval, ground-to-air cycle, etc., is presented.

INTRODUCTION

To commence this discussion with some introductory comments pertaining to the growing importance of fatigue in the design of the structure of flight vehicles would be highly superfluous since the scope of this symposium provides in itself sufficient evidence that its importance is duly recognized.

As indicated by the agenda for this meeting, this problem has many facets, one of major importance being the relationship between the effects of test loadings and the effects of service loadings. This aspect of the over-all fatigue problem brings into sharp focus two questions:

(1) How should flight test records be reduced to be of most significance from a fatigue standpoint?

(2) What is the quantitative relationship between the fatigue damage produced by the actual random service data and the spectrum of loads deduced therefrom which are used in fatigue tests or analysis?

It is, of course, obvious that if actual load histories could be applied in tests and if they could be applied directly in a fatigue damage theory, neither of these questions would warrant investigation. However, the formulation of a reasonably accurate fatigue damage theory has been sufficiently elusive when limited to the solution of simple spectrum-type loadings that hopes for an early solution to more complex loadings would appear to be unduly optimistic. It is conceivable that test techniques could be developed which would accomplish the direct application of service loadings, but they would be costly and it is conceivable that simpler procedures could accomplish the same objectives.

Failing in either of these, the two problems posed must be investigated if fatigue tests or analyses are to be made more meaningful as measured against service usage. This would be true even if a complete fatigue damage theory were developed and if the complexity of fatigue testing is to be kept at a relatively simple level. Irrespective of any advancements in analytical methods for fatigue prediction, testing will in all probability always remain an integral part of a fatigue evaluation program. As refined as stress analysis techniques have become for static strength determination, there are still many instances in which it is deemed necessary to make tests to substantiate the analysis. The fatigue strength of a structure being more sensitive to the local stress distributions than the static strength, detailed stress analysis much more refined than that sufficient for static analysis would be required if an analytical approach to fatigue prediction is to be successful. It is doubtful that within practical limitations such a stress analysis can be accomplished on complicated structures. Hence, fatigue tests will be required if for no other reason than inability to produce a sufficiently accurate stress analysis to be used in a fatigue theory even if a suitable one existed.

Unfortunately, the answers to the two questions posed above cannot be provided in this paper. The reason is simple. The necessary test data to provide the answers do not exist. Recently there has appeared in the literature the results of some investigations aimed at analyzing the fatigue problem

associated with truly random loadings. To the knowledge of the author, Ref. 1 is the most advanced of these studies. The remainder of this paper will describe current procedures for fatigue spectra determination, the reasons for their inadequacy and the research work required to make such spectra more meaningful in terms of actual aircraft usage. Even if the answers cannot be supplied in this discussion, the prospects for eventual solution are exceedingly good since the Air Force has seen fit to support an extensive research program, parts of which are described herein.

LOADINGS

In actual operation, aircraft are subjected to a wide variety of randomly applied loadings. These loads arise from various sources and include atmospheric turbulence, maneuvers, landings, taxiing, ground handling, and the ground-to-air-to-ground transition. In order to construct any realistic fatigue spectrum, statistical data from actual flight operations must be obtained. Through the years, such data have been collected by various means with particular emphasis on loadings arising from gusts and maneuvers. In recent years, considerable data obtained with VGH recorders which produce continuous traces of airplane response have become available. Fig. 1 shows a typical response to atmospheric turbulence and Fig. 2 is characteristic of an airplane response to rapid maneuvers.

If a recording mechanism were turned on continuously, that is, while the airplane was on the ground as well as in the air, a trace as depicted in Fig. 3 would be obtained. After sufficient data of this type have been collected to be statistically significant, the next question which arises is what is to be done with it? To be sure, a proper fatigue evaluation cannot be made without these data, but once the data are available a new series of problems are encountered. If the prime consideration is the determination of criteria for static strength, the problem is relatively simple and the miles of records must be searched for maximum conditions only. But if the analysis of these records is to contribute to the solution of the fatigue problem, the full range of magnitudes as well as frequencies and sequences must be considered. In the absence of suitable test information, some rather arbitrary decisions must be made as to the manner in which flight records are reduced into spectrum form.

REDUCTION OF FLIGHT RECORDS

The tremendous task of reducing flight records has been performed by the NASA, WADC, and others. It is not the intention of the author to belittle this effort since the data produced has been put to invaluable use during the past few years. However, as mentioned above, lacking certain basic data, many arbitrary decisions had to be made in the processing of the records and, quite naturally, this has obscured many of the variables which may be significant from the fatigue standpoint. Most of the records were reduced by making peak counts and accumulating the frequency of occurrence. This would appear to be at least one starting point. However, the first question that arises is what is a peak and what is not a peak? Referring to Figs. 1 and 2 it is seen that the traces do not possess the characteristics of simple sinusoidal curves for which the peak values would be obvious. In a major excursion from the 1g level, many hills and valleys may appear in each trace. Rather arbitrary rules have therefore

been established to determine which of these should be regarded as major peaks. In some instances these minor peaks are included and in others ignored. Even though these minor peaks do not begin their excursion from the base line (1g) they are sometimes so recorded. Considering turbulence records, it was noted that the frequency of positive peaks and negative peaks was approximately equal and hence such peaks were linked together to form complete cycles even though in the actual record the sequence of such peaks may have been separated by a substantial period of time.

Another aspect of this problem of fundamental importance in view of the characteristics of the load traces is whether peak counts or some other measure of the trace such as frequency of stress excursion is the more informative measure insofar as fatigue life is concerned. A systematic evaluation of the effect of these different variables has never been made.

Based on the general procedures described above, typical gust and maneuver frequency curves as shown in Figures 4 and 5 respectively have been derived. This type of information has then been used extensively in the determination of loading spectra for both fatigue analysis and test.

Using atmospheric turbulence data again as an example, what is the significance of the information contained in Fig. 4 compared with the actual experience of the aircraft. Let it be assumed that the actual recorded response of an aircraft is as shown in Fig. 6 (a). As a result of one peak count method of analyzing the data and the coupling of positive and negative peaks of equal magnitude, the response of the aircraft is in effect changed to that shown in Fig. 6 (b). Also shown in Fig. 6 (b) is the response that would be deduced if a range count system were utilized instead of a peak count system. In any event, it can be clearly observed that sequences are altered and that the magnitudes of reported load range are dependent on the method of record reduction which is used. The relationships between the fatigue damage caused by the loading pattern shown on Fig. 6 (a) and by those shown on Fig. 6 (b) has never been established.

FATIGUE SPECTRA

In the preceding section some of the shortcomings of flight data reduction techniques to adequately depict the loadings incurred by the structure were discussed. Even if such were not the case, and turbulence data such as contained in Fig. 4 provided an exact description of the frequency of gust loadings to which the aircraft was subjected, there are still many unanswered questions as to the method for constructing a discrete load fatigue spectrum. These questions are not eliminated with the use of the more sophisticated methods of random loading data reduction such as those provided by power spectral analysis techniques.

Let it be assumed that from the gust data of Fig. 4, a stress frequency distribution for a particular location on the aircraft is derived and it is an exact replica of the airplane response except for the sequencing of the loads. Fig. 7 is an example of such a frequency distribution. To construct a spectrum of discrete loadings the initial step is to convert the stress frequency distribution curve of Fig. 7 into a step function. There is an unlimited

number of ways in which this can be accomplished. Fig. 8 shows examples of a few of them. From the standpoint of a fatigue test, if the stress interval that is selected is large, the test procedure is simpler since it requires fewer changes in set-up. On the other hand, if the stress interval is small, it is more likely that the damage pattern will be more accurately duplicated.

After the stress interval has been selected, and the number of cycles to be applied at each stress level determined, a decision must be made as to sequencing of the loads. Here again a wide assortment of procedures may be followed. All of the low loads may be applied in a block, and then the next level, etc. An alternate method may be to take a certain percentage of the loads at each level and apply these in sequence and then repeat the procedure until the full loading is accomplished. A description of this process is shown in Fig. 9. At the present time very little is known about the effect of stress interval and block size on the fatigue damage produced by the derived test spectrum.

GROUND-TO-AIR CYCLE

As mentioned previously, an aircraft is subjected to a wide assortment of randomly applied loadings which originate from various sources such as turbulence, maneuvers, landing, taxiing, ground handling, etc. If loadings from each source are analyzed separately, the same problems in formulating a test spectrum as exemplified by the example above for turbulence would be encountered. These problems are further complicated when the totality of loadings to which the aircraft is subjected are examined. Fig. 3 is a schematic depiction of wing loads as an aircraft goes through a series of flights. It can be observed that there is a major cycle of load variation for each flight produced by the transition from the ground-borne to air-borne condition. This cycle is depicted in Fig. 10. Various quantitative definitions of stress range have been proposed for this cycle such as the variation from the 1g stress on the ground to the 1g stress in the air, minimum stress on the ground to maximum stress in the air per flight, rms stress on ground to rms stress in the air.

The contribution of this single cycle per flight to the total fatigue damage has been the subject of much conjecture. Depending upon the assumptions used to define the cycle, its importance to the total fatigue damage can vary from minor to major. There is little test evidence to support any position.

TEST PROGRAM

The preceding sections of this discussion have purposefully emphasized the serious limitations in existent capability for determining meaningful fatigue test spectra for aircraft. This was done to form a proper basis for a description of an extensive research program to be conducted by Lockheed Aircraft Corporation, California Division, under sponsorship of the Aircraft Laboratory, Wright Air Development Center, which it is hoped will provide answers to many of the questions that have been raised in this paper.

The overall purpose of this test program will be to evaluate the relationship of fatigue damage produced by random loads and the ordered spectra deduced therefrom. The possibility of making such tests has been greatly enhanced by the development of servo-controlled hydraulic jacks which can rapidly apply the complex service loading histories recorded on magnetic tapes.

In the initial phases of the investigation, axially loaded 7075-T6 aluminum sheet metal specimens with two different intensities of stress concentrations will be utilized. The starting point for the program will be the collection of actual flight records on electromagnetic tape from at least two different types of aircraft. These records will include gust, maneuver, and taxi data. A typical record for gust response would be as shown in Fig. 1.

A series of fatigue tests will then be made to determine the influence of load sequence in a random load pattern. This can be accomplished by running one set of specimens with the tape going forward and another identical set with the same tape run backwards. This could also be accomplished by resplicing the tape.

The next set of tests will be run in an attempt to determine the parameters of a random loading which are the most sensitive measure of the fatigue damage produced. At least two sets of specimens will be tested using random loads of significantly different characteristics. Utilizing automatic data reduction methods, the random loads recorded on the magnetic tapes will be reduced to spectrum form. Parallel sets of specimens will be tested using these derived spectra. The method of obtaining the derived spectra will be varied, first using peak counts, then range counts, then range counts associated with different mean stresses, etc. The time to failure for each of the sets of random loads will be compared with the failure time for the spectra derived by each of the different methods of reduction. The method which comes closest to producing the same ratio for the different sets of random loads will be recommended for use. Naturally, it would be indeed fortunate if a method were discovered in which this ratio turned out to be approximately unity. A ratio of unity signifies that the fatigue damage produced by the derived spectra is exactly the same as that produced by the random loadings. As desirable as a unit ratio may be, it is not essential to the success of the investigation.

The next essential phase of the test program is to investigate the effect of stress interval and block size on fatigue damage. As depicted in Figs. 8 and 9, it is possible to form many different sets of test spectra from a given stress spectrum and to sequence the loads in many different ways. Utilizing derived spectra from random flight records of different characteristics, a series of fatigue tests will be made in which both the stress interval and block size are varied. The objective of this part of the program is to determine the largest stress interval and the largest block size that can be utilized without essentially modifying the fatigue damage produced.

The next step in the program is to make fatigue tests using loading records which would be representative of an aircraft in continuous operations. Random-loading tapes which will include taxi, as well as flight data will be generated for both transport (bomber) and fighter type aircraft. A record from such a tape would be similar to that shown in Fig. 3. With these loadings, a series of fatigue tests will be made. The records on these tapes will then be reduced to test spectra, utilizing the information obtained from previous portions of the program. Again, a series of specimens will be fatigue tested and the damage ratios between the random and ordered test spectra obtained. These ratios will then be compared with those obtained in the first phase of

the program to determine if the relationships between random and ordered test loadings obtained for relatively simple records are also obtained when more complex loading histories are used. In this phase of the program the effect of the ground-to-air cycle will also be extensively investigated. Utilizing different definitions of ground-to-air cycle, constant load amplitude fatigue tests will be made (see Fig. 10) and the damage from these loadings compared with that produced by the random loadings.

CONCLUSIONS

In conclusion, may it be re-emphasized that there is a serious lack of basic information to properly evaluate the relationship between fatigue damage produced by the random loads which occur in aircraft service and ordered spectra of loads which are utilized either in tests or analysis. It is hoped that the test program that is briefly described in this paper will shed some light on the questions that have been raised and lead to an eventual solution of the problem.

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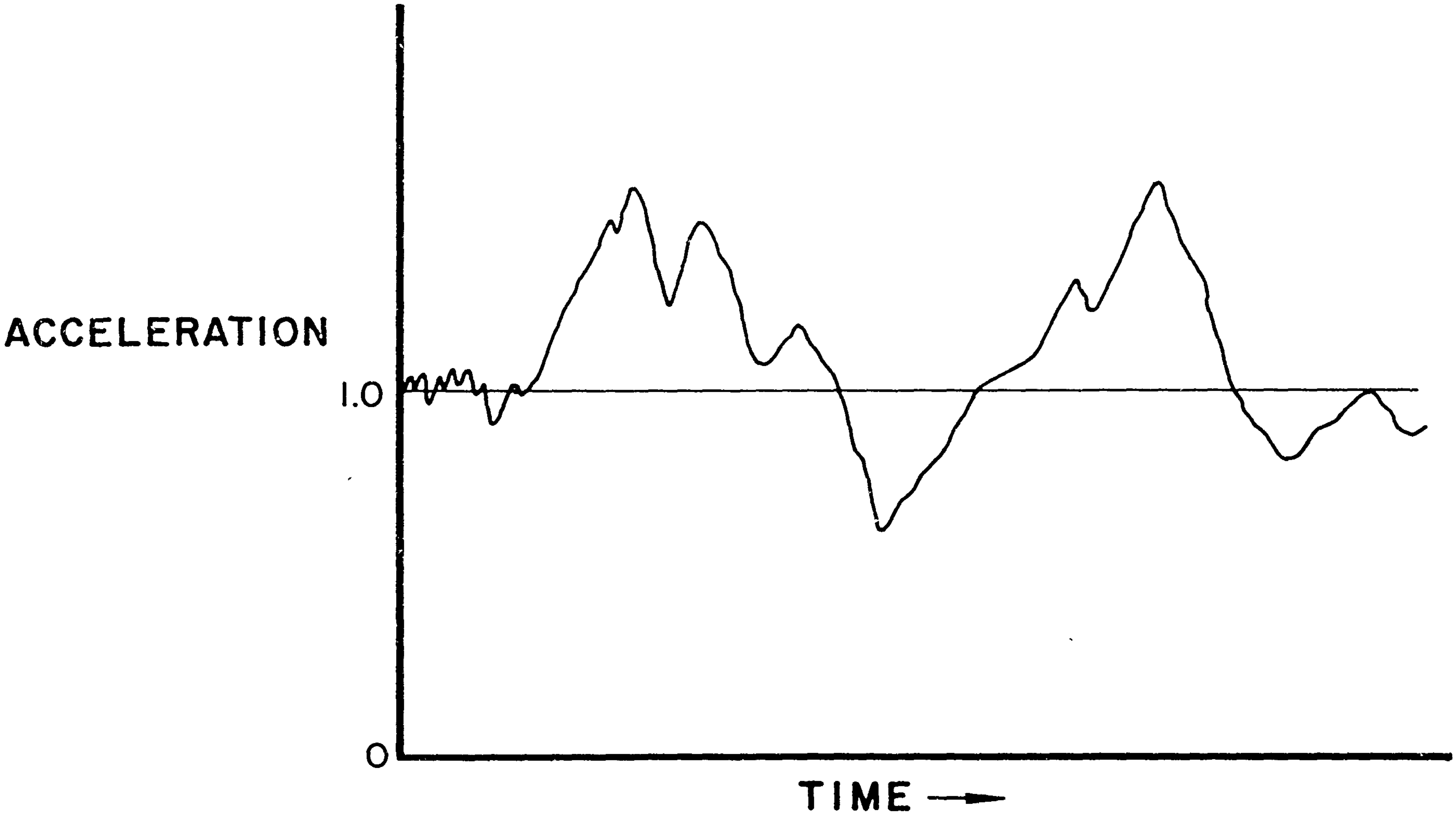

FIG. 1 TYPICAL RESPONSE OF AIRCRAFT TO ATMOSPHERIC TURBULENCE

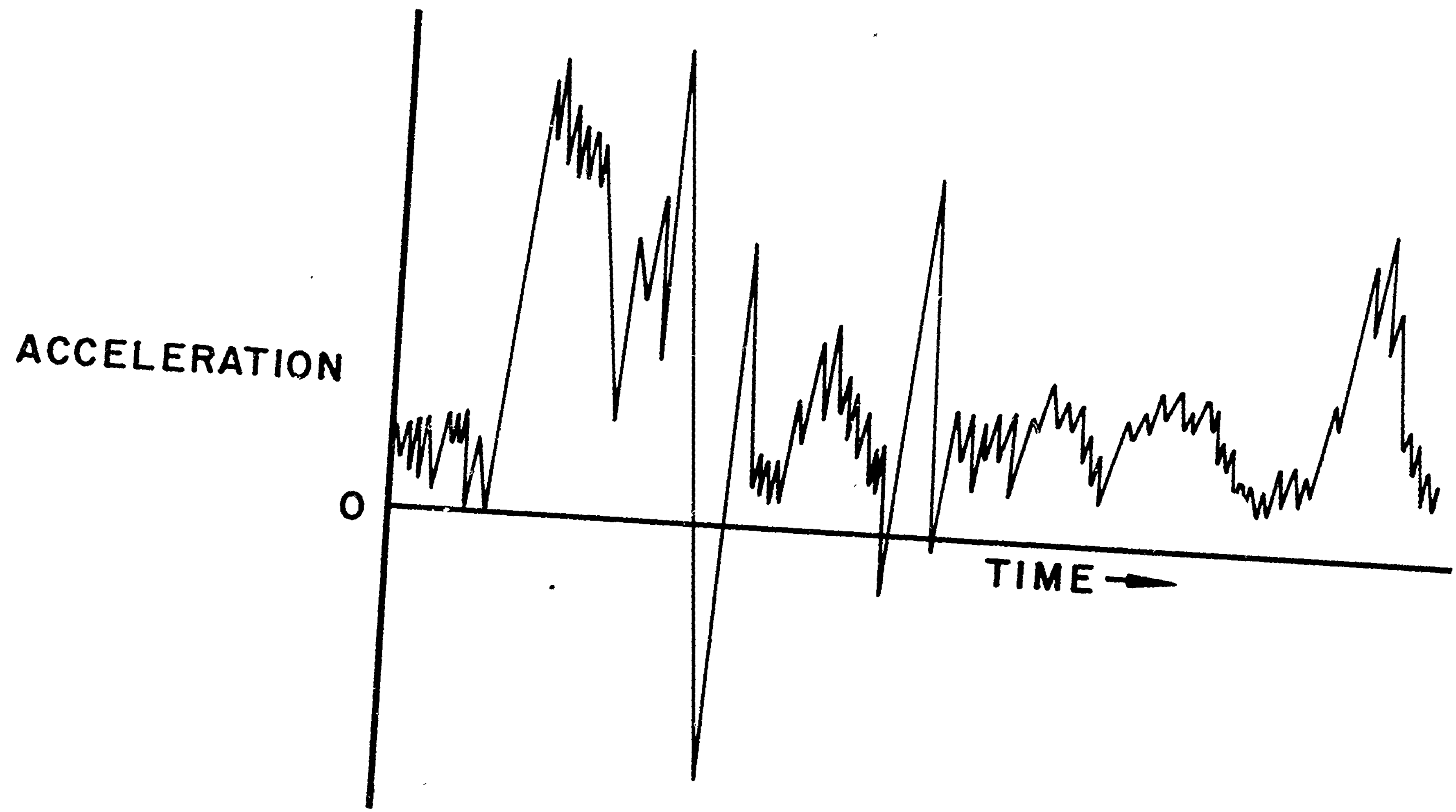

FIG. 2 TYPICAL RESPONSE OF HIGH PERFORMANCE AIRCRAFT TO RAPID MANEUVERS

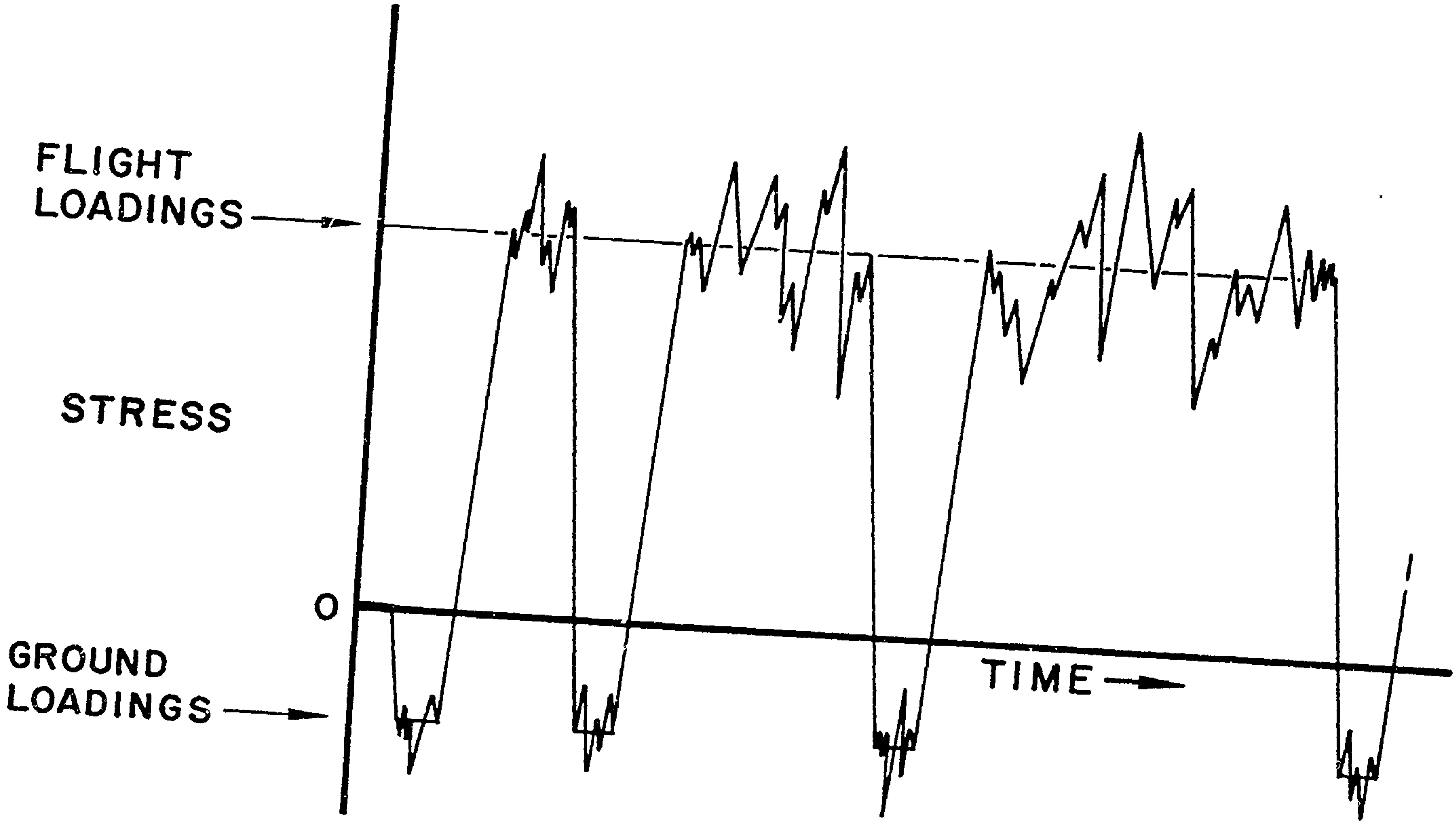

FIG. 3 SCHEMATIC REPRESENTATION OF SERVICE LOADINGS

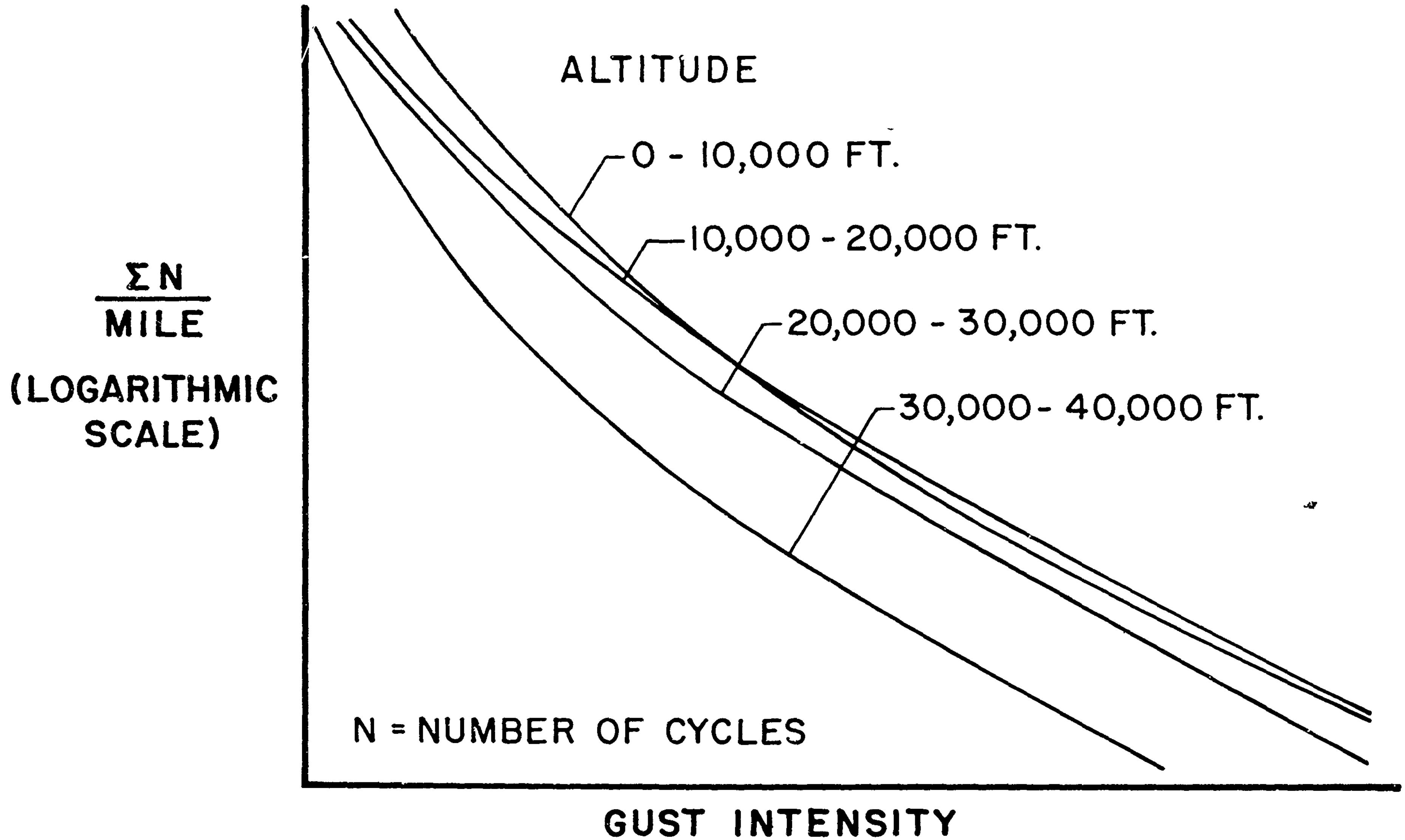

FIG. 4 SPECTRA OF GUST INTENSITIES FOR SEVERAL ALTITUDE INTERVALS

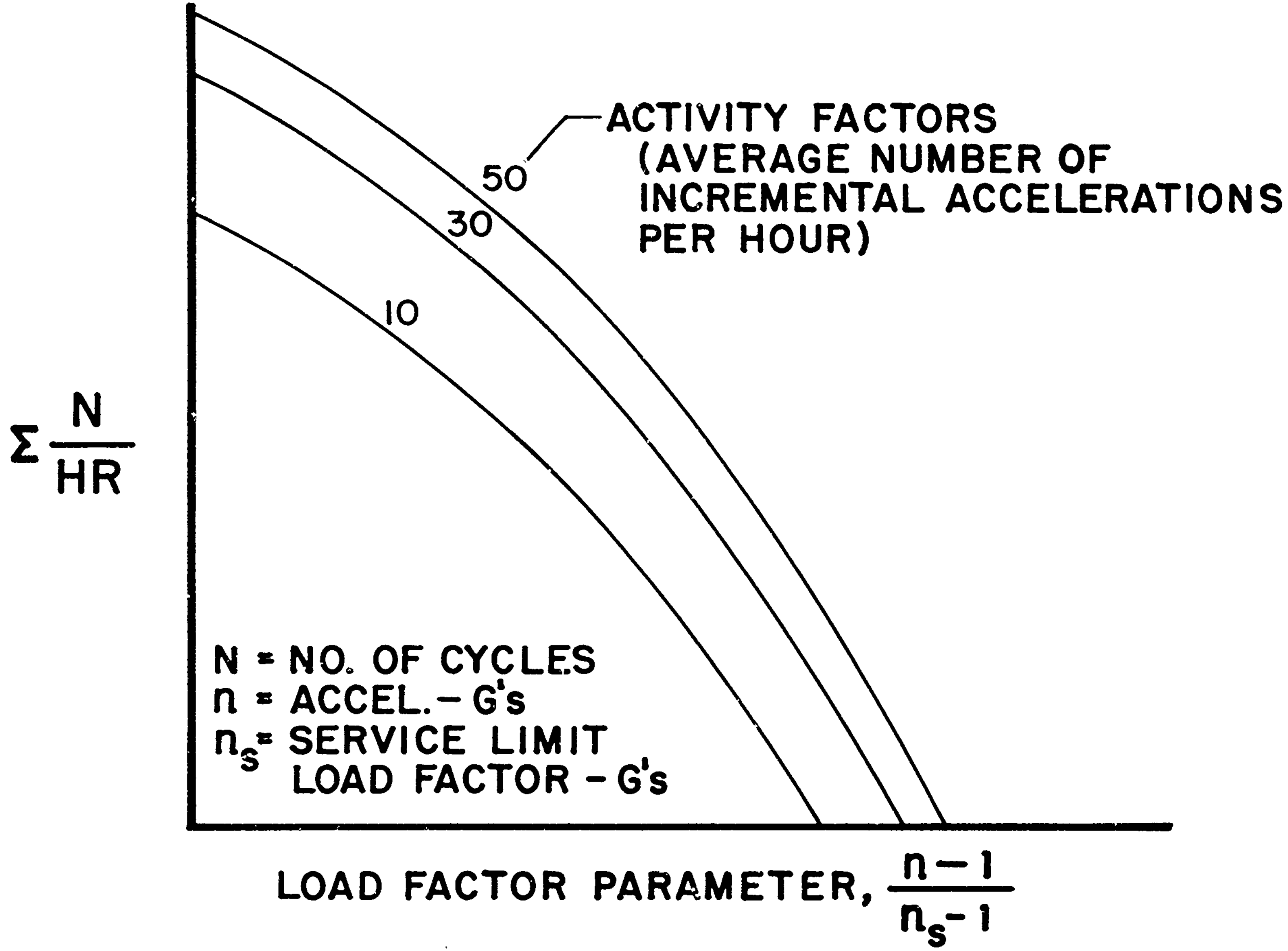

FIG. 5 ACCELERATION SPECTRA FOR HIGH PERFORMANCE AIRCRAFT

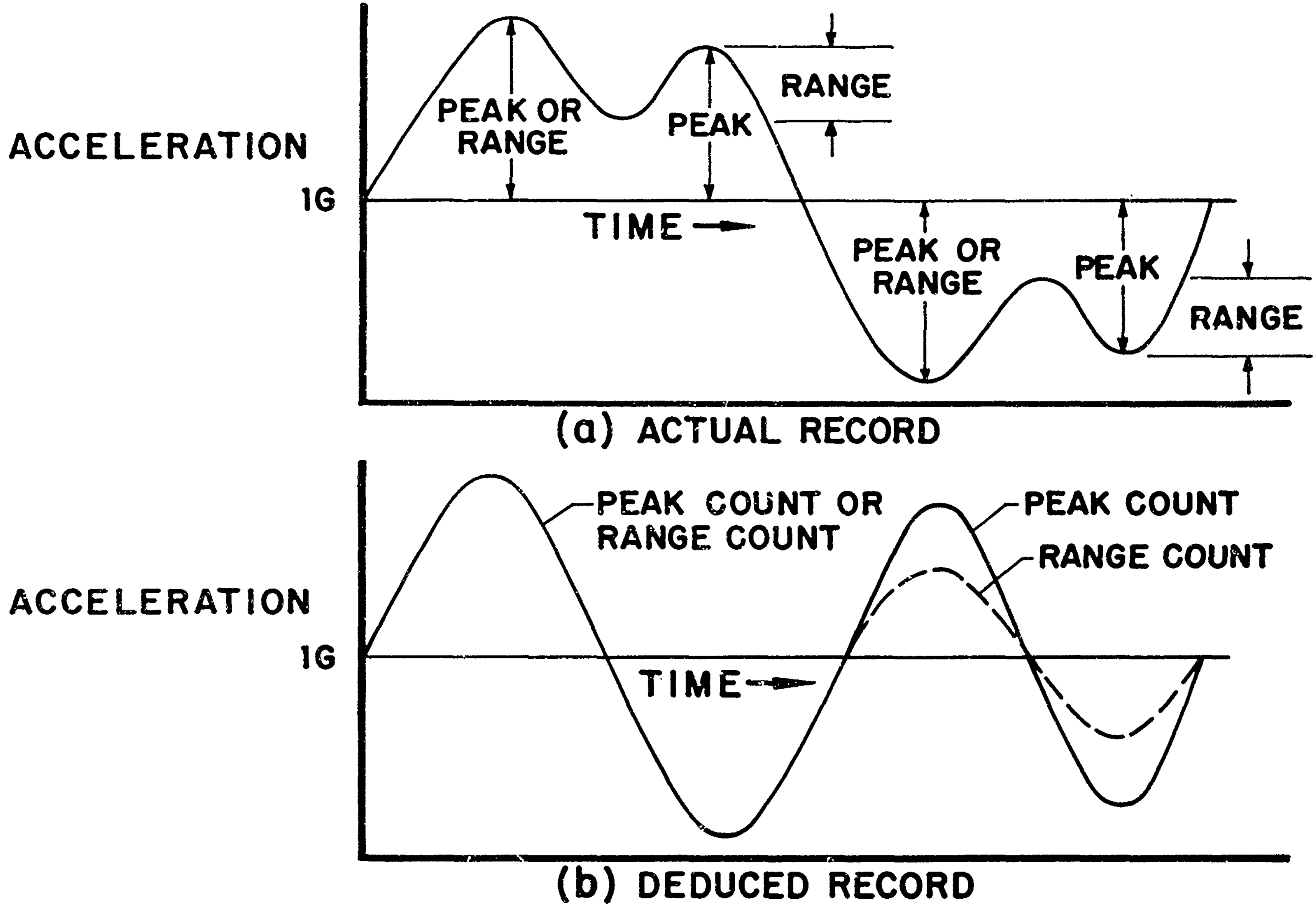

FIG. 6 SERVICE RECORD SIMPLIFICATION

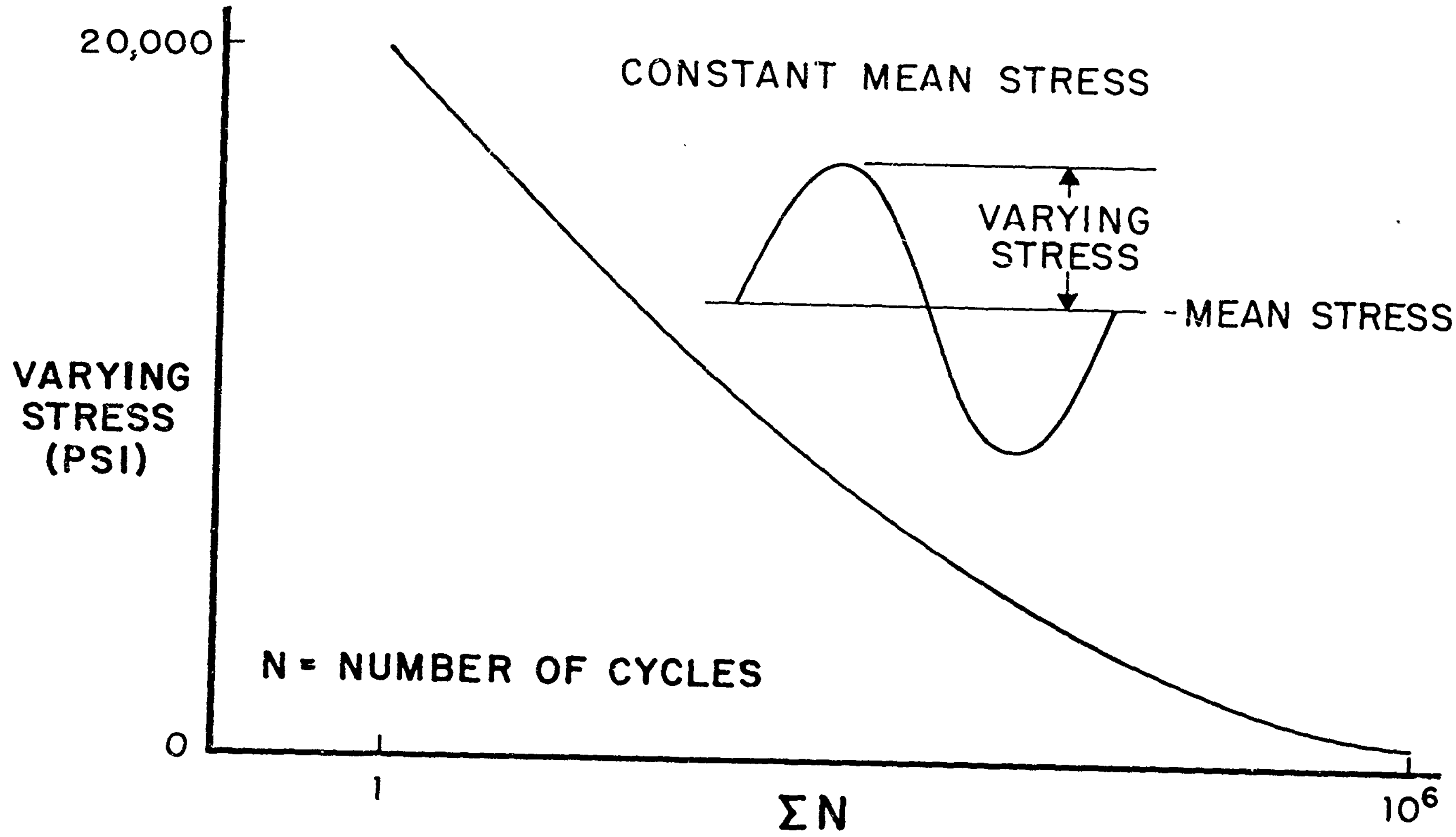

FIG. 7 REPRESENTATIVE STRESS SPECTRUM BASED ON GUST LOADINGS

VARYING STRESS - KSI →	2	4	6	8	10	12	14	16	18	20	ΣN
NUMBER OF CYCLES AT EACH LEVEL ↓	730000	215000	44000	8400	2000	450	109	29	9	3	10^6
	870000		124800		4902		277		21		10^6
	945000			54400			588			12	10^6
	975000				24921				79		10^6
	989000					11000					10^6
	994800						5200				10^6

FIG. 8 EXAMPLES OF POSSIBLE TEST SPECTRA DEDUCED FROM STRESS SPECTRUM ASSUMING DIFFERENT STRESS INTERVALS

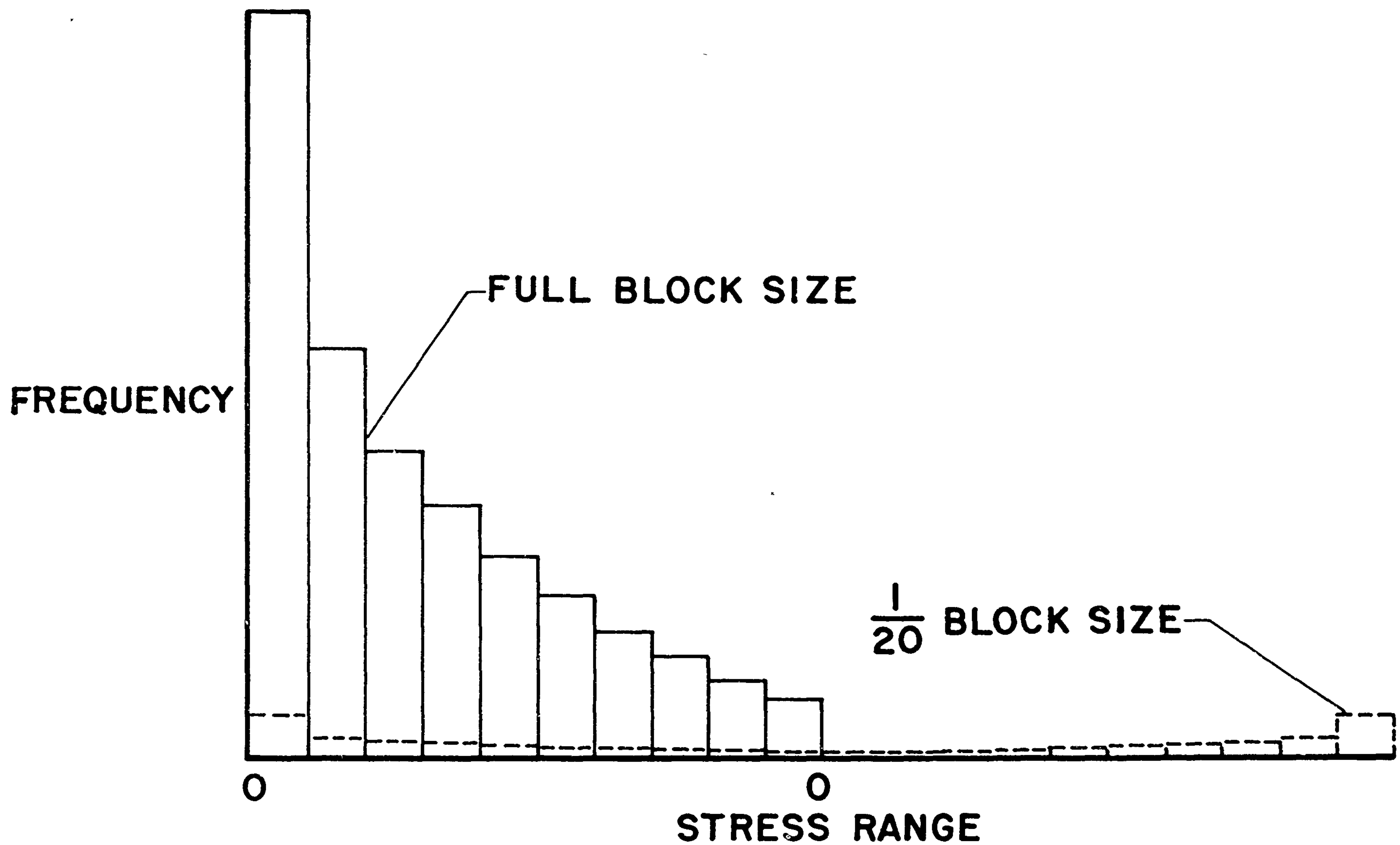

FIG.9 EXAMPLES OF ORDER OF APPLICATION OF LOADS AFTER STRESS INTERVAL DETERMINED - 2000 PSI INTERVAL ASSUMED

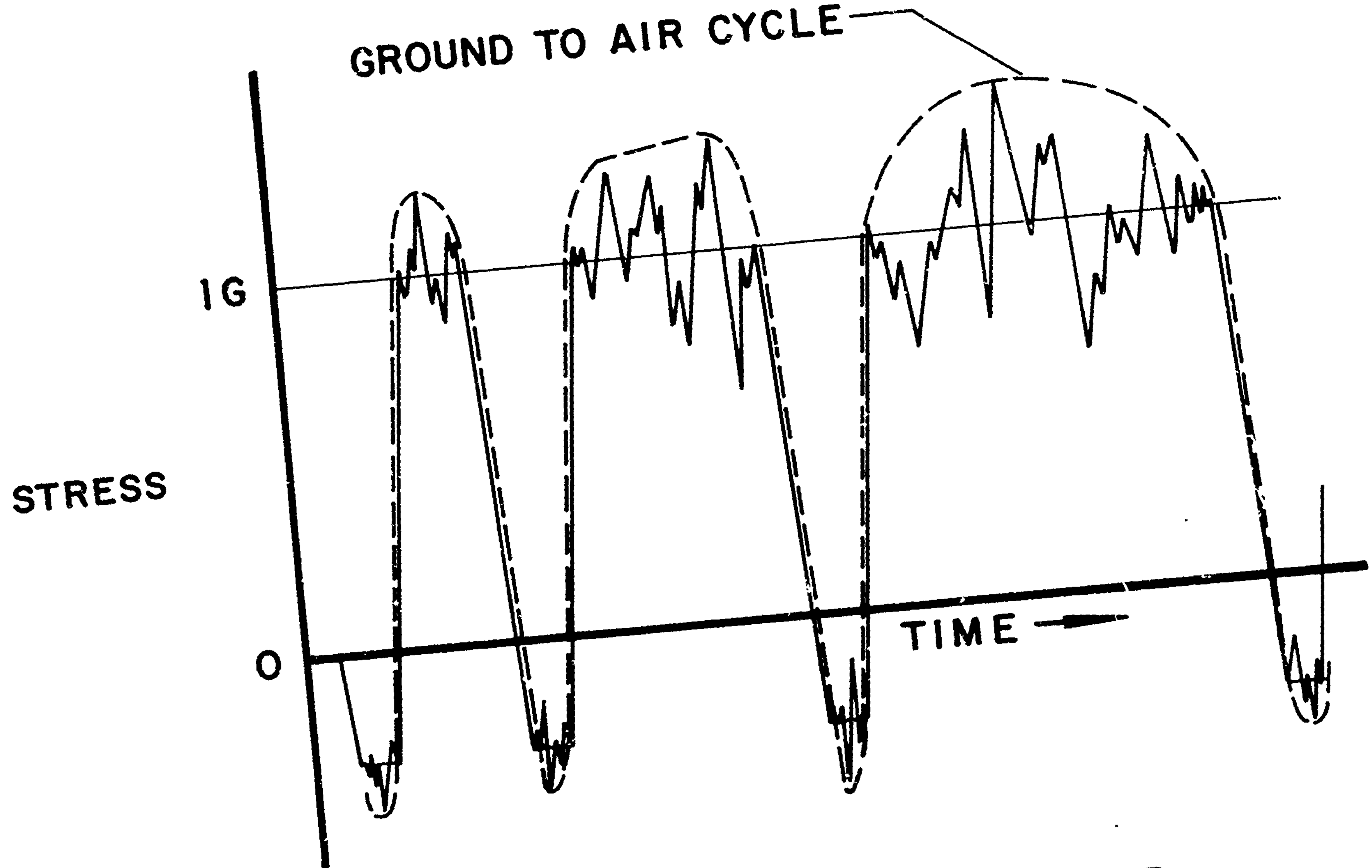

FIG. 10 REPRESENTATION OF GROUND TO AIR CYCLE

THE WADC FATIGUE TEST PHILOSOPHY

By

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This paper discusses the following seven areas of a full scale fatigue test program: (1) Test article requirements; (2) Test load spectrums; (3) Test load simulation; (4) Supplemental specimen test requirements; (5) Failure inspection techniques and requirements; (6) Test data interpretation; and (7) Future aspects of temperature simulation as an accumulative effect. The general highlights of each area are presented with an overall objective to correlate them all into what the WADC considers a complete full scale program. No attempt is made to detail exact procedures in any of these areas, only the general requirements are delineated and certain items that tend to be overlooked emphasized. As the title of the paper states, the philosophy of the WADC fatigue test requirements is presented. It represents the philosophy on this subject that exists today. Obviously, this philosophy may change as the state-of-the-art of fatigue certification advances.

INTRODUCTION

The Air Force Structural Fatigue Certification Program as delineated in WADC Technical Memorandum WCLS-TM-58-4, dated 27 June 1959, calls out specific fatigue tests as part of this certification program. It is the intent of this paper to elaborate on these fatigue test requirements and present the WADC test philosophy underlying the requirements.

The following major areas of full scale fatigue testing will be discussed: (1) Test article requirements; (2) Test load spectrums; (3) Test load simulation; (4) Supplemental specimen test requirements; (5) Failure inspection techniques and requirements; (6) Test data interpretation; and (7) Future aspects of temperature simulation as an accumulative effect.

It is immediately apparent that the major discussion areas mentioned are each in themselves worthy of a detailed discussion as lengthy as this paper. Of necessity, therefore, this paper will only touch on the highlights of each area with an overall objective to correlate them all into what the WADC considers a complete full scale fatigue test program.

TEST ARTICLE REQUIREMENTS

Due to the state-of-the-art of fatigue analysis procedures, especially as to accuracy of end results, considerable reliance must be placed on test data. From this standpoint WADC feels that a full scale, complete article, fatigue test is a must in most all instances. The test article must be completely representative of the tactical vehicle. This aspect is even more important in the case of fatigue testing as compared to normal ultimate load structural testing, since relatively minor structural differences which would not discernibly effect the ultimate strength might have a profound effect on the fatigue life of the area in question. This test article should, in those instances where the aircraft involved is still in production, be a virgin test specimen with no flight history or at the most a one-time flight from the fabrication site to the test facility, should they be in different locations. Where out of production aircraft are involved, careful selection of a low-time aircraft should be made, with particular attention being given to Form 1 write-ups that might indicate un-representative usage at some time (minor or major damage which was repaired, etc.). In addition, a complete pre-test inspection of the test article should be made to clearly establish the initial condition prior to test load applications.

It is also the WADC feeling that the test article should be complete, that is, all secondary structure such as fairing, wing leading edges, trailing edge sections in place, and all control surfaces such as flaps, ailerons, spoilers, etc., in place. Even though some of these items may not be loaded during the fatigue test, the induced loads they pick up and transfer due to relative deformation, and the stability they impart to the major structure can only be duplicated by their actual presence during the test. It is also felt that the completeness of the test article should be symmetrical, that is, for instance - both wings should be complete, both sides of the horizontal tail should be complete, etc. This procedure automatically provides for dual test specimens in these instances and in addition to this desirable feature, should test failure and repair render one side non-representative during the course of the program, continued valid results may be obtained from the alternate side without resorting to the addition of major virgin components during the program and with the attendant clouded results.

TEST LOAD SPECTRUMS

The selection and development of the test load spectrum is perhaps the most difficult and also the most important step in the test program. In this discussion it will be assumed that a detail load spectrum representing the best estimated service usage of the aircraft and including all dynamic effects has been developed and an initial fatigue analysis of the vehicle conducted to locate the potentially critical fatigue areas. From this point the selection of the test load spectrum is made. Considering the state-of-the-art today there is no one method or procedure for test spectrum development which can be recommended for all cases. The overall objective is to come up with a test spectrum that will allow the maximum simplification of load application and load programming and yet produce acceptable accuracy and reliability in results for the specimen in question. To accomplish this simplification, some form of the Cumulative Damage Concept must be utilized and an equivalent damage test spectra developed. Within the realm of this equivalent damage spectrum we have two extremes; the simplest approach being a straight single load level test for one type of loading, the other extreme being an elaborate duplication of most of the loading conditions occurring during a composite ground-air-ground mission profile, applied in an ordered or randomized sequence. In considering to what extremes to go to in the selection of the proper equivalent damage spectra, careful consideration should be given to all of the factors involved. It is felt that under certain circumstances there is still a place for the single load level test concept providing it is properly used and the results evaluated accordingly. Where this simplification will not suffice, the development of the spectra will have to be carried further. The point to be made is that many compromises will have to be made in reducing the detail "design" or "initial" service load spectrums down to a load spectrum that can be applied in the test laboratory within the practical aspects of test techniques. These compromises will each have to be carefully considered and the best overall compromise reached. It is imperative that these compromises in load spectra be adequately and carefully documented so that any need for possible re-evaluation of their effect may be readily accomplished.

TEST LOAD SIMULATION

In the area of test load simulation the WADC feels the most important aspect is the completeness of the simulation throughout the test article. As stated previously, at the start of the test program, certain areas of the aircraft have been selected by analytical means as being the most potentially dangerous in a fatigue sense and obviously these areas and their respective load patterns will key the test load applications. However, one of the major reasons for a full scale test program is to uncover any other areas that may be fatigue critical and the only way this may be successfully accomplished is to maintain the applied loads at their correct, or at least very close to correct magnitude and direction throughout the test article. As in the case of test load spectrum development, there is no one procedure or approach to the load simulation problem which should be applied across the board in all instances. Certain configurations such as high aspect ratio wings where chordwise center of pressure variations for different conditions have relatively small effects, lend themselves to a fairly simplified loading arrangement,

involving relatively few load application points and still provide for quite complete and accurate load duplication throughout the component. On the other hand, a delta wing configuration wherein large excursions in chordwise center of pressure are encountered from condition to condition, with the possibility of net load reversals across a chordwise section, will require a considerably more complex loading arrangement employing numerous load points. The point the WADC wants to make is that the first consideration in load simulation should be completeness of load duplication throughout the vehicle so that failures which might turn up in any location can readily be evaluated and considered as valid data. Within this limitation every advantage should be taken of load simplification to satisfy the practical aspects of available laboratory techniques. As for load application methods, the Center feels normal structural test load methods should be employed in most instances, utilizing bonded load attachment points rather than built-in mechanical connections to preclude local fatigue failures induced by the non-representative local structure, and/or the high local stress distributions.

SUPPLEMENTAL TEST SPECIMEN REQUIREMENTS

The term "test specimen" for this discussion will include the range from the relatively small specimens such as joint specimens, special fitting specimens, etc., up to and including complete components. As indicated in the title of this section, these specimen tests are considered supplemental tests. Because of the extreme difficulty of reproducing the actual boundary load conditions, in both magnitude and direction, on a specimen test it is felt that specimen tests by themselves cannot be relied upon to fully define the fatigue life of a complete aircraft. However, these specimen tests do play an important part in the overall test program and full advantage of their value should be taken. Some of the major uses of specimen tests advocated as part of the program are as follows:

a. When the first preliminary fatigue analysis is made to define and locate the general areas and components that appear fatigue critical, S-N data for the area or items in question must be utilized. If no existing data completely fits the item in question, and it most often does not, some small specimen tests should be run to obtain this S-N data.

b. Once a major failure has been obtained in the full scale test program it is often desirable to conduct several specimen tests of the area in question to provide back-up data - particularly on scatter - since only one full scale test is normally conducted. This is especially true if the full scale test is conducted with a fairly simplified loading spectrum.

c. Also, once a failure has been obtained in full scale test programs at some number of cycles less than required, the rework or redesign of the area should be developed and preliminary verification of its adequacy obtained from specimen tests before the full scale article is reworked and testing continued.

d. In spite of every attempt to obtain a full scale test article that is completely representative of all or at least the majority of the tactical vehicle involved, there are instances where the compromises required from overall program requirements preclude this. This may occur from the desire to take an early block aircraft in order to accomplish the program expeditiously, or there may be a considerable number of two or more configurations - both of which need certification

with only one test article authorized. In this case, the only recourse is to cover the alternate configuration with a specimen test. Under these circumstances the specimen required may well be a large section of the aircraft - even a complete component. In utilizing this procedure the "bugaboo" of boundary condition simulation must be carefully considered. The specimen should be large enough that jig effects can be eliminated or minimized. In addition, special instrumentation consideration should be made on the full scale test article to allow the acquisition of sufficient data to allow proper simulation of the boundary loads.

FAILURE INSPECTION TECHNIQUES

The inspection techniques utilized and the results obtained, in the full scale fatigue test program are of paramount importance in a successful test program. Every effort should be made to utilize any and all inspection means available during the full scale test program to - (a) locate and define any failures as near inception as possible, and (b) to minimize the danger of major failure and consequent loss of the test article. Although direct visual inspection is desirable in most instances, care must be taken in removing major structural access panels so many times that their normal load distribution characteristics are significantly altered. If at all possible, inspection by this means should be regulated so that removal of these access panels corresponds roughly to the normal service removal rate. In other words, if a given panel is normally removed during a 50 hour inspection, this panel should be removed for laboratory inspection after each simulated 50 hours of service life. It is apparent that some areas are going to be completely blind for visual inspection. In this case if the area lends itself, a remote crack detection device such as the brittle wire technique, should be employed.

The second phase of the test article inspections, and a very important one that tends to be overlooked, is the development and prove-out of specific service inspection techniques. In instances where a fatigue crack is located in the test article, possibly by a partial tear-down inspection, it is felt that the component should be reassembled to its original configuration and with this fore-knowledge of the exact crack location and orientation, an attempt made to relocate it with more simplified service inspection techniques. An excellent example of this procedure would be the establishment of portable X-ray techniques for a particular location. In this manner the service inspection instructions could give the potential crack location, exact film and X-ray head orientation and correct exposure data. Another example is the case of borescope inspection of a large cavity for a small crack. Without specific instructions as to where to look and how to look for the crack, this can be a tedious process with consequent loss in reliability of results. It is believed these examples illustrate the type of effort that should go into the development of service inspection techniques and since this is one of the most valuable by-products of the full scale fatigue test program it is felt that the necessary time and effort to perform this function should be included in the test program.

TEST DATA INTERPRETATION

This area of the test program is one of the most difficult and most important facets. Since the full scale test program produces the actual number of finite load applications that the structure is good for, this is the one number in the entire

program that is factual. In some manner this number of test load applications must be related to hours of service use to obtain the final end result of the program. In all cases this conversion to service hours will include several "factors" to cover certain phenomena which are known to exist but for which we have a dearth of finite data. The two items which have the greatest input into these "factors" are the make up of the loading spectrum and the scatter inherent in fatigue test results. These "factors" may be employed at various stages of the fatigue evaluation program. The initial load spectrums may have certain factors applied to account for aircraft utilization that past experience indicates should be considered but the actual planned utilization does not include; some load applications of limit load may be interposed at intervals during the test program to cover the occasional overshoot that is known to occur but is not in the planned usage; various statistical deviations to account for scatter may be applied to the early input data in one case, and in another case they may be accounted for as a factor on the number of load cycles applied in the test program. The fact that these various factors are utilized in putting all the members together to come out with a service life prediction are not in themselves so much a problem as is the fact that it must be clearly indicated when, where, and how they were applied. If this is not done it is possible, and quite probable, that the full scale test results might be mis-interpreted especially by personnel not immediately connected with the program. In order to provide for uniform interpretation of the full scale test data, it is felt that in those cases where all of the above-mentioned "factors" have been considered and applied to the input data of the test load spectrum, the test results should be presented in terms of service hours. However, in those cases where the full scale test results must be further factored to obtain the final service hours, these test results should be presented in terms of test load application or some similar term with no mention made of "hours".

FUTURE ASPECTS OF TEMPERATURE SIMULATION AS AN ACCUMULATIVE EFFECT

There is considerable lament that the state-of-the-art of fatigue evaluation (meaning fatigue effects due to load) is woefully lacking today. When one superimposes the problems of elevated temperature operations with its time dependent accumulative effects upon the repeated load picture, the situation is infinitely more complicated. Today we have no clearly defined analytical or test procedures to assess these added effects. It has been generally agreed that for aircraft up to and including the Mach 2 category, the overall aircraft temperature ranges are not high enough to warrant the added complexity and difficulty in considering these temperature effects in the fatigue life evaluation. However, as we enter the Mach 2.5 and higher flight regimes it appears that we must attempt to evaluate these elevated temperature effects. Since we are even more limited in our thermal analytical capability, it appears more reliance will have to be placed upon test data in evaluating the elevated temperature effects on the fatigue life. Unfortunately we are likewise more limited in our test capability (both technique and approach) in elevated temperature simulation.

Basically, there are two elevated temperature effects on the fatigue characteristics that must be simulated. These are the effects on the basic material fatigue properties when a portion of the repeated loads occur with the structure at temperature, and the accumulative effect such as creep resulting from various exposure times under elevated temperature conditions.

Considering today's state-of-the-art, and unless major break-through results from some of the current R&D effort in this field, it is believed that the elevated temperature test simulation in the fatigue programs for the next few years, including the Mach 3 bomber and interceptor vehicles currently planned, will have to be conducted upon a broad compromise basis. It is felt that the simulation of repeated load at temperature for its effect on the fatigue properties can readily be handled with the largest compromise arising from the problems of how these heat cycles are distributed within the test spectra. For simulating and assessing the creep or related effects due to the time at temperature, considerable compromises will have to be made and probably the entire effect accounted for by the application of "ignorance factors".

Editorial Note: Attention is directed to the editorial policies presented in the Preface which were followed in editing the impromptu questions and answers of the session.

QUESTIONS AND ANSWERS FOLLOWING MR. WARD'S PRESENTATION

CHAIRMAN, MR. SMITH:

Thank you, John. I think it was a very interesting paper. We have time for about one or two quick questions from the floor, so does anybody have anything that they'd like to bring up?

MR. LOWNDES:

Mr. Ward, could you give me the relative magnitude of the maximum single load level loads versus the maximum load to the two random loads? Were they anywhere near equivalent?

MR. WARD:

The maximum load level of the constant level test was 1.75. The maximum in the randomized step test was in the vicinity of 2.4 G, alternating movement.

MR. LOWNDES:

I wonder if the higher load level had been used in the single load level test would we not possibly have developed the same cracks in area C?

MR. WARD:

This is a possibility.

Editorial Note: End of questions.

REMARKS FOLLOWING DR. BARUCH'S PAPER

CHAIRMAN:

A very stimulating talk on fatigue and now we are better acquainted with what the problem really is! However, I can't quite agree with some of the conclusions and that is about the good judgment that you use in the modifying structures. I know a lot of cases where we beef them up and we make them worse!

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stress. We have here a riveted joint made out of vinyl plastic - we stretch it a little - and we see that the first rivet in this joint takes most of the load. This is a simple demonstration of photo-elasticity. Now, if we pull it slowly we'll see how it loads up right at the end. The next specimen will show us what happens when a shop man tries to cover up some of the bad holes. This is nothing but a piece of stretched vinyl between two polaroids. In the case where we have an elongated hole and the load is normal to the direction of the major axis of the ellipse we have a higher stress than when we have the load parallel to the major axis of the hole. Now, see what happens when there is a round hole. We'll now get on with the questions.

Statistically speaking, the probability of finishing all the questions is quite remote, so I will shuffle these cards up and we will just pick them off the top and let the chips fall where they may. Somebody might be lucky and not have to answer the most embarrassing ones! The first one is addressed to Mr. Watson. Mr. Shuler of Lockheed Aircraft Corporation asks: "Would you comment a bit on the connection of the B-47 failures under test and service conditions?"

MR. WATSON:

Well as mentioned in the talk, or perhaps I might have missed the point or not emphasized it strongly enough, the cyclic tests on the B-47 were conducted as a result of fatigue failures in the fleet. The airplanes were repaired with fixes at the critical joints and then were cyclic tested. They essentially were cyclic testing the repairs as well as trying to ascertain if any other critical areas would show up. I believe that's probably about the best answer I can give.

CHAIRMAN:

The next one is addressed to Mr. Huston from Mr. O'Brien of the Wright Air Development Center: "On your C-46 program was an attempt made to analyze static test data and correlate it with the fatigue test results. Do you think this is feasible?"

MR. HUSTON:

Well, we have some static test data on the airplane, essentially strain gauge measurements were made. I think the fatigue correlations which you have seen here today were essentially based on the measurements of load factor and I think the detail stress distribution is quite another thing entirely. I don't know how feasible it is, in other words.

CHAIRMAN:

I would say that neither do the rest of us! I for one wouldn't say that I could correlate this at all. The next question is addressed to Mr. Shuler from Mr. Moseley, San Antonio Air Materiel Area: "Has the principle of fail-safe design been successfully applied to a fighter or interceptor type aircraft?"

MR. SHULER:

To the best of my knowledge, there has been very little fail safe design

incorporated in the higher performance military aircraft. We are incorporating it in our little Jet Stars.

MR. MELCON:

I could make a comment on that. We made a fatigue test on the wing of the F-104. This was part of the initial structural program. It didn't result from any trouble. We continued to load the test article after fatigue failure. The wing fuselage intersection has five spar fittings in it and with at least 40% of all these attachments from the wing to the fuselage out, it is still capable of carrying limit load, so we consider the F-104 wing in its most vulnerable place to be highly fail safe.

CHAIRMAN:

The next question is addressed to Dr. Baruch: "You mentioned that there is a lack of scaling law in fatigue testing. I would like to ask you if you have a scaling law in the random noise field; more specifically, could you reproduce the noise field of a jet engine by some other means?" (Editorial Note: author is unknown)

DR. BARUCH:

Before I answer this I ought to apologize. Apparently I stepped on some toes when I spoke. Believe me it was not my intent. I do realize that engineers are working hard and are performing quite remarkable solutions to the fatigue problem. I was simply trying to stress the need for basic research for those problems. You are going to find them coming up again in the next five or ten years from now.

"Can you reproduce the noise field of a jet engine by some other means?" In the basic problem which I tried to outline here which is going from a sound field to a stress pattern in a structure, realize that to make this transition as long as we're in pretty much the linear region in material, the stress pattern resulting from the sound field is going to be pretty much independent of the amplitude in he sound field. Changing the decibel figure of the sound field is not going to change the spatial distribution of stress appreciably, so that we can produce jet engine noise using loud speakers or anything like that in order to go from the sound field to the stress pattern in the structure. For fatigue work where we are trying to actually produce the failure of the structure by a sonic field, we do not now have sound sources which are intense enough to put out the kind of acoustic power that a jet does. There is some effort being made now to build such sources but they are not yet operational. So for the actual fatigue performance, here again our scaling limitation is the same as the fatigue man's scaling limitation. We just can't make the sound 10 DB lower and expect the thing to last say point one five times as long, in a test. We don't have that kind of a scaling yet.

CHAIRMAN:

The next question is from Mr. Aubrey of Canadair, Ltd., and is addressed to Mr. Watson. "By applying periodic 90% limit loads during fatigue testing were you not concerned about preloading effect, if so what allowance was made for them? In using S-N curves to derive a life, what equivalent K_t did these represent?"

MR. WATSON:

To the first part of the question: in applying periodic 90% limit load applic- actions were we concerned about preloading, I would say that yes, we were. However, we feel that the frequency at which we applied this 90% limit load was fairly representative of actual airplane usage and felt that this being realistic, we would live with the situation that way.

"In using S-N curves to derive life, what equivalent K_t did these S-N curves represent?" Well, essentially they represented K_t factors varying from around three to six.

CHAIRMAN:

Thank you. Now the statistical shuffling of the cards doesn't seem to be working out too well. I think maybe I'll put these aside and see if I can't get somebody else that hasn't had a question so far. Is that dishonest? Here's one for Mr. Ward from Mr. Melcon of Lockheed Aircraft Corporation: "In applying cumulative damage calculation, were stresses based on calculated values or strain gauge values? After cracking started, was the stress adjusted for load redistribution?"

MR. WARD:

The cumulative damage calculations were done on the basis of applied alternating load factor. We established the S-N curve or load life time curve with constant level tests for given values of alternating load and then ran the randomized tests at given values of alternating load factors. Our over-all ratios were based on the alternating load factor values. We did not reduce the load after the crack initiated. We applied the alternating load factor through the end attachment after one G or whatever it might be. These were monitored by bridges on the wing panel.

CHAIRMAN:

The next question is addressed to Mr. Melcon from Mr. Bouton of Norair: "What kind of specimens will you load with your various types of loading sequences? Also, how many samples for each type?"

MR. MELCON:

The specimen that is to be used in this initial investigation will be a coupon type three inches wide, with an eliptical type hole in it representing a K_t of about four and one of about seven. These will be made out of 7075 material and at least five specimens will be tested under each loading condition.

CHAIRMAN:

This is one I'm afraid of and I thought something like this was bound to come up. It is for Dr. Baruch from one who signs himself "a former Convair client": "Certainly no group of highly trained specialists has ever been given a more effective treatment than

you have given this group. Where did you go to medical school?" "Are you a direct descendent of Bernard Baruch so that you can afford to say what you think?"

DR. BARUCH:

So far I've gotten two questions and a box with a gold brick in it!

I think the situation we are facing right now is such that I can't afford not to say what I think and I think the same goes for any of the scientists in the group here. Some people have taken the affronted feeling that I have said that engineers don't know what they're doing. This is a wrong statement. Engineers right now are fixing aircraft by a system of patching. It is the only thing that we have the scientific background to do at the present moment. Engineers are going to continue patching aircraft but when we start talking about a research program into fatigue, let us not confuse a research program with a development program designed to give us a faster method of deciding where to patch an aircraft. This is not the fundamental goal of research. To those who answer that the more fundamental type of research will not pay off for five or ten years, I can only reply in the words of the Colonel in the English army who was in India at one time and decided he wanted a particular type of tree for his compound. His houseboy said "Well, Colonel, that tree takes two hundred years to grow to maturity," the Colonel answered "All the more reason to plant it now." I think the same is true of our research program.

CHAIRMAN:

Here is a question addressed to Mr. Ward or Mr. Huston from Mr. Sweet, Naval Research Laboratory: "Did either of you gentlemen measure the lengths of the cracks which you found? If so, do you have plots of crack length versus number of cycles?"

MR. WARD:

The crack lengths were measured for each crack in the test. Most of these cracks as you know didn't grow to any great extent but the critical crack propagation curves are available. These data are found in NASA or NACA Reports TN-3190, TN-3847, and TN-4132.

CHAIRMAN:

Here is one directed to Mr. Wise from Mr. O'Brien, Wright Air Development Center: "Who manufactures porcelox ceramic?"

MR. WISE:

I don't know just now. I know we've been testing a lot of Solar ceramics and I don't know just who this tradename applies to in the manufacture.

CHAIRMAN:

This question also from Mr. O'Brien of the Wright Air Development Center:

"How many lamps and what rating are in your tension pad heater?"

MR. WISE:

There are four 500 watt infra red lamps in this tension pad heater. I can give you more detailed information on that if you want to see me after this meeting.

CHAIRMAN:

The next question is from Mr. C. D. Little, Convair, Ft. Worth. It is addressed to Mr. Watson: "Relative to either the B-52 F or G, what is the ratio of cycles in the test spectra to significant damage cycles in the detail load spectrum?"

MR. WATSON:

I think at this time it might be well to recognize my co-author on this paper, Mr. Gore. I would like to ask him if he wouldn't care to comment on this.

MR. GORE:

I might be willing to comment if I understood the question! I'm afraid I don't understand the question. Would you mind amplifying it?

MR. LITTLE:

In your loading spectrum there is a large number of cycles. Many of these are below the endurance limit of the S-N curve and these are thrown out. Of those left that do significant damage, these have been further condensed for testing purposes, the question is, what is the ratio of the condensed number to the uncondensed number?

MR. GORE:

By the "condensed" number you mean those which were represented and by "uncondensed" those that were thrown out?

MR. LITTLE:

Suppose you ended up with a million cycles of damage could you condense this to 200,000 or 100,000 for testing purpose to do the equivalent damage?

MR. GORE:

Well, this comes out more or less automatically after having selected the stress levels for your testing in determining what levels to throw out - of course in your history you can say that there are a large number - approaching infinity - of significant cycles. We over-simplified the spectrum then and we don't recognize for damage calculation any stresses which fall below a S-N curve actually on the order of ten to the eighth cycles. The levels at which we test are largely dependent upon time. I hate to say this but this is the case. The stress cycles that are significant, the threshold of our damage calculation, is in the order

of one or two thousand psi. The stress level' .s we applied the load to the test specimen - in the case of the B-47 - our lowest stres- .vel was on the order, of, three or four thousand psi. This varies. I hope that ε ɜwers your question.

CHAIRMAN:

The next question is from Mr. O'Brien of the Wright Air Development Center and is directed to Dr. Baruch: "Do you think the use of acoustic properties could be used to predict failure in static testing?"

DR. BARUCH:

I spoke to Mr. O'Brien briefly on this so I think maybe I ought to amplify this question a bit. I think Mr. O'Brien is asking "Can acoustic techniques be use in monitoring -" this is not producing failure by acoustic means but using acoustics as a monitoring or instrumentation technique for monitoring the onset of incipient failure, as I understand the question.

Now there are some techniques that are being developed in the laboratory for crystal dispersion analysis. They are very diaphanous at the moment. There are echo ranging techniques used for crack detection which as you well know have the limitation that the crack may be - must be - of appreciable size compared to the wave length. There are other dispersion techniques which can be used in many crack specimens, specimens with many cracks in it, to get the frequency dispersion produced by the sample but at the present time the type of failure that we are concerned with which are micro-cracks, slip planes, things like this, are just plain too small to be detectable with the kind of acoustic techniques that I know about at the moment. Maybe there are others in the audience who are more familiar with the variable frequency ultrasonics who could answer more positively on this method.

CHAIRMAN:

The next question is directed to Mr. Watson from Mr. Freyre of Lockheed Aircraft, Georgia Division: "Has there been an experimental evaluation of the dynamic magnification factor for gust loading on the B-47? If so, what is the general magnitude at the root, 50% wing semi-span, and at the wing tip?"

MR. WATSON:

There has been some determination of the dynamic magnification factors on the B-47 and as Mr. Jackson presented in his paper, current programs are in progress to further find some of these values. Mr. Freyre here asks for some values at particular stations. I think the only answers I can give here are that they are around one point one at the root and around one point three at the midspan and I believe approaching two out near the tip, some preliminary values. But Mr. Jackson in his paper I think indicated this pretty strongly - that there is considerable data yet to be obtained.

CHAIRMAN:

The next question is directed to Mr. Melcon and is from Mr. Schuette of Dow Metals: "How do you feel about the necessity for incorporating representative rest periods in a load spectrum? If they are necessary, what is needed so that we can learn how to do it in an accelerated fashion?"

MR. MELCON:

I think if I answered the first part I'd contradict myself on the second part. I really don't know the answer to the question because we all know that in a fast fatigue test we are not incorporating some of the environmental factors which would occur in the aircraft in service. How we would go about introducting these effects into a fatigue test or how we could give it a little rest and then give it a squirt of NaCl or something like that and say "this is now duplicating" what it gets in service, I'm not prepared to say. Maybe somebody else has an idea.

CHAIRMAN:

Well, I would say it is a very smart man who admits that he doesn't know. I think we will go on to the next question. It is addressed to Mr. Shuler and it is from Mr. Moseley, San Antonio Air Material Area: "What is the approximate weight penalty in per cent of total weight to achieve 'fail safe' design on an aircraft the size of the C-130?"

MR. SHULER:

We did just a little study of this to sort of get the feel of what we were paying in the way of a weight penalty. Now do you mean the take-off gross weight here?

MR. MOSELEY:

Yes, take-off gross.

MR. SHULER:

I'd say with the C-130 that it was considerably less than one per cent, something in the order of about three-tenths of one per cent.

CHAIRMAN:

This next one is to Mr. Melcon from Mr. Bouton of Norair: "In your Figure 3, you show ground loading as negative acceleration. Do you really mean acceleration or should this have been negative B.M." Now it says B.M. but being engineers that stands for bending moment.

MR. MELCON:

I am sure it is an error.

CHAIRMAN:

All right. You see I've been cheating you here. I've held out a lot of cards and the reason being that they've all been addressed to Mr. Watson, so if anybody gets stuck, why the rest of you could probably contact him personally after the meeting. We will have a few of them here, however. This question is from Mr. Hackman of the Wright Air Development Center: "In your opinion, could full scale fatigue testing be reasonably adapted to include static test conditions and therefore replace static testing with the saving of one full scale test specimen?"

MR. WATSON:

It is my opinion that full scale fatigue testing cannct be adapted reasonably to include the static test conditions we're after. I think two different things are included here and I think in this case, it has been our experience, at least in the B-52 and in the B-47 programs that different areas are probably involved and furthermore, we would stand an awful good chance of losing a specimen one way or the other to try to get both tests run with the same specimen. I don't know whether this answers the question or not but in my opinion, I don't believe it's feasible.

CHAIRMAN:

It looks as though we have a few more questions that we will not be able to answer at this forum. They are all addressed to Mr. Watson, however, and so I will just hand them to him. Those who felt that their questions weren't answered may contact him after the meeting. I would like to express my appreciation to all of the speakers at this afternoon's program for a very enlightening and entertaining program. Again thanks to you and to WADC and ARDC for having me on the program.

CLOSING REMARKS

By

Major General Stanley T. Wray, Commander

Wright Air Development Center

When the idea of this symposium was first presented, there was some question as to whether we could ask our speakers on such short notice to assemble the type of information that we wanted to have presented to you. The fact that this auditorium was filled up to the last break in this final session is a strong indication that our speakers have successfully performed prodigious efforts in providing important material; the information will assist all of us in improving our understanding of the theoretical, engineering, and managerial problems involved in the fatigue field. Also, it has been particularly pleasing to have as our session chairmen, men who stand out in the individual structural integrity research areas which were the basis of the program.

We in the military, who live day-to-day with the fatigue problem and have to "keep 'em flying," have first hand evidence of how critically serious this over-all fatigue and materials problem is in terms of those weapons systems which constitute our first line of defense. It has, therefore, been very pleasing to us to have such a wonderful response. I believe our attendance has been in the order of some six hundred people, not only from the government agencies but from the aircraft industries, the airlines, the metals industries, and the academic world.

As I said on the first morning of the symposium, I hoped that people would speak freely and that you would not harass the speakers until they got a chance to express opinions which didn't always agree with your own. I understand from the boys in the back room that this has been one of the outstanding marks of this meeting.

We hope that you people who have attended this symposium feel that your attendance has been profitable. We hope that you will carry the message back to your co-workers, especially in the design areas.

When the idea of the questionnaire was first proposed to me, I agreed with Colonel Taylor that this might be a start towards supplying one of the missing links in all of these large symposia; that is, the individual attendee's immediate feedback of ideas. We want to stress that this questionnaire was not an effort to get a program because we know that you people couldn't give us a program on such short notice. But your individual ideas, when they're presented "on the spot," mean a lot to the organizers of symposia who, in turn can then make them available to all by incorporating such contributions in the minutes or in the proceedings.

I understand that the results of the questionnaire were presented to you this morning. We were very happy with the response we had from you. Your comments have given us

clues as to additional effort my people must now embark upon. I realize that many of you wished to participate in the total symposium effort before getting your thoughts lined up to give us clues as to what might be good for the future. I hope as you travel home tonight, you can organize some of these thoughts and send them back to us. We welcome them.

As General Schriever indicated in his keynote address, the top commanders in the Air Force are supporting this effort of ours to reduce not only the incidence of fatigue failures in our present aircraft but also to prevent the problems of structural integrity from becoming a bottleneck in the development of aerospace vehicles as men go higher and faster and farther.

We're all interested in your thoughts concerning this symposium and the facets of structural integrity which might be included in another similar meeting, because this one has been held on somewhat short notice. We know that some of you people would have liked to have presented papers. We wonder if there isn't a requirement for another symposium of this type in a year, certainly within two years.

All of us who have participated in organizing this symposium thank you for your attendance. We appreciate in particular your undivided attention during the past three days.

The first symposium on Fatigue of Aircraft Structures stands adjourned.

SUMMARY REPORT
STRUCTURAL INTEGRITY RESEARCH PROGRAM QUESTIONNAIRE

By

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AIRCRAFT LABORATORY
WRIGHT AIR DEVELOPMENT CENTER

INTRODUCTION

The idea of a questionnaire to be filled in by symposium attendees was first conceived by Colonel John P. Taylor while plans were being formulated for the symposium program. Adoption of the questionnaire was pushed since it appeared to offer an excellent vehicle for the feedback of ideas and comments by those attending and because it would promote the interchange of ideas and thoughts between conferees. The potential value of suggestions to improve government planning of the structural integrity program and its effectiveness in helping those engaged in resolving the fatigue problem was also recognized.

Some of the factors which made questionnaire contributions potentially valuable were: the rather short time in which launching of a structural integrity program to resolve weapons systems fatigue problems took place; the inevitable difficulties in achieving a balanced program within limitations too numerous to mention; and the many aspects governing the effectiveness of the program such as compilation and timely distribution of data and other types of appropriate information. Thus, it was thought likely that some good would result from such a survey - even if it only provided the attendees an opportunity to express their points of view to the symposium organizers.

While having the questionnaire reviewed with respect to ARDC research, procurement and legal policies, it was found that many purposes could be fulfilled by such a simple questionnaire form. For example, constructive ideas might be obtained not only from the answers, but from efforts associated with preparation and analysis of the form. It was felt that these ideas would be useful in guiding and encouraging research so as to enable development of advance weapons systems in a more orderly and timely manner. In another sense, it was determined that the names of individuals and organizations obtained in the survey could provide helpful data on procurement sources for structural research; and, again, the exercise and its results could provide an input to better technical research area breakdowns for procurement purposes.

The above examples point out, on the one hand, the danger of trying to make such a survey serve too complex an objective. On the other hand, they point out the usefulness of a questionnaire in providing some helpful ideas and suggestions

in the sense of making the effort worthwhile. The latter viewpoint is more simple, if not better, and the one most representative of the approach in this particular effort. The circumstances of the timing and inexperience in preparing such forms make the simplest possible viewpoint mandatory. There was no intention at any time of using the results as a definition of, or even a substitution for a well planned structural integrity research program.

QUESTIONNAIRE FORMAT

A brief review of the questionnaire format, see Exhibit 1, is now in order. Basically, the information desired from each attendee was solicited by Questions 7 and 8, namely, "What structural integrity research, in the opinion of the attendee, is most needed----?" This naturally opened up a host of problems - for the attendee - to which there were several solutions. But essentially, and first of all, a brief description, in a nutshell, of any specific, critically needed research item uppermost in the mind of the attendee was called for. More general answers to these questions provided useful information. The cost and time data requested to accomplish the proposed research projects merely served to define further the research effort suggested.

In connection with Questions 7 and 8, as well as with other questions, an outline was given by which suggested research could be classified as shown on Page 1 of the questionnaire. The formulation of this outline posed a problem. The problem of a short, general outline covering all possibilities was difficult to achieve and difficult to understand; a specific, narrowly defined outline was too long, and by far the most difficult to make all inclusive. Purely by chance, an outline in the general form, but open ended to allow infinite variation was adopted. This satisfied both schools of thought and restrained no one except within the broadest outline, i. e., the four major categories as shown on Page 1 of the questionnaire had to be fixed, at least for later convenience in analysis.

Questions 1 through 6 were designed primarily to provide an introduction to the questionnaire to obtain a rough idea of the technical backgrounds and interests of the attendees who participated. There was no intention that the answers to these particular questions would have far reaching statistical significance.

Question 9 asked for the names and affiliations of individuals, or the names of organizations known to have demonstrated capabilities in some area of structural integrity research. Those not likely to be known to personnel of WADC were of particular interest. The importance of this question may not be evident to those unfamiliar with the heavy workload borne by many of the recognized experts; but there is a need to find new sources of capable people for structures research.

It will suffice to say that ample space was provided in the questionnaire for additional comments and ideas not included in the first nine questions.

SIDELIGHTS

The questionnaire was distributed before lunch on the first day of the symposium with an explanation of the format. Attendees were particularly encouraged to submit "off-the-top" ideas on important research jobs needed to support improved structural integrity designed into flight vehicles. Fifty-four questionnaires were returned before the start of technical sessions on the morning of the second day. Early returns were limited by a fully scheduled first day agenda.

An early return of completed questionnaires was requested in order to report the general results of the survey before adjournment of the symposium. Another reason for requesting the quick return of the questionnaires was to obtain the desired "off-the-top"reply. At any rate, the general results contained in the questionnaires on hand were reported at the start of the third day sessions.

A statement was included in the questionnaire on Page 1 indicating that proprietary information should be submitted through normal government agency channels. In order to allow maximum freedom to the attendee in transmitting ideas within this restriction, a ground-rule was established whereby only the general results obtained in the survey would be published. In accordance with this ground-rule, detail data relative to the research projects and other ideas which were submitted by attendees must be withheld from this report. The detail comments and data received in the survey will be used by government agency personnel to improve planning of the structural integrity program.

GENERAL REMARKS ON RESULTS

From the broadest viewpoint and in fairness to both sides, the results of the survey revealed that no one really mounted the platform to say what should be done with as much emphasis as those who did mount it to say what should not be done. But there was a lot learned by the authors of this article. We failed to learn more only because of the lack of time during the symposium and not because of a lack of original ideas submitted.

QUESTIONNAIRE RESULTS

The over-all response to the questionnaire is given in Figure 1. Of some 450 questionnaires distributed, a total of 108 were returned. Some 268 items of research were suggested; 66 of these items appeared to be about the right number to consider as duplicate items, leaving a grand total of 202 different, unique research projects to be considered. The total cost of these unique items of research was estimated at $74.6 million. It was estimated that it would take five years to do 90 percent of the suggested research.

Before proceeding further, it is worthy of mention that only a small handful of experts took advantage of the open ended outline on Page 1 of the questionnaire to improve the outline or better classify their particular proposal. The following

are examples of the classifications which were entered: sonic fatigue; mathematical analysis of structures; solid state mechanics; damage criteria; dynamic loading effects on materials in complex stress states; tests of parts and assemblies removed from service; crack propagation studies; basic fatigue mechanisms; structural strength after fatigue appearance; education; and new materials and techniques. Careful analysis indicates that many of these can be visualized as being covered by the outline already provided. No comment was received concerning a different or better approach to the outline.

The technical backgrounds and interests of the individuals who returned completed questionnaires are contained in Figure 2. This information was solicited by Questions 1, 2 and 3 which covered the organizational affiliation, job function and primary flight vehicle interest of the attendee. Of those who returned the form 53% were from the aircraft industry; 50% had a primary interest in aircraft; 44% were engineers; and 25% had research positions. Some 47% chose Structural Life Design in Question 4 as their special field of interest, while the other 53% split three ways almost evenly among the Materials Research, Fatigue Testing and Fatigue Loads categories.

All of the above data indicate that the results were largely submitted by people from the aircraft industry, including the missile area as well as the engine and propeller manufacturing areas, who are specialists in some type of structural life design. Figure 2 also shows that there was an almost even split (24 to 26.3%) of non-government sponsored research among the four main categories of structural design, materials, testing and loads (Question 5). This research is being carried out or planned by the organizations with which the individuals are affiliated.

The response obtained from Question 6 concerning the research area in which the individuals thought their organizations were most capable and qualified for doing research, is summarized in Figure 3. These results show in a very general way that research capabilities are evenly divided over the four major areas of Structural Life Design, Materials Research, Fatigue Testing and Fatigue Loads.

Data on the number of research items suggested, their estimated cost and the time to carry out the jobs are presented in Figure 4 for each of the four major areas of research. Detailed breakdowns for each of these four major areas of research are given in Figures 5 through 8. Observed trends in these data are as follows:

The amount and cost of research suggested was considerably greater (by at least 50%) in the area of Structural Life Design than in any of the other three major areas of research (Figure 4). Further, in the area of Structural Life Design, suggested research was primarily, and equally, concentrated in the three minor classifications of Structural Configurations, Life Prediction Techniques and Criteria Formulation as shown in Figure 5.

Materials Improvement and Solid State Physics were the minor research classifications most favored (by a factor of about 2) in the Materials Research area which accounted for 39 of a total of 56 research items (Figure 6). Twenty-seven out of a total of 52 research items suggested under the Fatigue Loads classification (Figure 7) were for Measurement and Analysis. As shown in Figure 8, the numbers of research items suggested in the Fatigue Testing Area were about evenly divided (from 13 to 16 items) among four of the minor classifications with only 5 of the total of 64 items falling in the Facilities Equipment Area.

The above data provide no real clues as to the relative importance of research in the different areas. The consensus of opinion of the respondees, nevertheless, corresponds with the opinion of many of the speakers. That is, there is agreement that the most important problem area lies in the structural life design area. Pending further evaluation no attempt will be made to ascribe further significance to the above results.

Typical examples of the research suggested have been paraphrased and listed in Figure 9. Many of these and of the other unlisted research projects which have been suggested already are, or are planned to be, in the structural integrity research program. Particularly prominent was the recurrence of suggestions for the compilation and dissemination of worthwhile data related to fatigue design. A system of reporting the findings of detail case history studies, and related efforts to explain the details was suggested. Although there are projects in being, or planned, in the area of design practice manuals, their objectives can and will be improved on the basis of the aforementioned suggestions which were received in the questionnaires.

A closer look is also being given to the problem of worthwhile data compilation and dissemination. Final appraisal of the actual effect of the questionnaire results on the structural integrity program must await review by those technical experts concerned with the particular research areas involved. Whether the ideas that were suggested, which are not already in the program, are adopted will depend upon the judgment of interested experts and the priority and availability of funding. Areas of responsibility in so far as the various government agencies are concerned also will be considered.

A list of the individuals and organizations recommended as having the capabilities and qualifications necessary for structural integrity research (Question 9) is included in Figure 10. This information will be used in supplementing present listings on procurement sources for structural integrity research.

CONCLUSION

In general, the response to the questionnaire, although not overwhelming, was very good considering the factors involved as discussed earlier in the report. Approximately twenty-five percent of the questionnaires were returned.

Probably the most important reason for the low percentage of returns was that the attendees were kept too occupied by the full agenda. Provision of some free time during the symposium to fill in a questionnaire might be the answer to this problem in the future.

Another important factor contributing toward limited response was the newness of the concept of a total structural integrity program and the relationship of structural integrity research to this program in all of its ramifications.

On the basis of the discussions touched off and of the suggested research and valuable comments received, it is considered that our objectives were met and that the effort was more than worthwhile.

The results obtained in this first experiment with a symposium questionnaire indicate that the method could prove to be a valuable technique for large symposia. It is believed that other symposia organizers may realize many advantages, as we have, from the ideas and comments fed back by attendees.

EXHIBIT

SYMPOSIUM ON FATIGUE OF AIRCRAFT STRUCTURES

STRUCTURAL INTEGRITY RESEARCH QUESTIONNAIRE

This questionnaire was prepared to obtain your opinion as to what efforts you consider are most needed in the structural integrity research area. The Department of the Air Force and other interested Government agencies plan to review the information submitted and, where possible use it to strengthen the overall structural integrity research program.

The general results of this survey will be published in the Symposium Proceedings. Detailed results will be used in connection with Government program planning.

Careful completion of the form is essential in making the results of the survey useful. Proprietary information should be submitted thru normal Government Agency channels by means of unsolicited proposals, if you so desire (copies of ARDC Form 91 are available and will be furnished upon request for this purpose).

A breakdown of Structural Integrity Research Program Areas to which questions 4 thru 9 are keyed is provided below. Program Areas I thru IV and Sub-headings (a) thru (e) should prove adequate in the majority of cases. Sub-headings (f) and (g) should be completed if your suggested research projects cannot readily be classified within the defined areas. This outline is intended to apply for all types of flight and ground loads including sonic and thermal; also, programs related to aircraft, guided missiles and aerospace vehicles are implied.

STRUCTURAL INTEGRITY RESEARCH PROGRAM AREAS

(Refer to this code when answering the following questions)

I Structural Life Design
- (a) Structural Configurations
- (b) Life Prediction Techniques
- (c) Prediction Parameter Studies
- (d) Criteria Formulation
- (e) Design & Manufacturing Practices
- (f) ____________
- (g) ____________

II Materials Research
- (a) Unique Testing Methods
- (b) Materials Improvement
- (c) Application Techniques
- (d) Solid State Physics
- (e) Facilities; Equipment
- (f) ____________
- (g) ____________

III Fatigue Loads
- (a) Prediction Methods
- (b) Prediction Parameter Studies
- (c) Measurement & Analysis
- (d) Instrumentation; Systems-Data
- (e) Facilities
- (f) ____________
- (g) ____________

IV Fatigue Testing
- (a) Load Simulation
- (b) Combined Environments-Simulation
- (c) Effects of Parametric Variations
- (d) Instrumentation; Inspection
- (e) Facilities; Equipment
- (f) ____________
- (g) ____________

Complete each question which follows by checking or circling one or more answers, as appropriate, or by writing a brief statement where necessary to express your views.

1. My organizational affiliation is:
   - ☐ University
   - ☐ Res. Institute
   - ☐ Acft. Industry
   - ☐ Metals Industry
   - ☐ Government
   - ☐ Other

2. My primary job function is:
   - ☐ Research
   - ☐ Engineering
   - ☐ Test
   - ☐ Academic
   - ☐ Management
   - ☐ Staff

3. My primary interest is with:
   - ☐ Aircraft
   - ☐ Guided Missiles
   - ☐ Aerospace Vehicles
   - Other ____________

4. My special field of interest in Structural Integrity Research Program Areas is: (circle one)

   I II III IV
   (see page 1 for code)

   More specifically, it is in the area of: ______________________________
   (short title)

5. My organization is engaged in or planning non-government sponsored research in the following Structural Integrity Research Program Areas: (Include brief titles in "Remarks" on page 5, if desired)

   I II III IV
   (see page 1 for code)

6. My organization is best qualified and capable of doing research in the Structural Integrity Research Program Areas indicated: (Circle no more than a total of three of the a to g sub-headings; elaborate on page 5 under "Remarks" if desired)

   I a b c d e f g

   II a b c d e f g

   III a b c d e f g

   IV a b c d e f g
   (see page 1 for code)

7. In the areas in which I am most technically qualified, structural integrity research projects which I consider must be accomplished in the future with rough estimates of cost and time are:

(From the code or outline given on page 1 of this questionnaire, select and enter in Column 1 below the Structural Integrity Program Research Area which best fits each specific research item you enter in Column 2: i. e., enter III(b) to indicate a research item under the program area of "Fatigue Loads - Prediction Methods")

Research Program Area (Col. 1)	Brief Title and Description of Specific Research Items (Col. 2)	Estimated Total Cost (Col. 3)	Estimated Time in Years (Col. 4)

Note: The difficulty of estimating costs and time in relation to research is recognized; however, your best estimates will help define the size program you are suggesting.

8. I also consider that structural integrity research programs should be implemented in areas other than my own special field as follows:

(From the code or outline on page 1, select and enter in Column 1 the Structural Integrity Research Program Area which best fits the more specific program you wish to place in Column 2; indicate why you think the suggested program is required in Column 3)

Research Program Area (Col. 1)	More Specific, Brief Title of Suggested Research Program (Col. 2)	Basis For Suggested Program (Col. 3)	Rough Cost Estimate (Col. 4)	Rough Time Estimate (Col. 5)

9. An effort is being made to compile a master list of highly qualified individuals and organizations capable of participating in the structural integrity program. To assist this effort, the names of individuals and their organizational affiliation, or the organizations that I consider outstandingly qualified in the Structural Integrity Research Program Areas with which I am most familiar are:

Note: Of particular interest are those individuals and organizations that have a demonstrated capability in the structural integrity area but may not be known to interested government agency personnel.

Research Program Area	Names of Persons & Affiliation	Names of Organizations

10. Additional Remarks:

(Reference the question to which each "Remark" pertains, as applicable)

Optional

Name

Organization

Return completed questionnaire to the Registration Desk in the Main Lobby or at the entrance to the Grand Ballroom in the Dayton Biltmore Hotel by 9:00 A.M. Wednesday morning, 12 August 1959. A board of technical experts representing government agencies will review the forms received as of this time and will announce the more general results obtained before the end of the symposium.

NOTICE: This is not to be interpreted as a solicitation by the Government for bids or proposals for accomplishing research and/or development in the problem areas referred to herein. The Government is not obligated to enter into contracts for any research and/or development with any of the participants in this symposium or the companies which they may represent. By signature and/or submittal of this completed questionnaire the participant agrees that neither its accomplishment by him or utilization by the Government of any of the opinions or information contained herein will be made the basis of a future claim against the Government.

NOTES

FIGURE 1

SYMPOSIUM QUESTIONNAIRE

SUMMATION OF SALIENT RESULTS

Number of Questionnaires Distributed	450
Number of Questionnaires Returned	108
Items of Research Suggested, Total	268
Number of Unique Research Items	202
Total Cost of Unique Research Items	$ 74.6 million
Time to Complete 90% of Items	5 years

FIGURE 2

SYMPOSIUM QUESTIONNAIRE

TECHNICAL BACKGROUNDS AND INTERESTS OF CONTRIBUTORS

QUESTION NO. 1
Organizational Affiliation

Aircraft Industry	52.8%
Government	13.9%
Research Institute	8.3%
University	6.5%
Metals Industry	4.6%
Other*	13.9%

* Air Lines (2%), Instruments, electronics, etc.

QUESTION NO. 2
Primary Job Function

Engineering	43.5%
Research	25.4%
Management	15.9%
Test	8.0%
Academic	5.1%
Staff	2.2%

QUESTION NO. 3
Primary Interest

Aircraft	50.0%
Guided Missiles	21.3%
Aerospace Vehicles	18.0%
Other*	10.7%

*Instrumentation, electronics, metals, etc. for air and space vehicles

QUESTION NO. 4
Structural Integrity Interest Field

Structural Life Design	47.1%
Materials Research	18.5%
Fatigue Testing	17.6%
Fatigue Loads	16.8%

QUESTION NO. 5
Non-government Sponsored Research

Structural Life Design	26.3%
Fatigue Testing	25.1%
Materials Research	24.6%
Fatigue Loads	24.0%

FIGURE 3

SYMPOSIUM QUESTIONNAIRE

ORGANIZATIONAL RESEARCH AREA QUALIFICATIONS *

Structural Life Design

Structural Configurations	41.7%
Life Prediction Techniques	29.6%
Criteria Formulation	27.8%
Design & Manufacturing Processes	20.4%
Prediction Parameter Studies	10.2%

Fatigue Testing

Load Simulation	33.3%
Combined Environments - Simulation	27.8%
Facilities; Equipment	16.7%
Effects of Parametric Variations	15.7%
Instrumentation; Inspection	13.9%

Materials Research

Unique Testing Methods	32.4%
Application Techniques	24.1%
Materials Improvement	19.4%
Solid State Physics	10.2%
Facilities; Equipment	9.3%

Fatigue Loads

Measurment & Analysis	28.7%
Prediction Methods	21.3%
Instruments; Systems-Data	19.4%
Prediction Parameter Studies	14.8%
Facilities	5.6%

*Percentages are based on the total number of checks made for a particular category divided by the number of questionnaires (108) multiplied by 100. This indicates the percentage of people marking the particular category. All checks, even when more than three were selected, were counted.

FIGURE 4

SYMPOSIUM QUESTIONNAIRE

SUMMARY OF SUGGESTED RESEARCH - MAJOR AREAS

Research Program Area	Number of Unique Research Items Suggested	Cost In Millions	Time to Complete 90% of Items
Structural Life Design	74	$ 29.5	3 years
Materials Research	41	$ 15.1	5 years
Fatigue Loads	39	$ 12.8	3 years
Fatigue Testing	48	$ 17.2	4 years

FIGURE 5

SYMPOSIUM QUESTIONNAIRE

SUMMARY OF SUGGESTED RESEARCH - STRUCTURAL LIFE DESIGN

Research Program Sub-Area	Number of Items Suggested Total	Number of Unique Research Items Suggested	Cost In Millions	Time to Complete 90% of Items
Structural Configurations	21	18	$ 5.4	3 yrs
Life Prediction Techniques	23	13	$ 1.8	3 yrs
Prediction Parameter Studies	15	14	$ 6.8	3 yrs
Criteria Formulation	24	19	$10.4	3 yrs
Design & Manufacturing Practices	13	10	$ 5.1	2 yrs

FIGURE 6

SYMPOSIUM QUESTIONNAIRE

SUMMARY OF SUGGESTED RESEARCH - MATERIALS

Research Program Sub-Area	Number of Research Items Total	Number of Unique Research Items Suggested	Cost In Millions	Time to Complete 90% of Items
Unique Testing Methods	9	9	$ 2.8	2 yrs
Materials Improvement	20	14	$ 4.9	3 yrs
Application Techniques	8	8	$ 1.8	3 yrs
Solid State Physics	19	10	$ 5.7	5 yrs
Facilities; Equipment	0	0	-	-

FIGURE 7

SYMPOSIUM QUESTIONNAIRE

SUMMARY OF SUGGESTED RESEARCH - FATIGUE LOADS

Research Program Sub-Area	Number of Research Items Total	Number of Unique Research Items Suggested	Cost In Millions	Time to Complete 90% of Items
Prediction Methods	13	10	\$4.4	3 yrs
Prediction Parameter Studies	5	5	\$0.8	3 yrs
Measurement & Analysis	27	18	\$6.8	3 yrs
Instrumentation; Systems-Data	7	6	\$0.8	3 yrs
Facilities	0	0	-	-

FIGURE 8

SYMPOSIUM QUESTIONNAIRE

SUMMARY OF SUGGESTED RESEARCH - FATIGUE TESTING

Research Program Sub-Area	Number of Research Items Total	Number of Unique Research Items Suggested	Cost In Millions	Time To Complete 90% of Items
Load Simulation	16	11	\$ 1.3	3 yrs
Combined Environments-Simulation	14	12	\$ 6.6	4 yrs
Effects of Parametric Variations	13	11	\$ 5.5	3 yrs
Instruments; Inspection	16	10	\$ 2.4	2 yrs
Facilities; Equipment	5	4	\$ 1.4	2 yrs

FIGURE 9

SYMPOSIUM QUESTIONNAIRE

TYPICAL RESEARCH PROJECT SUGGESTIONS (PARAPHRASED)

1. Make systematic, continuing analysis of all fatigue failure experiences, and make wide distribution of detailed data to all designers.

2. Determine effects of automation in manufacturing processes on repeatability of fatigue test results.

3. Study fundamental fatigue phenomena - misalignment of atoms (dislocations, vacancies, etc.) and energy necessary to create and propagate cracks.

4. Investigate pressure fluctuations and related fatigue loadings in local areas due to shock waves and sonic loads.

5. Develop non-destructive fatigue detecting techniques.

6. Expand research on spectral techniques for describing fatigue loads and predicting structural life.

7. Translate and consolidate fatigue literature for practicable use by the design engineer.

8. Develop automated procedures and load programming for fatigue tests, including data reduction systems.

9. Study biaxial and triaxial stress/strain in elastic and plastic realms and their relation to fatigue.

10. Establish the methods and data bases for determining allowable design stresses.

FIGURE 10

SYMPOSIUM QUESTIONNAIRE

QUESTION NO. 9 RESULTS - SUGGESTED SOURCES FOR STRUCTURAL RESEARCH

No.	Name	Organization	Location
1.	Babington, W.	Bell Telephone Laboratories	New York 14, N.Y.
2.	Bean, Dr. W. T.	Research Consultant	Detroit, Mich
3.	Bondat, Dr. Julius	Ramo-Wooldridge Corp	Los Angeles 45, Calif
4.	Bradley, Wilson Jr.	Endevco Corp	Pasadena, Calif
5.	Brown, W. G.	Propulsion Test Facility,Inc.	New Haven, Conn
6.	Butler, J.P.	Boeing Airplane Co	Seattle 24, Wash
7.	Carleton, R. J.	Gilmore Industries	Cleveland 3, Ohio
8.	Choo, Dr. Pei	Drexel Institute of Technology	Philadelphia, Pa
9.	Clark, Dr. D. S.	California Institute of Tech	Pasadena, Calif
10.	Clemett, Harold	Continental Aviation & Engr. Corp	Detroit, Mich
11.	Conover, G. B.	Propulsion Test Facility, Inc.	New Haven, Conn
12.	Dean, W. J.	Temco Aircraft Corp	Dallas 22, Texas
13.	De Money, Dr. F. W.	Kaiser Aluminum Co	Spokane, Wash
14.	Dolan, T. J.	University of Illinois	Urbana, Ill
15.	Dorn	University of California	Berkley, Calif
16.	Dranetz, A.	Gulton Industries, Inc	Metuchen, N. J.
17.	Dresselhouse, D. E.	Chrysler Missile Division	Detroit 31, Mich
18.	Duell, Dr. A. J.	Armour Research Foundation	Chicago, Ill
19.	English, Dr. Morley	University of California, Los Angeles	Los Angeles, Calif
20.	Farrell, J. W.	Temco Aircraft Corp	Dallas 22, Texas
21.	Findley, W.	Brown University	Providence, R. I.
22.	Fronkin, N. G.	Propulsion Test Facilities, Inc	New Haven, Conn
23.	Fruedenthal, A. F.	Columbia University	New York, N. Y.
24.	Gadd, Charles	General Motors Research Center	Detroit, Mich
25.	Gentleman, Frederick	McDonnell Aircraft Corp	St. Louis 66, Mo
26.	Gerner, R.	Gulton Industries, Inc	Metuchen, N.J.
27.	Girard, George	New York University	New York 53, N.Y.
28.	Goland, Leonard	Kellett Aircraft Co	Willow Grove, Pa
29.	Goland, Martin	Southwest Research Inst	San Antonio, Texas
30.	Hample, W. G.	Hughes Tool Co	Culver City, Calif
31.	Hyler, Walter S.	Huck Manufacturing Co	Detroit 7, Mich
32.	Klein, E.	Propulsion Test Facility, Inc	New Haven, Conn.
33.	Krug, Maurice	Technology Incorporated	Dayton 2, Ohio
34.	Lazan, Dr. B. J.	University of Minnesota	Minneapolis, Minn
35.	Lemcoe, M.M.	Southwest Research Inst	San Antonio, Texas
36.	Levy, Alan	Hughes Tool Co	Culver City, Calif
37.	Lowell,	Adelphia College	Garden City, N. Y.
38.	Lupfer, D.	Gulton Industries, Inc	Metuchen, N. Y.
39.	Marin, Dr. J. W.	Pennsylvania State College	State College, Pa
40.	Matson, R. L.	General Motors Tech. Center	Detroit, Mich
41.	McDowell, Dr. E.L.	Armour Research Foundation	Chicago, Ill
42.	Meyer, John H.	Atom Apply	St. Ann, Mo
43.	Morrow, J.	University of Illinois	Urbana, Ill
44.	Murry,	Massachusetts Inst of Tech.	Cambridge, Mass
45.	Newman, Malcolm	Republic Aviation Corp	Long Island, N. Y.
46.	Niemier, B.A.	Reynolds Metals Co	Richmond, Va

47.	Odesskt, L.	Propulsion Test Facility, Inc	New Haven, Conn
48.	Poehle, Fred	Brooklyn Polytechnic Inst	Brooklyn, N. Y.
49.	Rhodes, James E.	Endevco Corp	Pasadena, Calif
50.	Robertshaw, A.	Hughes Tool Co	Culver City, Calif
51.	Rodden, W. P.	Norair	Hawthorne, Calif
52.	Rowe, Dr. Robert S.	Duke University	Durham, N. C.
53.	Schjelderup, Dr. H.C.	National Engineering Science Co	Pasadena, Calif
54.	Shanley, F. R.	University of California, Los Angeles	Los Angeles, Calif
55.	Sinclair, G. M.	University of Illinois	Urbana, Ill
56.	Smith, Henry G.,Jr.	Hughes Tool Co	Culver City, Calif
57.	Smith, Ronald H.	Norair	Van Nuys, Calif
58.	Spangler, C.J.	Boeing Airplane Co.	Seattle 24, Wash.
59.	Sylvestiowicz, W.	Bell Telephone Laboratories	New York 14, N.Y.
60.	Tatnall, Frank	Tatnall Measuring Systems Co	Phoenixville, Pa
61.	Truit, Dr. R. W.	Virginia Polytechnic Inst.	Blaksburg, Va.
62.	Waisman, J.L.	Douglas Aircraft Co. Inc	Santa Monica, Calif
63.	Waters, Peter	Republic Aviation Corp	Long Island, N.Y
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73.		Polyphase Instrument Co	Bridgeport, Pa

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